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FINAL REPORT 5

SYSTEM DESIGN AND SPECIFICATIONS

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY (EOS)

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

ONE SPACE PARK • REDONDO BEACH, CALIFORNIA 90278

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SPECIFICATIONS

SP-1	Earth Observatory Satellite System
SP-11	Observatory Segment
SP-312	Central Data Processing Facility
SP-313	Low-Cost Ground Station
SP-1111	Spacecraft Structure Assembly
SP-1112	Communications and Data Handling Module
SP-1113	Electrical Power Module
SP-1114	Attitude Determination Module
SP-1115	Actuation Module
SP-1116	Solar Array and Drive Module
SP-1124	Wideband Communications Module

1. INTRODUCTION

The capability of the EOS program to produce a flexible, multi-purpose, low-cost spacecraft depends greatly on the thoroughness of tradeoff and design efforts and the accuracy with which this information is conveyed to the ultimate system implementor. EOS Study Report 3, and its appendices, present our design (Figure 1) and the tradeoffs and analyses which led to it. The specifications we present herein that define our design can be used as a basis for procurement.

Throughout the design effort and specification preparation, we have emphasized modularity and the flexibility which arises from configuring various systems from basically standard modules. We have incorporated a means for on-orbit servicing and techniques for easily obtaining growth. As an example of how these specifications reflect design points, we require that space be left for greater redundancy in all modules, for larger wheels in the actuation module, for more batteries in the power module, and for more computer memory in the communications and data handling module. Our data bus is specified so that up to 32 separate modules can attach to it. Our thermal design is based on near-isolation between the modules so it is essentially independent of the number or types of modules or instruments included.

A brief summary of our design follows to aid in interpreting the specifications. Other than the thematic mapper and HRPI specifications submitted in July (Report 2), we have not detailed the payload section of the Observatory. We have specified constraints on any future instrument for incorporation in a payload. These include size, weight, power, and magnetic cleanliness as well as interfaces with the data bus, payload structure, and launch environment.

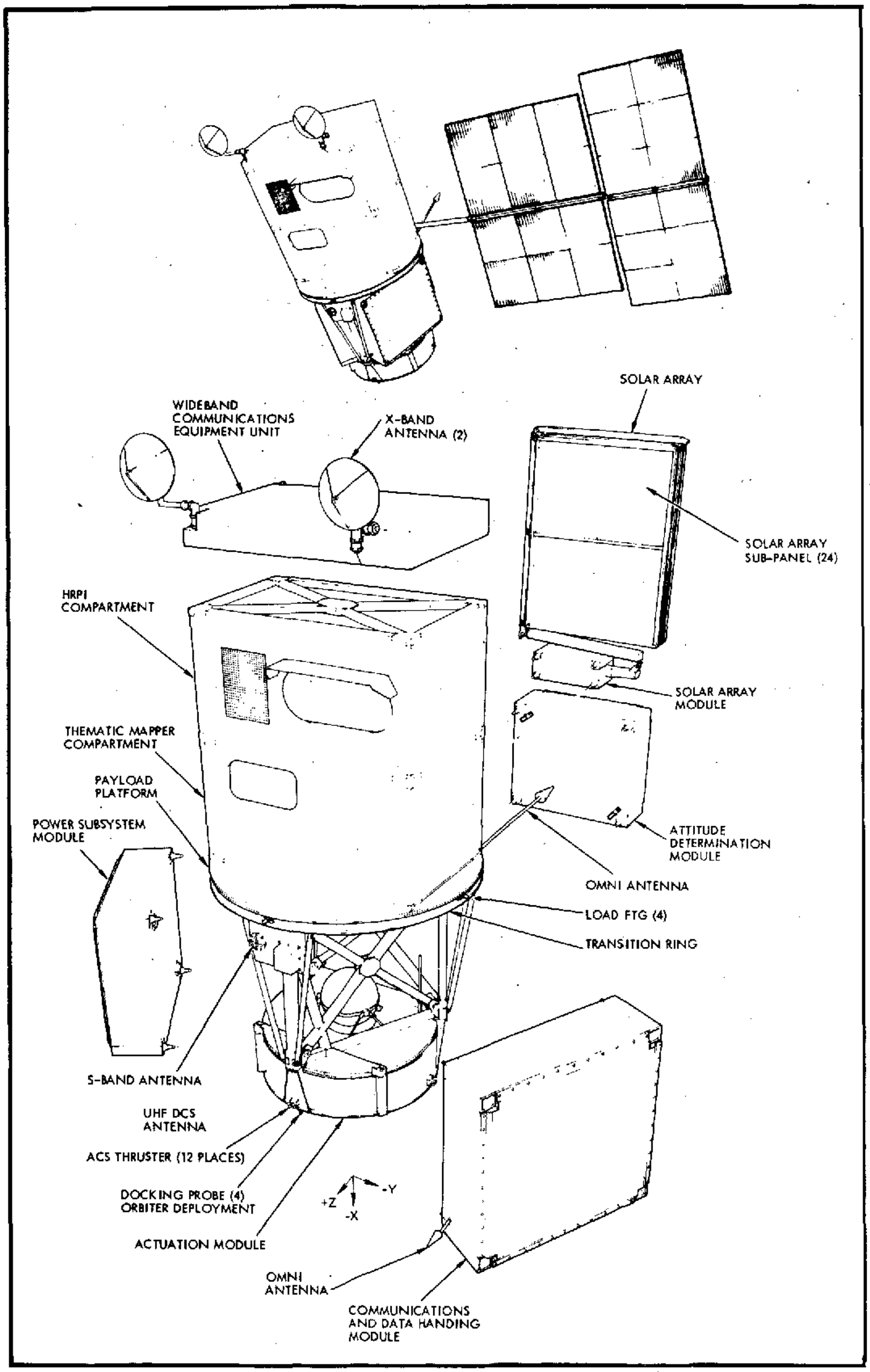

Figure 1. EOS Baseline Configuration

2. DESIGN SUMMARY

The EOS system provides a capability to support a wide variety of NASA missions. The Observatory, consisting of the complete spacecraft and instrument payload, can be launched into a mission-specified orbit by the Space Shuttle or an expendable launch vehicle from either ETR or WTR. The Central Data Processing Facility (CDPF), operating through the OCC and the designated STDN stations, receives mission data and processes the data into useful formats for user application. The Operations Control Center (OCC), operating through designated STDN stations, provides Observatory status telemetry monitoring, tracking, and ground command. Low-cost ground stations (LCGS) receive, record, and process selected mission data at a reduced bandwidth from the wideband downlink.

The baseline integration and test concept planned for EOS-A is a progressive departure from the conventional approach. It is based upon a single contractor performing the integration and test functions for both modules and Observatory. In general, the baseline concept calls for extensive qualification testing to validate the modular design parameters and limited acceptance testing at the spacecraft and Observatory level. In this way, cost savings over the conventional approach are realized over a large number of observatories and mission applications. An overview of the EOS-A baseline and the conventional test program is given in Table 1 and detailed in Section 7 of Report 3.

2.1 OBSERVATORY

The EOS modular concept, Figure 1, has been implemented with mission flexibility, low cost, and commonality as major criteria. The general Observatory configurations each consist of a spacecraft section including a transition ring and a forward payload section which is designed to be readily altered by reconfiguring its basic structure for accommodation of various payloads.

The baseline spacecraft portion of the Observatory has five standard modules plus a structure assembly to collect these modules. These modules are:

- Electric power module, containing the power control unit and sufficient batteries to support the mission payload

- Communication and data handling module, with receivers, transmitters, omni antennas, and computer

- Attitude determination module, with star trackers, inertial reference unit, magnetometer, and sun sensor

- Actuation module, with momentum wheels, magnetic torquers, nitrogen attitude-control system, and hydrazine orbit maintenance system. The capacities of the elements of this module are payload-dependent.

- Solar array and drive module, containing array drive and drive electronics and supporting a mission-dependent solar array

Table 1. Overview of EOS Baseline and Conventional Test Programs

	EOS Baseline	Conventional
Qualification		
Component (unit)	Functional Random vibration (3 axes) Thermal vacuum EMI/EMS (1)	Functional Sine and random vibration (3 axes) Thermal vacuum or solar simulation Shock EMI/EMS
Subsystem (module)	Functional Acoustics Thermal vacuum EMI/EMS	Functional (2) Thermal (2)
Spacecraft	Payload interface checks	Functional Payload interface checks
Observatory	Functional Low frequency sine vibration (3 axes) Acoustics Ordnance firing shock Thermal vacuum Electromagnetic compatibility	Functional Sine and random vibration (3 axes) Modal survey Acoustics Ordnance firing shock Solar simulation and/or thermal vacuum Electromagnetic compatibility Static load
Acceptance		
Component (unit)	Functional Random vibration (3 axes) Thermal vacuum	Functional Sine and random vibration (3 axes) Thermal vacuum
Subsystem (module)	Functional Acoustics (3) Burn-in (3)	Functional
Spacecraft	None	Functional Payload interface tests
Observatory	Functional (4) Acoustics (4) Burn-in (4)	Functional Acoustics or vibration Thermal vacuum Burn-in (5)

NOTES: (1) Engineering data only
(2) Large integrated payloads only
(3) Refurbish items only
(4) Complete Observatory launch
(5) Special cases (i.e., storage, launch delay, or suspected infant failure)

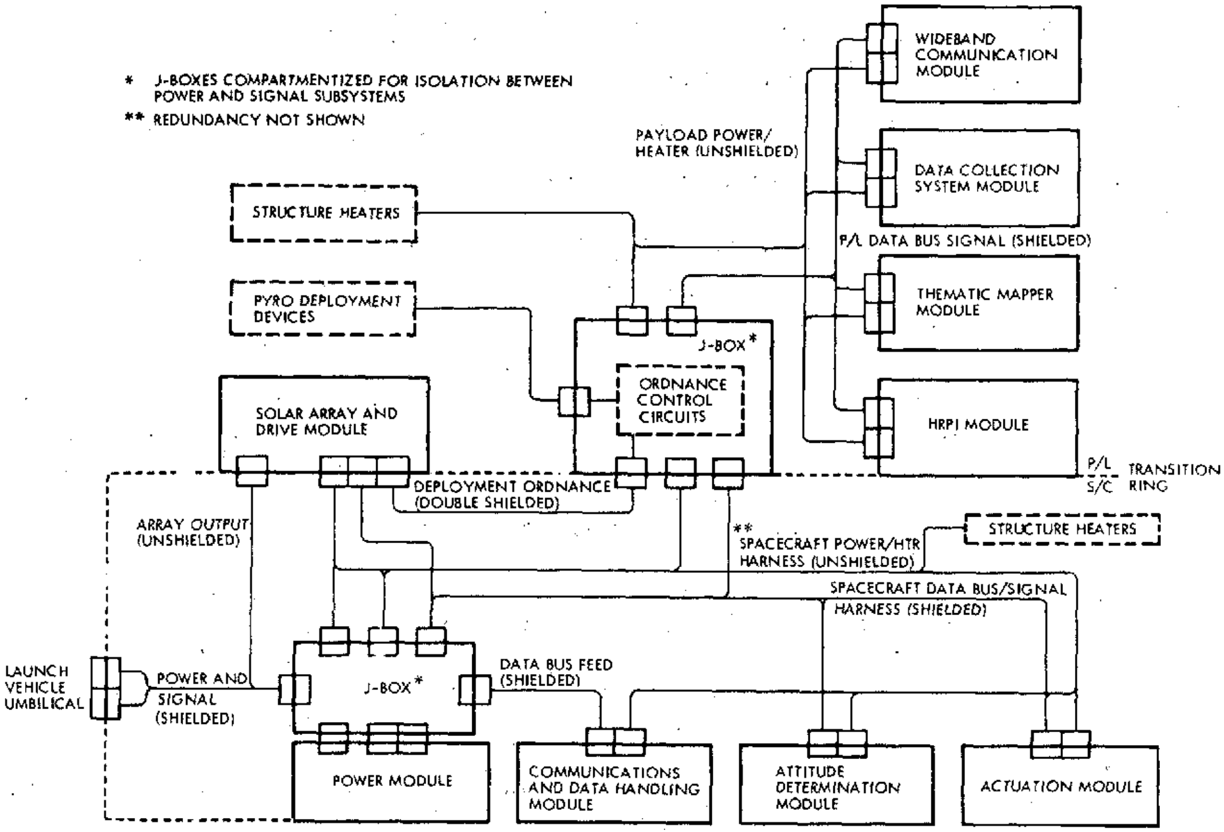

Figure 1. Baseline Modular Block Diagram

Each of the modules is structurally and thermally independent and removable from the spacecraft by means of Shuttle-SPMS compatible latches.

The first three standard modules are structurally similar with varied internal configurations. The actuation and solar array and drive modules differ in configuration because their size is payload dependent. While these modules, together with the structure comprise the spacecraft, they are not required to contribute to fundamental spacecraft structural integrity. The design is detailed in Sections 1 through 6 of Report 3.

2.1.1 Spacecraft Structure

The spacecraft, that part of the Observatory from the transition rings aft, includes the solar array module located just forward of the transition ring. The spacecraft has a structure, a group of subsystem modules, and mechanisms to release, separate, and deploy various spacecraft and payload devices.

The baseline EOS-A spacecraft design and test goals are to minimize the need for detailed analytical load predictions through generous safety margins, and to replace costly static and modal survey testing with economical and effective three-axis sinusoidal base excitation tests.

The baseline structural configuration has two major sections: the primary structure and the equipment modules. The structure has four major subassemblies:

1) Equilateral triangular mainframe outlined by three longerons projecting aft from the plane of the transition ring
2) Aft stiffened cylindrical shell structure which mounts the forward section of the separation joint
3) An I-section transition ring and sandwich bulkhead in the plane of the ring, plus load fittings for Shuttle compatibility
4) Six struts joining the transition ring to the forward end of the aft cylindrical shell structure.

The EOS-A baseline design has five subsystem modules. Three modules, the power, communications, and attitude determination, are similarly configured: each is 48 x 48 x 18 inches deep with square corner tubes for latching mechanism mounting. The actuation module includes hydrazine and nitrogen systems, and is configured to occupy available volume within the spacecraft structure. The solar array module, which includes the solar array drive assembly and associated equipment, is located above the transition ring because of the location of the array itself.

The EOS-A baseline structures use aluminum as the primary structural material. In local areas, titanium and/or fiberglass is used for thermal isolation since the thermal conductivity of these materials is significantly less than aluminum. The design is detailed in Sections 6.8 and 6.9 of Report 3.

2.1.2 Electrical Power Module

The electrical power module (EPM) regulates primary power, provides fault protection, and stores energy for the spacecraft and the payload. Primary unregulated power is routed to the EPM from the solar array and drive module by the spacecraft harness. Common point failures due to shorts in the raw-power harness are prevented by isolation diodes. The central ground point is in the EPM.

Major EPM elements (Figure 2) are the data interface unit, power control unit, battery assembly, secondary power and bus protection assembly, diode assembly, and power disconnect assembly.

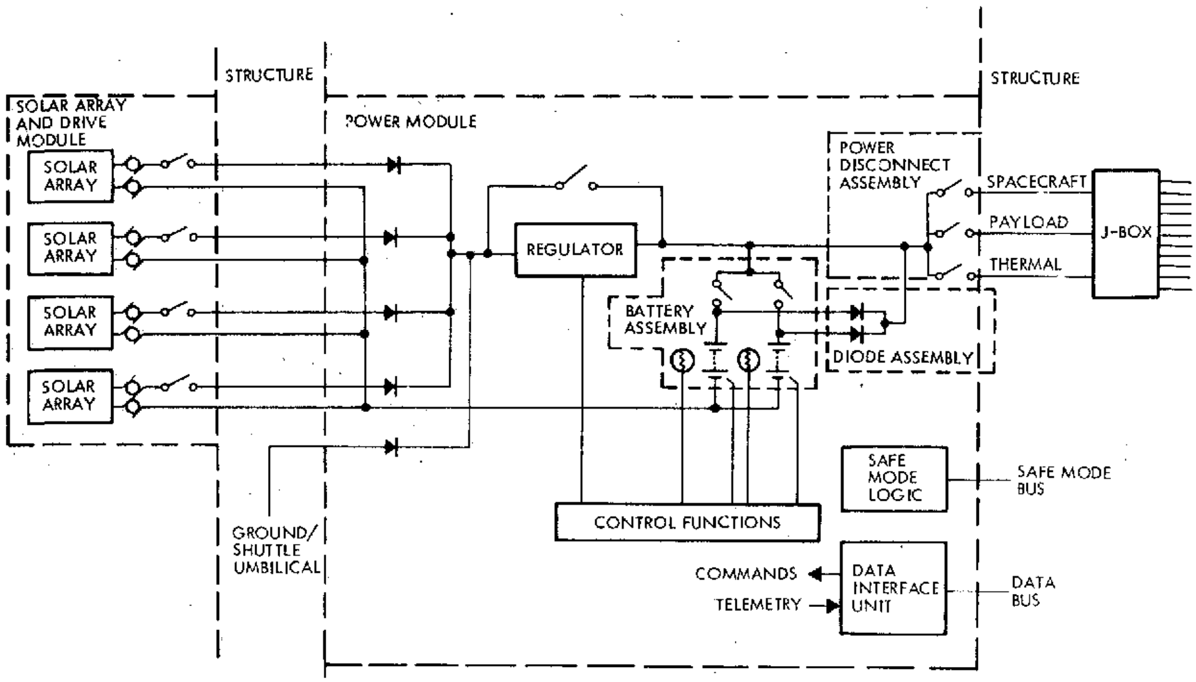

Figure 2. Power Module and Interface Block Diagram

The power control unit controls the battery state-of-charge throughout the three orbital charge/discharge routines: eclipse, initial sunlit period, and final sunlit period. During normal operations, battery enable/ disable switches and power disconnect switches are closed; during abnormal conditions the appropriate switch is opened. Main bus power is controlled by a pulsewidth modulation buck regulator. Fault sensing circuitry controls redundant regulator switching and provides safe mode enable signals.

Power conversion in all basic modules is accomplished with one of two standard designs. One design yields a total power output of 40 watts and the other an output of 80 watts. Both deliver their power at ±5, ±10, ±15, or +28 volts. Each output is regulated to 5 percent over the range of input-voltage and load variations.

Diodes are provided in the module for isolation of faults and battery power control. The solar array bus and the launch/Shuttle umbilical bus are connected to the main bus through diodes to avert a single-point failure due to a short in the unregulated power harness. The availability of battery power to the main bus is ensured by diodes which are in parallel with the charge enable/disable switches. These switches may be operated in the event of a battery cell fault or abnormal temperature conditions.

The power disconnect assembly contains circuitry necessary to remove all input and output power from the module, and to automatically open individual lines in the event of a fault. Circuitry consists of relays that are controlled by external commands and that open automatically in the event excess current is flowing in the circuit. Each relay is a hermetically sealed magnetic latching relay with two sets of contacts. The main set of contacts (single-pole, double-throw, double-break) are rated at 50 amperes and auxiliary contacts (single-pole, double-throw) are rated at 5 amperes.

The EPM uses nickel-cadmium batteries to provide supplemental power during eclipse and peak loads.

Ground/Shuttle umbilical interfaces to the EPM allow total control of power, so power can be removed from any and all modules within the Observatory during Shuttle refurbishment operations.

The design is detailed in Section 6.4 of Report 3.

2.1.3 Communications and Data Handling Module

The communications and data handling (CDH) module receives information from, and transmits information to, NASA ground stations. It provides the spacecraft with the capability to receive, demodulate, and process uplink command information; collect, process, and transmit housekeeping telemetry and medium-rate user data; coherently transpond range information; and centrally perform on-board computations. Finally, the CDH implements and controls the spacecraft data bus system.

The baseline CDH (Figure 3) provides cost-effective, reliable operation by using space-proven equipment and technology. Uplink data, transmitted from NASA STDN stations on a phase-modulated carrier within the 2050 to 2150 MHz range, is received by one of two omni-directional antennas having opposite senses of circular polarization. The earth-pointing antenna is fix-mounted; the other is boom deployed. Outputs from the two antennas are connected by diplexers to a hybrid combiner. Since the ground station transmits to the spacecraft with either left or right hand antenna polarization, it can be commanded via antenna diversity or polarization switching. The orthogonal antenna patterns guarantee uplink and downlink communications for all spacecraft orientations.

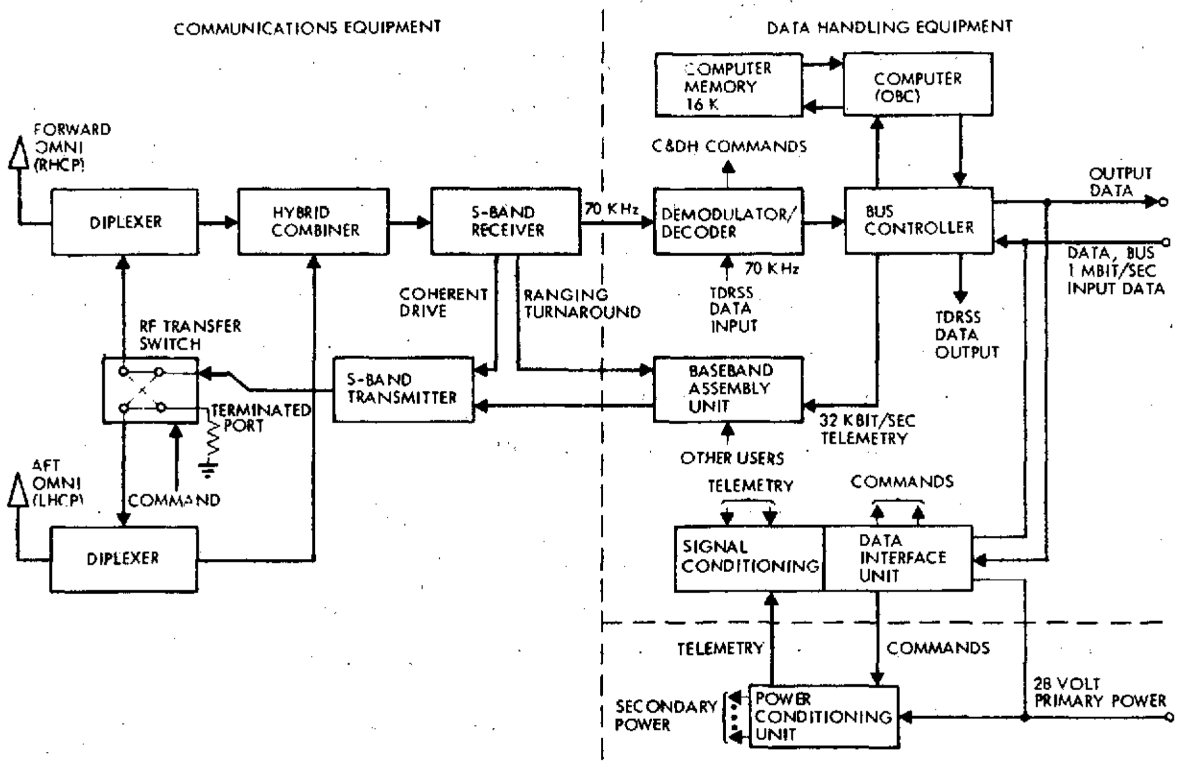

Figure 3. Communications and Data Handling Module Block Diagram

The CDH communications system uses S-band channels for command, telemetry, and ranging. The command channel receives a phase-modulated signal and demodulates it, providing an input to the data handling equipment. From the data handling equipment, the telemetry channel receives a composite signal consisting of 32 kbit/sec telemetry data biphase modulated onto a 1.024 MHz subcarrier, and a signal consisting of either 512 kbit/sec direct digital data or 500 kHz range data. This data is phase modulated and transmitted in the 2200 to 2300 MHz range.

The data handling equipment detects and decodes baseband command information; stores on-board commands; controls the on-board data bus; disseminates real-time or stored commands via the data bus; on-board computations; and assembles 32 kbit/sec housekeeping data, 512 kbit/sec medium-rate user data, and range data for transmission to the communications equipment. The design is detailed in Section 6.1 of Report 3.

2.1.4 Attitude Determination Module

The attitude determination module (ADM) provides an on-board attitude reference for a wide variety of orbits. The attitude determination hardware, built of existing state-of-the-art components, is used for any mission type, without redesign. Software functions are mechanized in the on-board computer. The ADM can incorporate added redundancy (e.g., three star trackers, six gyro assemblies) and/or special elements (e.g., a precision null-seeking sun sensor), if needed for certain missions. Minor changes, generally simplifications, in the on-board computer software accommodate missions which are not earth pointing.

The ADM, Figure 4, houses all of the equipment which provides sensor data to the on-board computer for control of the actuation module: inertial reference units, star trackers, magnetometer, sun sensor, transfer assemblies, data interface units, and star tracker shades.

During normal operation, the precision three-axis strap-down inertial reference unit provides the high degree of stability required for the broad spectrum of EOS missions. Drifts over long time spans are eliminated by periodically updating by means of body-fixed star trackers which give an absolute inertial reference. An on-board ephemeris model transfers inertial data into an earth-referenced coordinate system. The magnetometer senses earth's magnetic field to provide necessary data for control of the three orthogonal magnetic torquers in the actuation module which unload the reaction wheels.

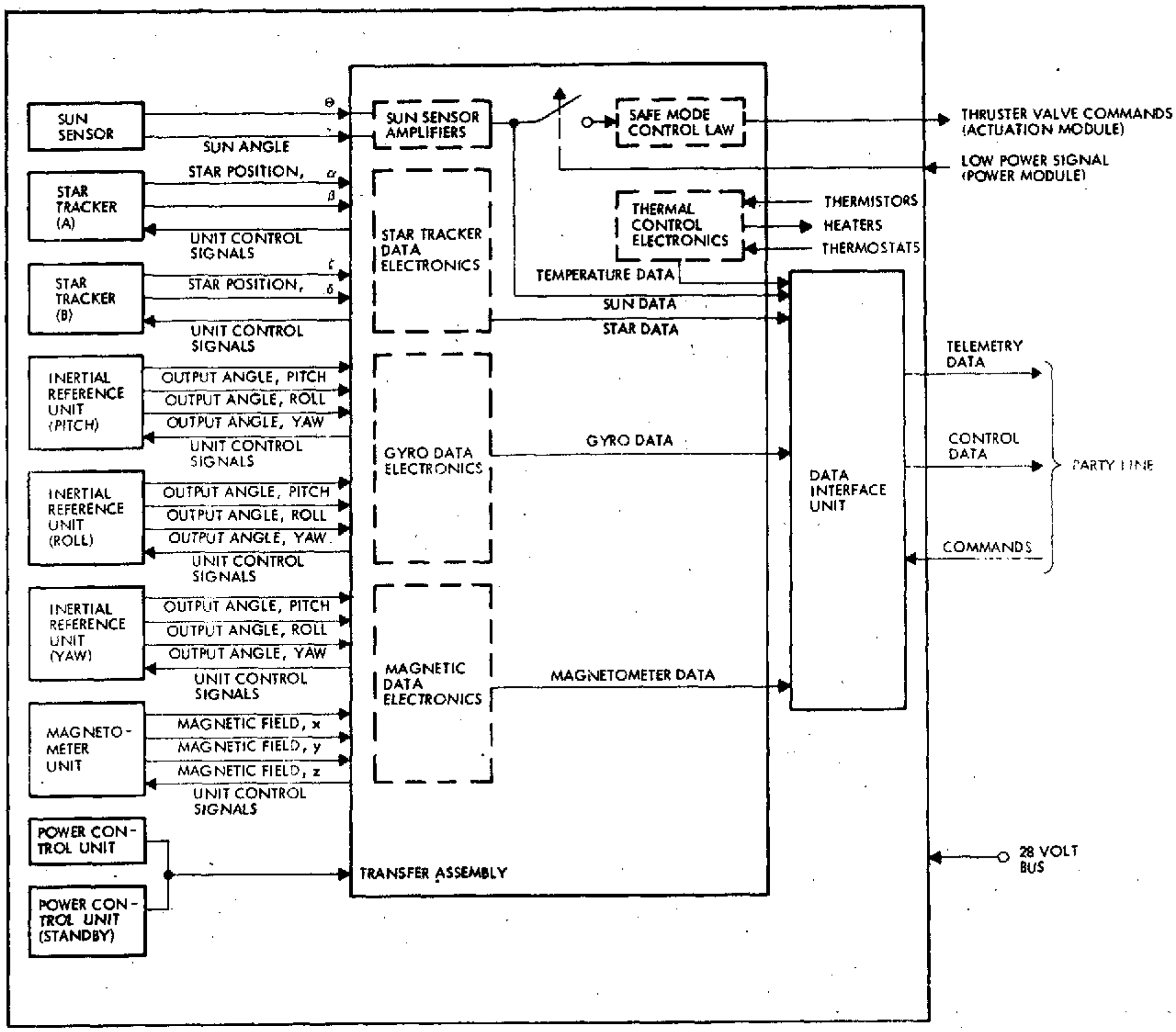

Figure 4. Attitude Determination Module Block Diagram

During the safe mode, the sun sensor provides data to automatically orient the spacecraft for maximum illumination of the solar array. Return from the safe mode is ground commanded. The design is detailed in Section 6.2 of Report 3.

2.1.5 Actuation Module

The mission-peculiar attitude and orbit control equipment — reaction wheels, torquers, a hydrazine orbit transfer/adjust propulsion system, and a cold gas nitrogen reaction control propulsion system — are integrated into the actuation module (Figure 5). The multi-mission actuation module

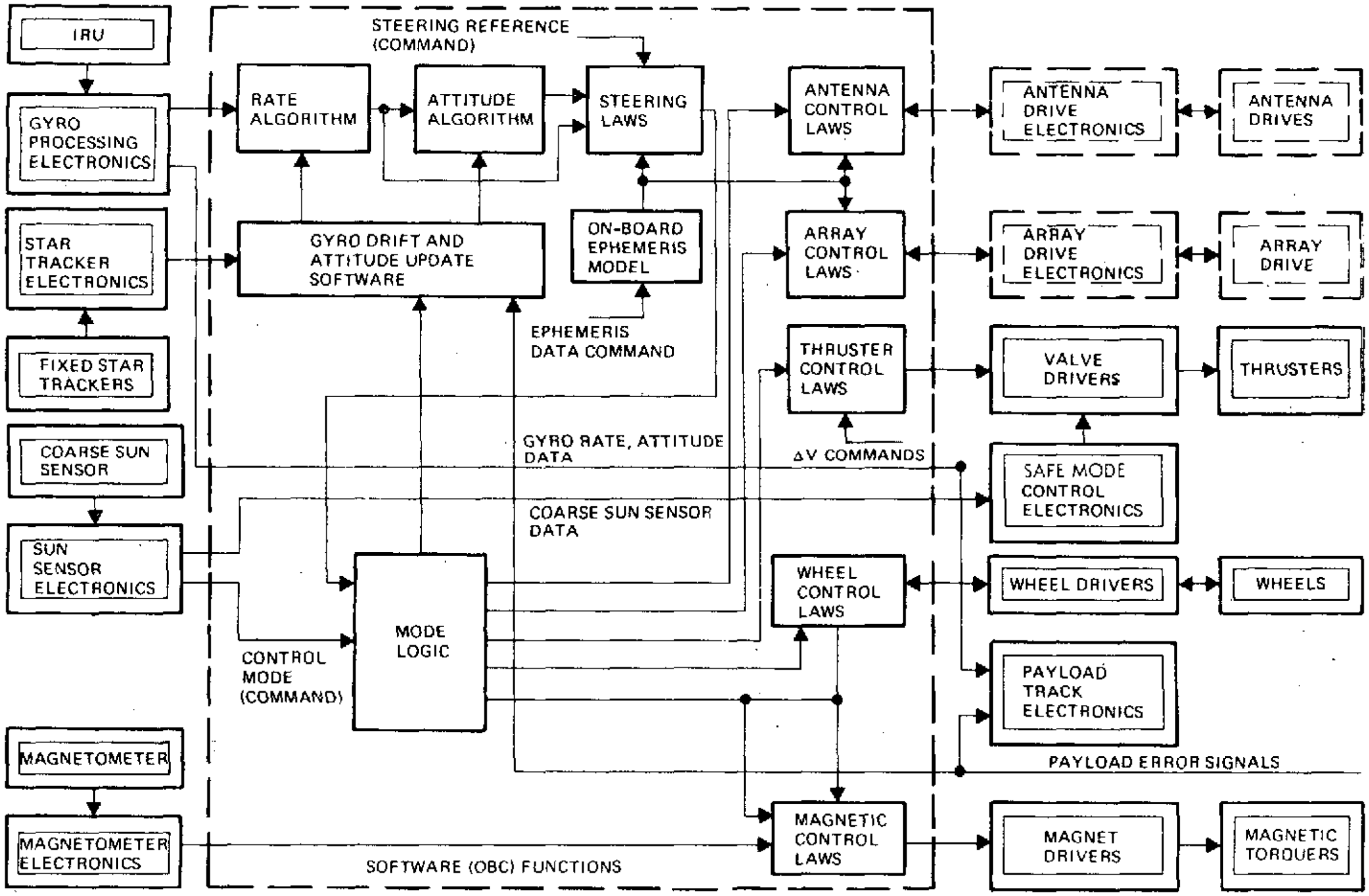

Figure 5. Attitude Control Functional Block Diagram

design accommodates a wide variety of missions by providing space and actuation electronics for all units within the specified range.

Attitude control torques for fine-pointing control during normal operation are obtained from a set of three orthogonal body-fixed reaction wheels. A set of three orthogonal electromagnetic torquers which interact with the earth's magnetic field to create control torques on the spacecraft prevents wheel momentum saturation. High-level cold-gas thrusters provide control torques during other modes of operation, such as acquisition and orbit-adjust engine firing, which entail relatively rapid reorientation and/or large disturbance torques. These thrusters can also operate in a low-level mode, and are used to control attitude during the safe mode.

Except for the safe mode and direct control via the payload, all control laws and other computational functions related to the attitude and orbit control are implemented by the on-board computer within the communications and data handling module. The on-board computer generates commands for the attitude control torquers (reaction wheels, magnets, thrusters) to achieve the desired spacecraft orientation.

During safe mode the gyros, reaction wheels, and magnetic torquers are disabled. The design is detailed in Section 6.3 of Report 3.

2.1.6 Solar Array and Drive Module

The solar array and drive module accommodates the variety of missions encompassed by the modular concept through a standard solar cell subpanel which can be assembled in various quantities. The array is attached to a solar array drive module containing a 1 degree of freedom drive mechanism, springs, and associated electronics (Figure 6), which is removable for on-orbit servicing.

Solar array power is transmitted through the subarray drive assembly and the power disconnect assembly to the power module. The solar array is electrically divided into four sections. Each section delivers power to the drive interface by means of a power line and a return line. The array drive contains both power-handling slop ring assemblies, and slip ring assemblies for solar array temperature telemetry signals. The slip ring assemblies each contain redundant brushes for maximum reliability.

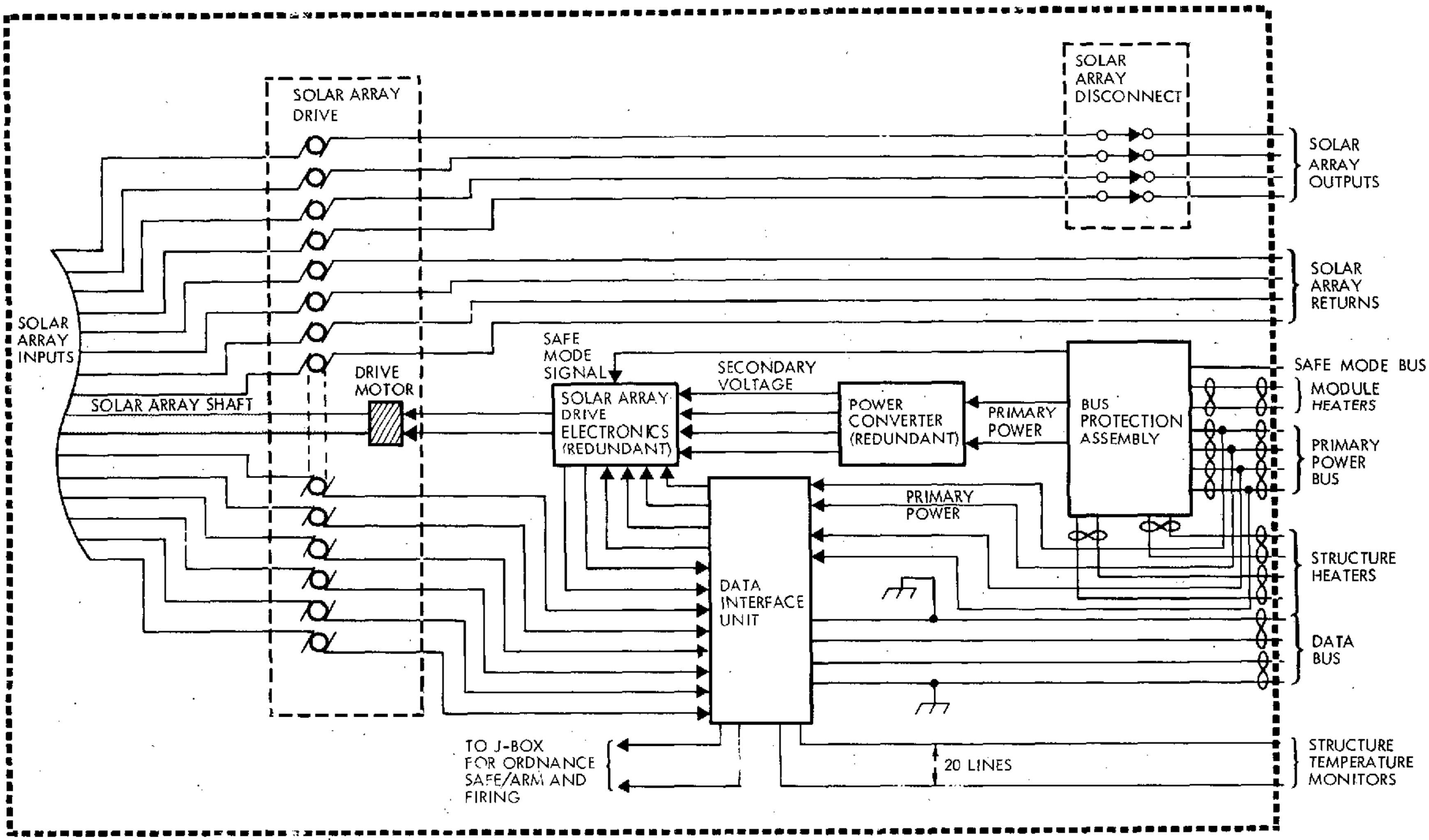

Figure 6. Solar Array and Drive Module Block Diagram

The baseline EOS solar array design consists of panels which fold together for stowage during launch. The basic building block of the various EOS solar cell arrays is a honeycomb subpanel upon which solar cells are bonded in a flat lay-down configuration. The primary power bus furnishes electrical power to operate the solar array drive, drive electronics, power converter and all other electronic functions within the module. The power disconnect assembly contains contactors for each power line to disconnect the solar array from the electrical distribution system in the event of spacecraft retrieval by the Shuttle. The contactors can be opened only by a command transmitted through the Shuttle umbilical connector. The design is detailed in Section 6. 6 of Report 3.

2. 2 WIDEBAND COMMUNICATIONS MODULE

The wideband communication module (WCM) is an instrument module which contains the digital RF equipment required to telemeter full frame TM and HRPI data to wideband data collection stations and selected thematic mapper (TM) data to low-cost ground stations. The baseline WBC module (Figure 7) consists of a data handling (digital) system, and a wideband communications (RF) system; it is discussed in Section 6. 7 of Report 3.

The wideband communications system modulates, amplifies, and transmits data received from the LCGS speed buffer and the TM and HRPI multiplexers. Two X-band links relay the sensor data to NASA STDN stations and to any selected one of the set of low-cost ground stations. The 395 MHz bandwidth allocated to these links is between 8.025 and 8. 400 gHz. The system accepts 20 Mbit/sec data from the speed buffer, and two 128 Mbit/sec data streams from the TM and HRPI modules. The combined TM and HRPI outputs are converted to quadriphase modulation at 256 Mbit/sec. The power amplifiers and X-band antennas for the 20 and 256 Mbit/sec channels are identical. Low-cost transfer switches ensure that either power amplifier and antenna may be used for either signal, increasing reliability at minimal cost.

The data handling system contains the LCGS speed buffer and the multi-megabit operational data system (MODS) controller, which controls and operates the high-speed PCM encoder and formatter contained in each

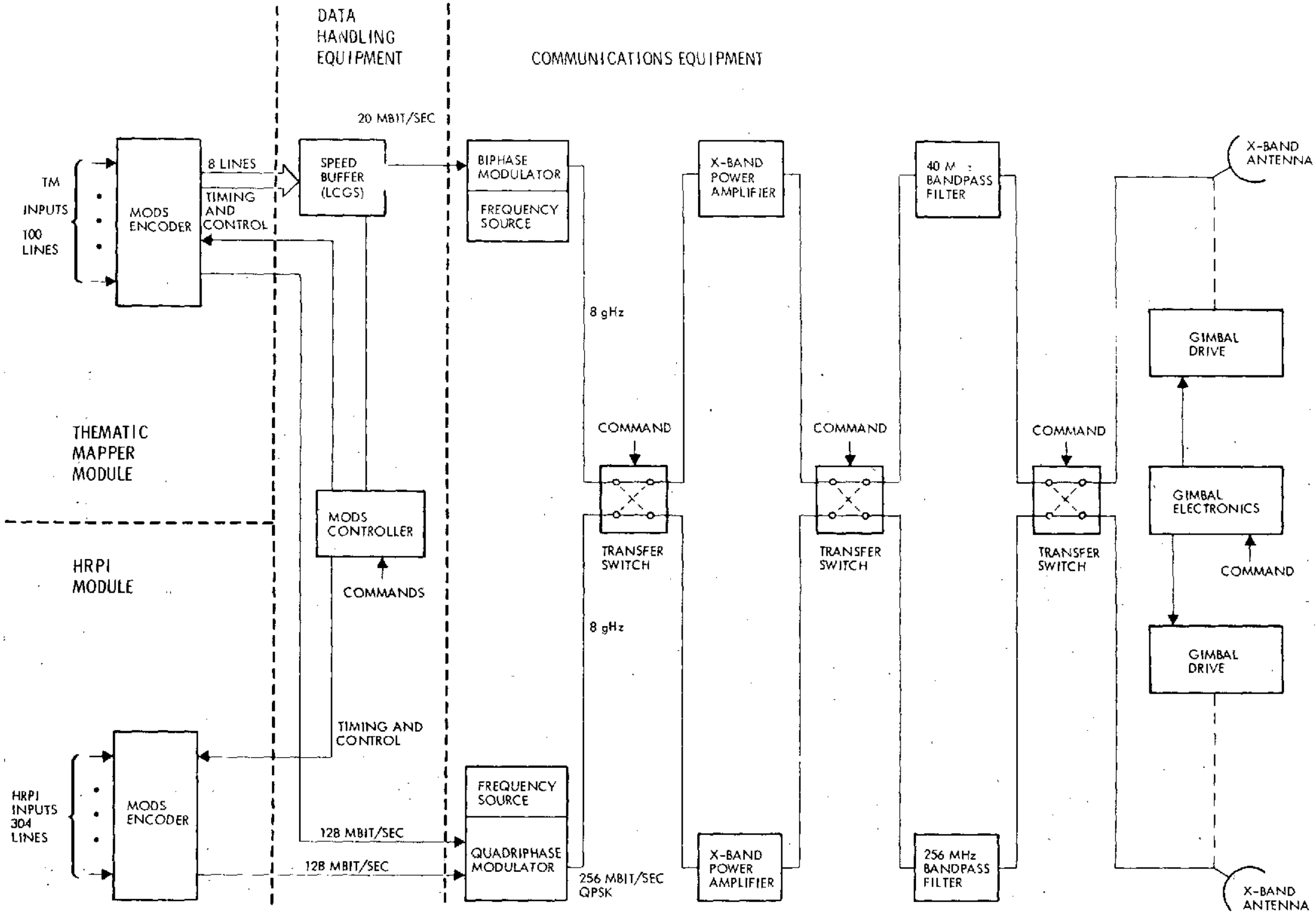

Figure 7. Typical Wideband Communication Module Block Diagram

instrument package. The system encodes, multiplexes and outputs a serial digital data stream to the transmitter modules. It also edits, reformats, compresses, or encodes the data when required.

2.3 CENTRAL DATA HANDLING FACILITY

The Central Data Processing Facility (CDPF) receives wideband sensor data from three receiving stations, and auxiliary spacecraft and calibration data from the Orbit Determination Group and control center. User and LCGS requests for coverage are processed and transmitted to the control center, and output film and magnetic tape products to NASA investigators and to the EROS data center for dissemination to other users.

The baseline design is three operating systems: Data Services Laboratory (DSL), Information Management System (IMS), and Image Processing System (IPS). Their functions are summarized in Figure 8.

The DSL is the user/EOS data interface; the IMS monitors and controls internal CDPF data handling. The baseline DSL/IMS uses existing Sigma 5 hardware with added core memory, all applicable ERTS IMS software, and the ERTS photo production facility.

The IPS handles image corrections and formatting, and output product generation. The Xerox 550 system is the basic building block for IPS hardware. Certain new hardware components such as laser beam recorders and high density digital tape generation equipment are required.

Cost effectiveness, crucial to EOS program survival, is stressed in the design philosophy:

- Use existing capabilities – ERTS hardware and software, EROS data center, ERTS foreign ground stations
- Construct two critical throughput algorithms in special-purpose hardware, the remainder of the image processing system in software for flexibility
- Structure all image processing functions except distortion estimation as parallel input/output processes
- Minimize manual interactions during processing
- Use standard processing modules, adaptable to future EOS missions.

The design is detailed in Section 9.3 of Report 3.

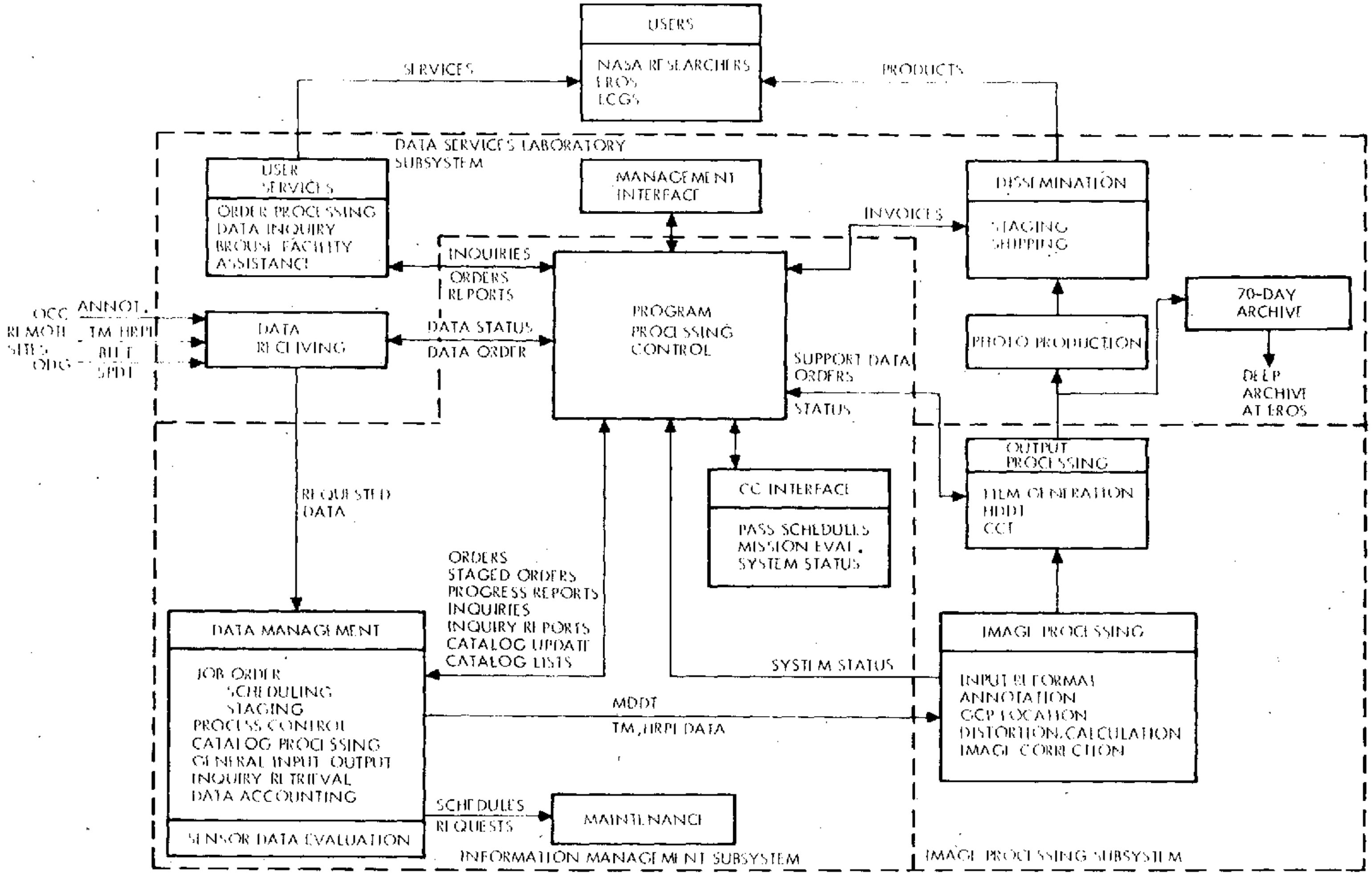

Figure 8. Central Data Processing Facility Functional Structure

2.4 LOW-COST GROUND STATION

The Low-Cost Ground Station (LCGS) quickly provides earth observation data to users in a format compatible with their unique needs. The baseline design (Figure 9) uses the same acquisition equipment and NASA operational interface to support alternative image processing subsystems: the direct display, and the record and process. The lower cost direct display produces film output products recorded in real time. The range of output products available by recording data in real-time, and processing later with a mini-computer-based system, however, results in data tailored to the users' needs and commensurate quality to the central data processing facility (CDPF) product.

X-band downlink thematic mapper sensor data, edited on board, is acquired by an 8-foot parabolic dish mounted on a paper-tape programmed tracking pedestal, a cost-effective design approach that accommodates the narrow beamwidth. The remaining RF equipment – parametric amplifier, X-band biphase demodulator, bit synchronizer, decommutator – are existing current inexpensive technologies from programs such as ERTS.

The LCGS, which receives only thematic sensor data, can acquire up to 10 scenes in one pass. The principal NASA/LCGS interface is within the CDPF, which also supports siting and equipment calibration, and provides requested technical data.

The design is detailed in Section 9.1 of Report 3.

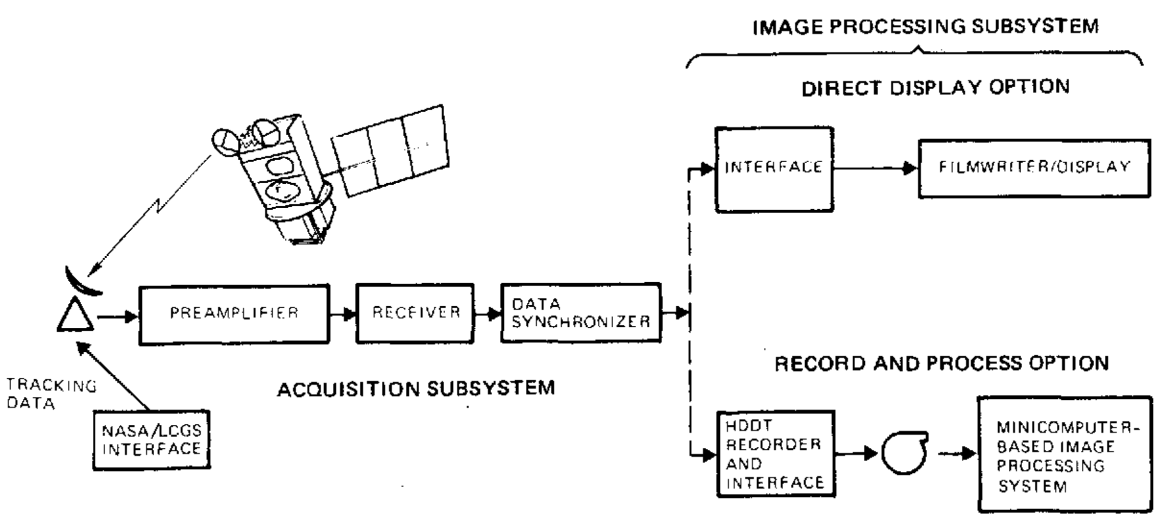

Figure 9. LCGS Functional Block Diagram

TITLE

PRIME ITEM DEVELOPMENT SPECIFICATION
FOR THE

EARTH OBSERVATORY SATELLITE SYSTEM

DATE 20 SEPT 1974 NO. SP-1

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
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SPECIFICATION SP-1

EARTH OBSERVATORY SATELLITE SYSTEM

1. SCOPE

This specification establishes the performance, design, development and test requirements for the Earth Observatory Satellite System.

2. APPLICABLE DOCUMENTS

2.1 Government Documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superceding requirement.

SPECIFICATIONS

NASA

SP-11	Observatory Segment Specification
SP-21	Launch Segment Specification
SP-31	Ground Segment Specification
SP-115	Observatory Environmental Criteria Specification
SP-311	Operations Control Center Specification
SP-312	Central Data Processing Facility Specification
SP-313	Low Cost Ground Station Specification
SP-314	STDN Stations Specification
SPM-XX	Mission Requirements Specification

Military

MIL-Q-9858A	Quality Program Requirements

STANDARDS

Military

MIL-STD-130	Identification Marking of U.S. Military Property

MIL-STD-143B	Standards and Specifications, Order of Precedence for Selection
MIL-STD-454C	Standard General Requirement for Electronic Equipment
MIL-STD-470	Maintainability Program Requirements (For Systems and Equipments)
MIL-STD-749B	Preparation and Submission of Data for Approval of Nonstandard Electronic Parts
MIL-STD-882 15 Jul 1969	System Safety Program for Systems and Associated Subsystems and Equipment, Requirements for
MIL-STD-1472A	Human Engineering Design Criteria for Military Systems, Equipment, and Facilities

OTHER PUBLICATIONS

NASA

SL-E-0002	Electromagnetic Compatibility Control Plan
EOS-3.3-1	Program Authorized Parts List
EOS-3.3-2	Program Authorized Materials List
EOS-3.3-3	Program Authorized Processes List
EOS-3.3-4	Electromegnetic Compatibility Control Plan
EOS-3.3-5	EMI/EMS Limits and Test Methods
EOS-3.3-6	Marking of Parts and Assemblies
EOS-3.3-7	Safety Design Criteria

2.2 Non-government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. If no revision or date is shown, the latest released issue of the applicable document shall apply. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superceding requirement.

EOS 4.1	System Effectiveness Program Plan
EOS 4.2	Integrated Test Plan

3. REQUIREMENTS

3.1 System Definition. The Earth Observatory Satellite system is composed of the following system segments:

(a) Contractor-furnished configuration items

(1) Observatory segment

- Spacecraft and transition ring
- Instrument payload and ancilliary equipment
- Aerospace ground equipment
- On-board computer software.

(2) Ground segment

- Operations Control Center (OCC) equipment
- Central Data Processing Facility (CDPF) equipment
- Low-cost ground station equipment
- STDN stations equipment.

(b) Government furnished items, facilities, and services which interact with contractor furnished configuration items:

(1) Launch vehicles and fairings

- Expendable launch vehicles as may be specified in the Mission Requirements Specification, SPM-XX.
- Space Shuttle system as may be specified in the Mission Requirements Specification, SPM-XX, for launch, retrieval from orbit, or on-orbit servicing operations.

(2) Western Test Range or Eastern Test Range launch facilities and services as specified for the mission in the Mission Requirements Specification, SPM-XX.

(3) Low-Cost Ground Station

(4) Operations Control Center

(5) Designated STDN stations.

3.1.1 General description. The EOS system concept provides for a flexible capability to support a variety of instrument payloads for NASA missions using the same basic modular spacecraft as a platform. The spacecraft modules and assemblies are intended to provide for a flexibility of choice in the complement of functional equipment to satisfy the peculiar requirements for each mission.

The Observatory, consisting of the complete spacecraft and instrument payload, will be capable of being launched into a mission specified trajectory by the Space Shuttle or one of the following expendable launch vehicles:

(a) Thor-Delta

(b) Titan III B

(c) Titan III D

(d) Titan III C

(e) Titan III E - Centaur

Launches may be from ETR or WTR, depending on the final orbit inclination specified. Circular orbit altitudes will be specified for each mission and may be in the range from low-earth orbit >300 n mi to synchronous altitude.

The observatory design will provide for either retrieval or on-orbit module replacement by the Space Shuttle system in the case of low-earth orbit missions.

The Operations Control Center (OCC) operating through designated Space Tracking and Data Network (STDN) stations will provide for status telemetry monitoring, tracking, and ground command control of the Observatory.

The Low-Cost Ground Station (LCGS) will receive, record, and process selected mission data from the Observatory at a reduced bandwidth from the wideband downlink.

The Central Data Processing Facility (CDPF) operating through the OCC and the designated STDN stations receives mission data and processes the data into useful formats for user application.

Foreign users may receive mission data when the Observatory is programmed to transmit such data by the OCC at times when the Observatory is within line-of-sight of the foreign user ground station.

A data collection subsystem may be specified as part of the Observatory payload on some missions. When so equipped, the Observatory may perform the function of relaying data from remote sensors to data collection subsystem users during the time the Observatory is within line-of-sight of both terminals.

When a tracking and data relay satellite (TDRS) is available, the observatory may be required to transmit telemetry and mission data and relay uplink commands to the observatory via the TDRS to designated ground stations. The TDRS may also perform the function of providing range and range rate tracking data to support observatory ephemeris determination.

3.1.2 Missions. A general objective of the EOS system is to provide flexibility of application in the use of a basic spacecraft platform which is capable of supporting a variety of mission payload instruments and ancilliary payload equipment to carry out a variety of NASA missions involving earth resource observations and scientific observations for various user agencies. The orbiting observatory will be modular in design so that an adaptive flexibility is provided to perform the specified missions through the use of standardized building block equipment components within the supporting subsystem modules and mission-peculiar payload modules with standardized spacecraft mounting provisions.

Examples of missions which may be implemented are shown in Table 1.

The specific mission requirements shall be as specified by the procuring agency in the applicable Mission Requirements Specification, SPM-XX

3.1.3 System diagrams. The system functional block diagram is shown in Figure 1 and the system segment relationships are shown in Figure 2.

3.1.4 Interface definition. The interface definition between EOS system segments shall be as specified in the Mission Requirements Specification, SPM-XX.

3.1.5 Government-furnished property list. The government-furnished property shall be as specified in the Mission Requirements Specification, SPM-XX

3.2 Characteristics

3.2.1 Performance. The EOS system shall be designed to orbit instrument payloads at altitudes from 200 to 1000 n mi at inclinations in the range of 28 degrees to sun synchronous and at or near geosynchronous altitude at inclinations in the range of 0 to 28 degrees.

The system shall provide for transmission of payload collected data which may be processed and transmitted in real-time or processed and stored for later dump to the ground segment.

The designated STDN stations and the LCGS shall receive the Observatory transmitted data and shall relay this data to the OCC and the

Table 1. Example EOS Mission Characteristics

Mission	Typical Payload Equipment	Typical Payload Support Requirements					Orbit and Launch Vehicle
		Weight (lb)	Volume (cu-ft)	Power (watts)	Data Rates	Attitude Control	
Land Resources Management (LRM) Implemented primarily with visible spectrum instruments for mapping and spectral analysis.	• 5-band multispectral scanner • Thematic mapper • Wideband tape recorder • Wideband data handling module	600	60	400	variable to 135 Mbit/sec	Earth pointing 36 arc sec accuracy	Low altitude, sun sychronous, 17-day trace repeat, (e.g., 386 n mi altitude) Thor-Delta launch
SEASAT Demonstrate space monitoring of ocean surface conditions	• Synthetic aperture radar • Passive microwave radiometer • Infrared imager • Data collection system • Wideband data handling module	500	600	500	variable to 10 Mbit/sec	Earth pointing 0.25 degree accuracy	Low altitude, non-sun synchronous (baseline is 39 n mi 82 degree inclined) Thor-Delta launch
Solar Maximum Mission (SMM) Study fundamental mechanism and effects of solar flares	• Ultraviolet magnetograph • EUV spectrometer • X-ray spectrometer • Hard X-ray imager • Low energy polarimeter	1430	13.5	175	5 kbit/sec	Sun pointing, 5 arc sec accuracy 1 arc sec drift/5 minutes	300 n mi altitude 33 degree inclined, Thor-Delta launch
Synchronous Earth Observatory Satellite (SEOS) Resource and weather monitoring from stationary platform; timely warming and alerts	• Large operture survey telescope • Microwave sounder • Framing camera • Atmospheric sounder and radiometer • Wideband data handling module	2640	350	145	60 Mbit/sec	Earth pointing with scan, 5 arc sec accuracy 1 arc sec drift/12 minutes	Geostationary launched by Shuttle/Tug or large expendable launch vehicle.

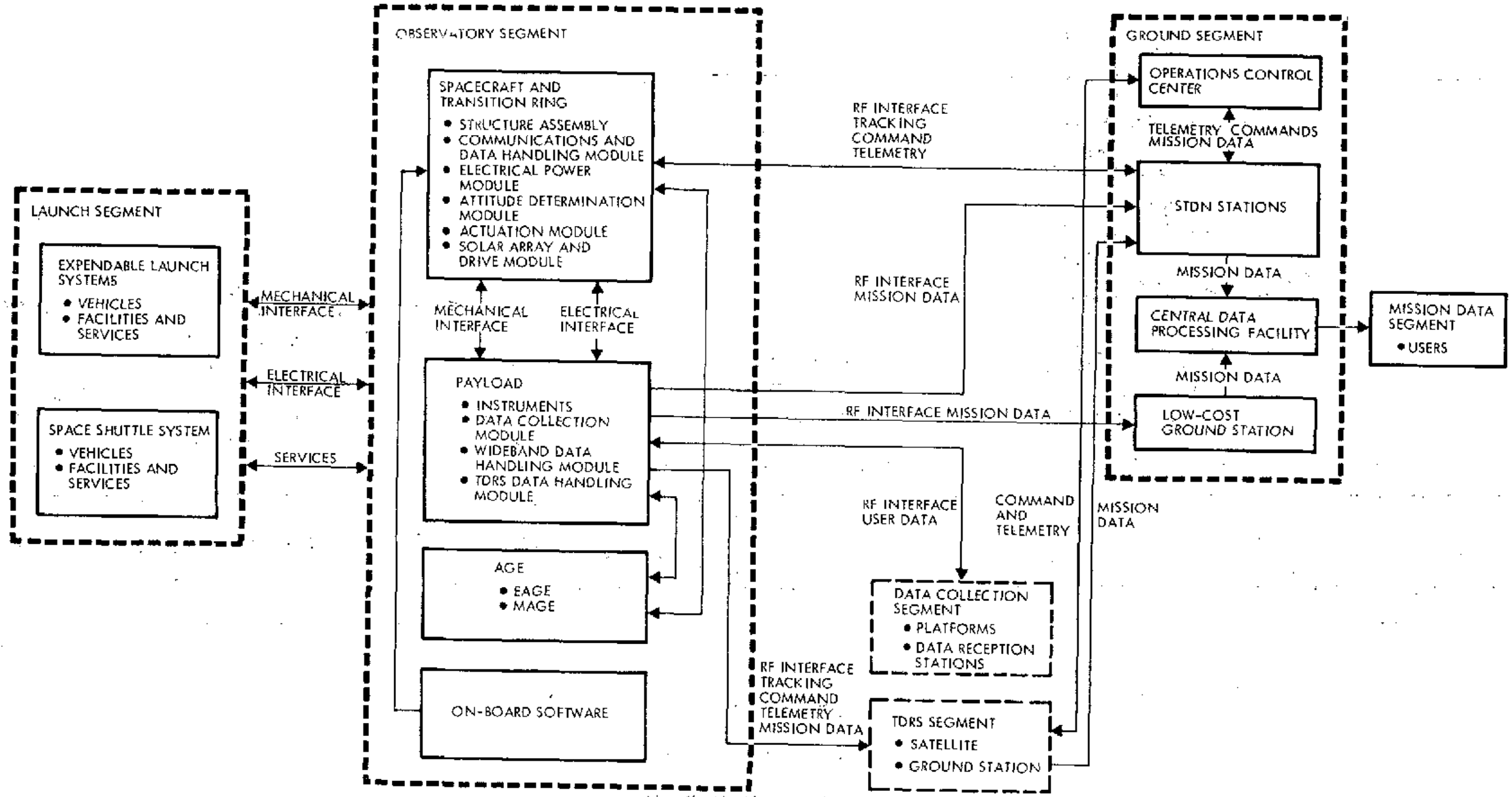

Figure 1. EOS System Functional Block Diagram

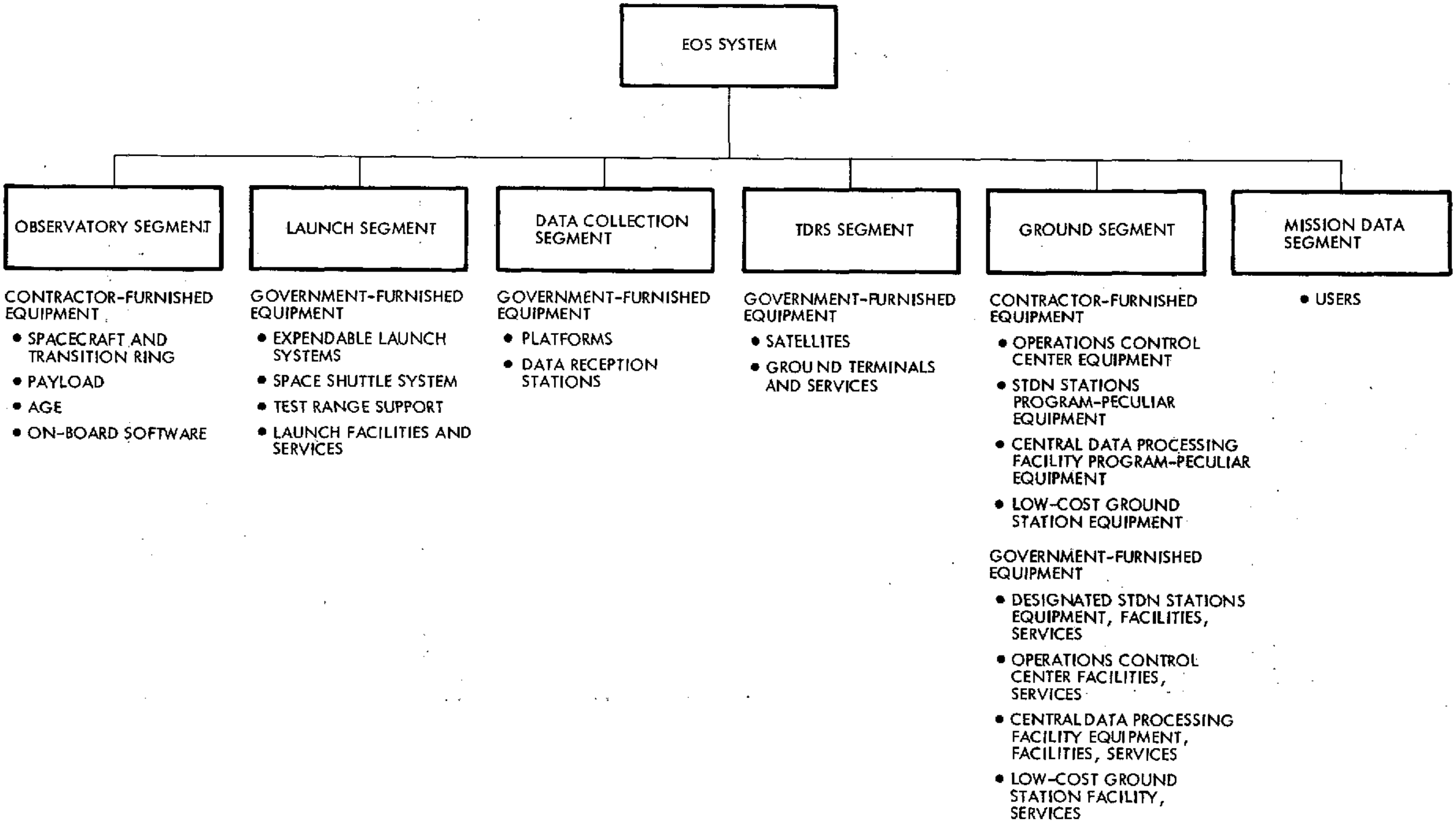

Figure 2. System Segment Relationships

CDPF in the form of tape recordings or in real-time in the case where the STDN station is collocated with the OCC and the CDPF.

The OCC operating through the designated STDN stations shall monitor the Observatory status and shall determine and execute the necessary commands to the Observatory to accomplish the mission objectives.

The CDPF shall process the received instrument payload data into a form suitable for user application.

For some missions, the TDRS system (when available) may be required to support the EOS system by providing a tracking, telemetry, command, and instrument payload data communication relay between the Observatory and the TDRS ground station for transmission to the OCC and CDPF via surface link. This link will enable the tracking, telemetry, command, and payload data transmission at all times when the Observatory is within line-of-sight with the TDRS.

A data collection system (DCS) function may be required on some missions to relay ground instrument data from remote locations to ground collection stations. In this event a DCS module shall be provided on the Observatory to implement the RF relay between the remote location and the collection station when both are within line-of-sight with the Observatory.

The spacecraft portion of the Observatory shall be designed to be launched on any of the launch vehicles identified in paragraph 3.1 and the design shall comply with the range safety requirements for the selected launch site.

The EOS system shall perform the mission and conform to the requirements as specified in the following documents (see 3.8):

(a) SPM-XX, Mission Requirements Specification

(b) SP-11, Observatory Segment Specification

(c) SP-21, Launch Segment Specification

(d) SP-31, Ground Segment Specification.

3.2.2 Physical characteristics

3.2.2.1 Observatory weight. The Observatory weight at launch plus the weight of mission-peculiar adapters for mating the Observatory to the launch vehicle shall not exceed 95 percent of the designated launch vehicle nominal performance capability for the specified launch trajectory on an estimated weight basis and shall not exceed 95 percent of the launch vehicle nominal performance capability on an actual weight basis.

3.2.2.2 Observatory dimensions. The Observatory dimensions in the launch configuration shall be such as to preclude inadvertant contact with the aerodynamic enclosure of the launch vehicle in the specified launch environment.

3.2.2.3 Ground segment

3.2.2.3.1 Operations Control Center. A dedicated facility with floor space of approximately TBD square feet shall be provided to contain the OCC. This facility shall be located as closely as practicable to the CDPF.

The support functions required to support 24-hour per day operation of the OCC shall be provided. These support functions shall include:

(a) Air conditioning/heating/humidity control per TBD

(b) Electric power to support required facility operations

(c) Building maintenance

(d) Sanitary facilities

(e) Communications to support required facility operations.

Proper layout of the OCC shall be considered a design requirement. Location of facilities and equipment shall be designed to support the accomplishment of the OCC mission related tasks. Attention shall be directed to location of displays, controls, maintenance equipment, and their accessibility. The design of the OCC shall meet the requirements of SP-311.

3.2.2.3.2 Central Data Processing Facility. The general requirements as specified in paragraph 3.2.2.3.1 shall also apply to the CDPF except that the facility floor space shall be approximately TBD square feet. The design of the CDPF shall meet the requirements of SP-312.

3.2.2.3.3 Low-Cost Ground Station. A dedicated facility with floor space of approximately TBD square feet shall be provided to contain the LCGS. Additional floor space for administration, security, and services shall be provided as required.

The support functions required to support 24 hour per day operation of the LCGS facility shall be provided. These support functions shall include:

(a) Air conditioning/heating/humidity control per TBD

(b) Electric power to support station operations

(c) Building maintenance

(d) Security guards

(e) Messing

(f) Sanitary facilities

(g) Communications to support required station operations.

Proper layout of the LCGS shall be considered a design requirement. Location and arrangement of facilities and equipment shall be designed for efficient accomplishment of the LCGS related tasks. Attention shall be directed to the location of displays, controls, operating equipment, maintenance equipment, and their accessibility. The design of the LCGS shall meet the requirements of SP-313.

3.2.2.3.4 STDN stations. Certain STDN stations will be designated to support specific mission operations. Mission-peculiar modifications to these stations may be required to support specific EOS missions. These modifications shall be made within existing constraints imposed by stations to the extent practicable. The design of modifications shall be in accordance with SPM-XX and SP-314 (see paragraph 3.8).

3.2.3 Reliability

3.2.3.1 Observatory. The Observatory shall perform as specified herein and survive launch, ascent, separation, orbit circularization, and initial deployment with a probability of success equal to or greater than 0.98. The Observatory shall perform on-orbit after initial deployment as specified herein with the probability of success and for the time specified in the Mission Requirements Specification, SPM-XX. The on-orbit reliability shall consider all spacecraft and payload functions essential to achieving the required mission objectives.

3.2.3.2 Launch vehicle. The launch vehicle availability is a function of the reliability/maintainability of the launch vehicle prior to liftoff and the reliability from liftoff to satellite separation at the specified state vector. The launch vehicle shall have an availability for launch equal to or greater than 0.95 at the preselected launch time. The launch vehicle shall perform from liftoff through Observatory separation with a probability of success equal to or greater than 0.9.

3.2.3.3 Ground Segment availability. The equipment functional, inherent availability of each ground segment element shall be TBD. Inherent availability shall exclude preventive maintenance time. Reliability and maintainability shall be apportioned within the constraints imposed on downtime.

The maximum downtime for any reason, including any combination of preventive and/or unscheduled maintenance, shall be no greater than TBD minutes during any 24-hour period. This value is taken at the 95 percent upper confidence limit (95th percentile point of the distribution of downtime). In the calculation of functional inherent availability, a TBD minute time increment shall be included as part of the basic equipment MTTR. This TBD minute period is the time required for system and subsystem level fault isolation and repair verification. Any corrective maintenance action that can be performed without interference with equipment functional operation shall be excluded from the calculation of MTTR. The maintainability requirements specified herein shall be implemented in accordance with MIL-STD-470.

3.2.4 Maintainability. The EOS system equipment shall be designed in accordance with the requirements of MIL-STD-1472, paragraph 5.9, as implemented by EOS-4.1.

3.2.5 Environmental conditions. The EOS system equipment shall be designed to withstand or shall be protected against the worst probable combination of applicable environments as specified in SP-115.

3.2.6 Transportability. Transportability requirements shall be considered in the design of EOS system equipment such that it can be transported by all standard modes with a minimum of special packing or precautionary measures.

3.3 Design and construction

3.3.1 Parts, materials, and processes

3.3.1.1 Selection of parts, materials, and processes. Selection of parts, materials, and processes (PMP) shall be to the requirements identified in MIL-STD-143 Group I. Such selections shall be identified as standard PMP. When PMP selection cannot be made within this requirement, the item is nonstandard and justification must be made in accordance with MIL-STD-749. Control of this action shall be effected by the EOS parts, materials, and processes control board (PMPCB), with the PMPCB approving all selections, both standard and nonstandard, in accordance with EOS-4.1.

3.3.1.2 Program authorized parts list. All sections of electronic parts shall be identified and authorized by EOS-3.3-1. This list will reflect all PMPCP electronic part selections.

3.3.1.3 Program authorized materials list. All selections of materials shall be identified and authorized by EOS-3.3-2. This list will reflect all PMPCB material selections.

3.3.1.4 Program authorized processes list. All selection of processes shall be identified and authorized by EOS-3.3-3. This list will reflect all PMPCB process decisions.

3.3.2 Electromagnetic radiation. The EOS system equipment shall be designed for compliance with the radiated emission and susceptibility requirements of NASA/JSC Specification SL-E-0002 as modified or amended by EOS-3.3-4 and EOS-3.3-5.

3.3.3 Nameplates and product marking. EOS system equipment shall be identified in accordance with MIL-STD-130 as implemented by EOS-3.3-6. Where practical, a minimum character height of 0.09 inch shall be used. Marking shall include the manufacturer's name or initials, assembly part number and revision status, assembly serial number, contract number, and others as delineated in the component equipment specifications.

3.3.4 Workmanship. The EOS system equipment shall be constructed, finished, and assembled in accordance with MIL-STD-454, Requirement 9, and the specifications and drawings specified herein.

3.3.5 Interchangeability and replaceability. The EOS system equipment shall be designed to permit removal and replacement of components with a minimum of disturbance of associated or adjacent equipment. Design for service and access shall conform to the principles and requirements of MIL-STD-470.

3.3.6 Safety. The EOS system equipment shall be designed to meet or exceed the requirements of EOS-3.3-7. The design criteria shall include but are not limited to those set forth in MIL-STD-882.

3.3.7 Human performance/human engineering. MIL-STD-1472A shall be used for the design of man/machine interfaces.

3.4 Documentation. The system documentation plan for each mission shall be as specified in the Contract Data Requirements List.

The system specification tree is shown in Figure 3 (see paragraph 6.1).

3.5 Logistics. TBD

3.6 Personnel and training. TBD

3.7 Functional area characteristics. TBD

3.8 Precedence. In the event of conflict between the requirements of any documents referenced in this specification and the requirements stated in this document, the contents of this specification shall take precedence, except that, requirements stated in the Mission Requirements Specification, SPM-XX, shall take precedence over this document where specific mission requirements may supercede the requirements of this specification.

4. QUALITY ASSURANCE PROVISIONS

4.1 General. The quality assurance program controls shall be in accordance with the requirements of MIL-Q-9858.

4.1.1 Responsibility for inspection and test. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified

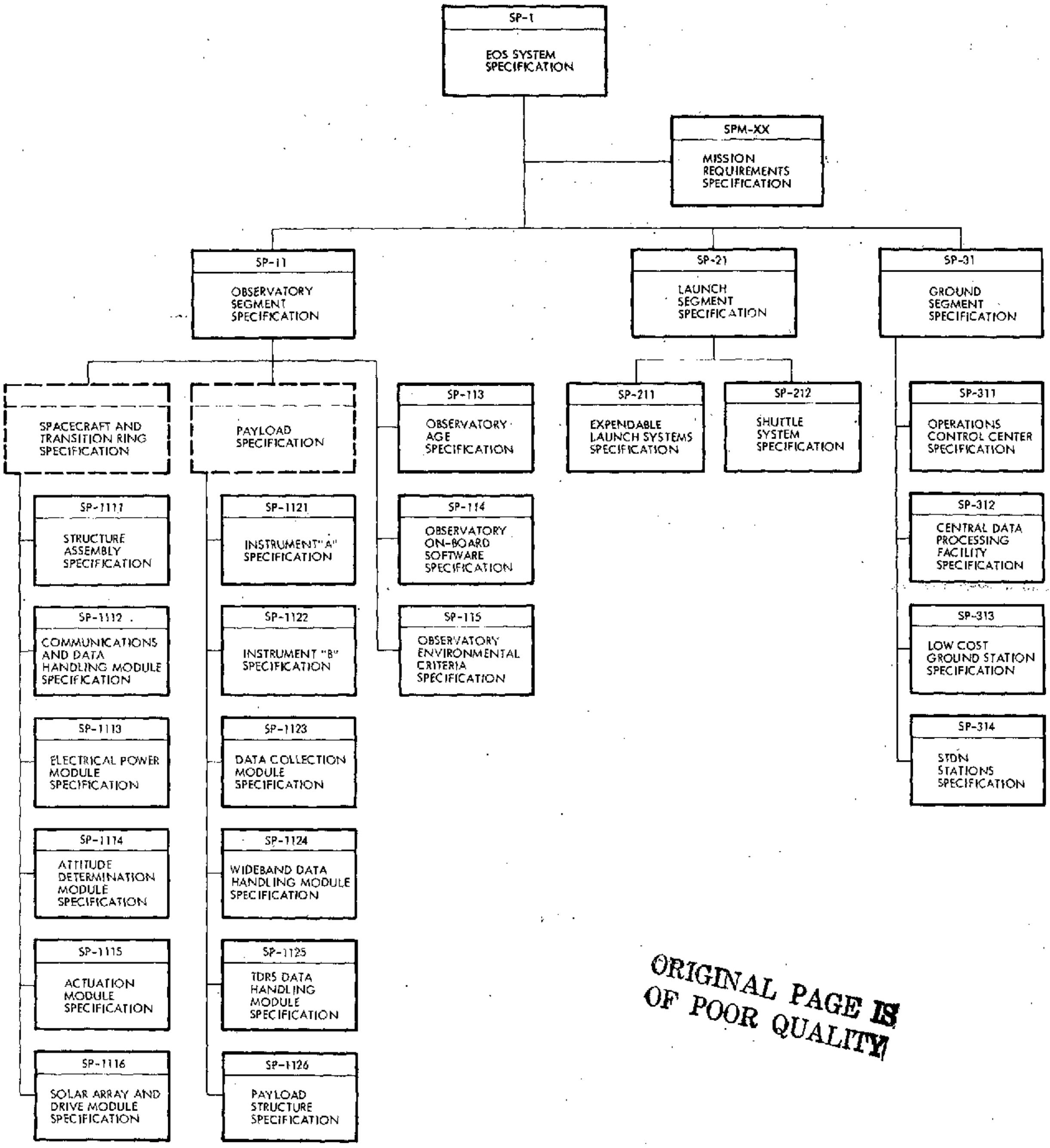

Figure 3. System Specification Tree

herein. Except as otherwise specified, the supplier may use his own facilities or any commercial laboratory acceptable to the government. The government reserves the right to perform any of the inspections set forth in the specifications shere such inspections are deemed necessary to ensure that supplies and services conform to prescribed requirements.

4.2 Quality conformance inspections

4.2.1 Category I tests

4.2.1.1 Development tests. Development tests shall be conducted to verify the system performance parameters and proper interfacing between elements. The development tests shall be conducted at the unit and/or integrated module levels. Tests shall be conducted in accordance with EOS-4.1 and EOS-4.2.

4.2.1.2 Qualification tests. Qualification shall be accomplished by means of qualification tests on the individual units and/or as a normal consequence of having the units installed in the qualification test space vehicle during its qualification testing.

5. PREPARATION FOR DELIVERY

5.1 General. The EOS system equipment shall be protected from degradation by environments anticipated during shipment, handling, and storage. Standard commercial packaging practices are acceptable provided they fulfill these requirements.

6. NOTES

6.1 Launch segment specifications. The launch segment specification portion of the specification tree will require some expansion to include the definitions, performance, and physical characteristics of the EOS program-peculiar equipment necessary to support the EOS missions, e.g., the module exchange mechanism (MEMS) and manipulation and support system for the Shuttle system, peculiar adapter equipment employed in the expendable launch systems, peculiar electrical umbilical provisions and circuitry, fiaring options and RF window provisions, launch vehicle payload weight and center of gravity location constraints, etc.

TITLE

PRIME ITEM DEVELOPMENT SPECIFICATION
FOR THE

OBSERVATORY SEGMENT

DATE 20 SEPT 1974 NO. SP-11

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

ONE SPACE PARK • REDONDO BEACH, CALIFORNIA 90278

SPECIFICATION SP-11

OBSERVATORY SEGMENT

1. SCOPE

This specification establishes the performance, design, development, and test requirements for the Earth Observatory Satellite prime item herein referred to as the Observatory.

2. APPLICABLE DOCUMENTS

2.1 Government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

SPECIFICATIONS

NASA

SP-1	EOS System Specification
SP-113	Observatory AGE Specification
SP-114	Observatory On-Board Software Specification
SP-115	Observatory Environmental Criteria Specification
SP-1111	Structure Assembly Specification
SP-1112	Communication and Data Handling Module Specification
SP-1113	Power Module Specification
SP-1114	Attitude Determination Module Specification
SP-1115	Actuation Module Specification
SP-1116	Solar Array and Drive Module Specification
SP-1124	Wideband Data Handling Module Specification
SPM-XX	Mission Requirements Specification

IFS-T/D-TBD	THOR/DELTA/Observatory Interface Specification
IFS-TIIIB-TBD	TITAN IIIB/Observatory Interface Specification
IFS-TIIIC-TBD	TITAN IIIC/Observatory Interface Specification
IFS-TIIID-TBD	TITAN IIID/Observatory Interface Specification
IFS-TIIIE/ CENTAUR-TBD	TITAN IIIE/CENTAUR/Observatory Interface Specification
IFS-SS-TBD	Space Shuttle/Observatory Interface Specification
IFS-LF-TBD	Launch Facilities/Observatory Interface Specification

Military

MIL-C-45662A 62 Feb 09	Calibration System Requirements
MIL-B-5087	Bonding, Electrical and Lightning Protection for Aerospace Systems
MIL-E-8983A 71 Nov 30	Electronic Equipment, Aerospace Extended Space Environment, General Specification for
MIL-Q-9858A	Quality Program Requirements

STANDARDS

NASA

X-560-63-2	Aerospace Data Systems Standards (GSFC document)

Military

MIL-STD-100A 67 Oct 01	Engineering Drawing Practices
MIL-STD-130D 71 Jul 30	Identification Marking of U.S. Military Property
MIL-STD-143B 69 Nov 12	Standards and Specification, Order of Precedence for the Selection of

MIL-STD-454C 71 Dec 01	Standard General Requirements for Electronic Equipment
MIL-STD-462	EMI/EMS Test Methods
MIL-STD-749B	Preparation and Submission of Data for Approval of Nonstandard Electronic Parts
MIL-STD-882 69 Jul 15	System Safety Program for Systems and Associated Subsystems and Equipment, Requirements for
MIL-STD-1247B 68 Dec 20	Marking, Functions and Hazard Designations of Hose, Pipe and Tube Lines for Aircraft, Missile and Space Systems
MIL-STD-1472A 70 May 15	Human Engineering Design Criteria for Military Systems Equipment and Facilities
MIL-STD-1512 72 Mar 21	Electro Explosive Subsystems, Electrically Initiated Design Requirements and Test Methods

OTHER PUBLICATIONS

NASA

SL-E-0001	Electromagnetic Compatibility Requirements, Systems for the Space Shuttle Program
SL-E-0002	Electromagnetic Compatibility Control Plan
- - -	Instrument Constraints and Interface Specification (TRW Doc. No. 22296-6001-RU-01)
EOS-3.3-1	Program Authorized Parts List
EOS-3.3-2	Program Authorized Materials List
EOS-3.3-3	Program Authorized Processes List
EOS-3.3-4	Electromagnetic Compatibility Control Plan
EOS-3.3-5	EMI/EMS Limits and Test Methods

Military

Manuals

AFWTRM 127-1	Range Safety Manual
AFM 71-4 68 May 29	Packaging and Handling of Dangerous Material for Transportation by Military Aircraft
AFM 127-100 71 Dec 02	Explosive Safety Manual
AFM 160-39 64 Apr	Handling and Storage of Liquid Propellants

Handbooks

DH 1-6 71 Jul 20	System Safety

2.2 Non-government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. If no revision or data is shown, the latest released issue of the applicable document shall apply. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

DRAWINGS

10.3	Satellite Command Allocation ICD
10.4	Satellite Telemetry Allocation ICD
10.5	Observatory/Payload ICD
10.6	Observatory Envelope Dimensions, ICD
10.7	Spacecraft/Modules Mechanical ICD
10.8	Instrument/Payload Structure Mechanical ICD

OTHER PUBLICATIONS

EOS-3.3-7	Safety Design Criteria
EOS-3.3-8	Configuration Management Plan
EOS-4.1	System Effectiveness Program Plan

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EOS-4.2	Integrated Test Plan
EOS-4.6	Component/Module Development Test Plan

3. REQUIREMENTS

3.1 Item definition. This specification establishes the performance and design requirements for the configuration item identified as the Observatory in SP-1.

The Observatory shall be modular in design such that,

(a) It is adaptable to accommodating a variety of mission payloads for earth and celestial observations in low earth orbits or synchronous altitude orbits.

(b) Supporting modules can be modified independently as required to support the mission requirements.

(c) Modules may be replaced independently on-orbit by the Space Shuttle system and, alternatively, replaced on the ground after Space Shuttle system retrieval of the Observatory.

3.1.1 Item diagrams. Figure 1 shows a functional block diagram for the Earth Observatory Satellite (EOS). Example of payload equipment is shown for purposes of defining these interfaces.

The Observatory shall consist of two major separable elements: 1) the payload section which contains the payload modules, payload structure and associated electrical and thermal integration items; and 2) the spacecraft section which contains the supporting modules, spacecraft structure, associated electrical and thermal integration items, and the transition ring assembly which joins the spacecraft and payload sections, supports the solar array and drive module and the payload junction box, and provides the structure for attachment to the launch vehicle adapter.

3.1.2 Interface definition

3.1.2.1 External interfaces. The external interfaces with the Observatory are identified in Figure 2 for an example EOS mission. These may vary for different missions and such variation shall be specified in the Mission Requirements Specification (SPM-XX).

3.1.2.1.1 Observatory/launch vehicle. The Observatory shall be compatible with and be capable of being launched by any one of the following launch vehicles in the interface environment as specified for such vehicle.

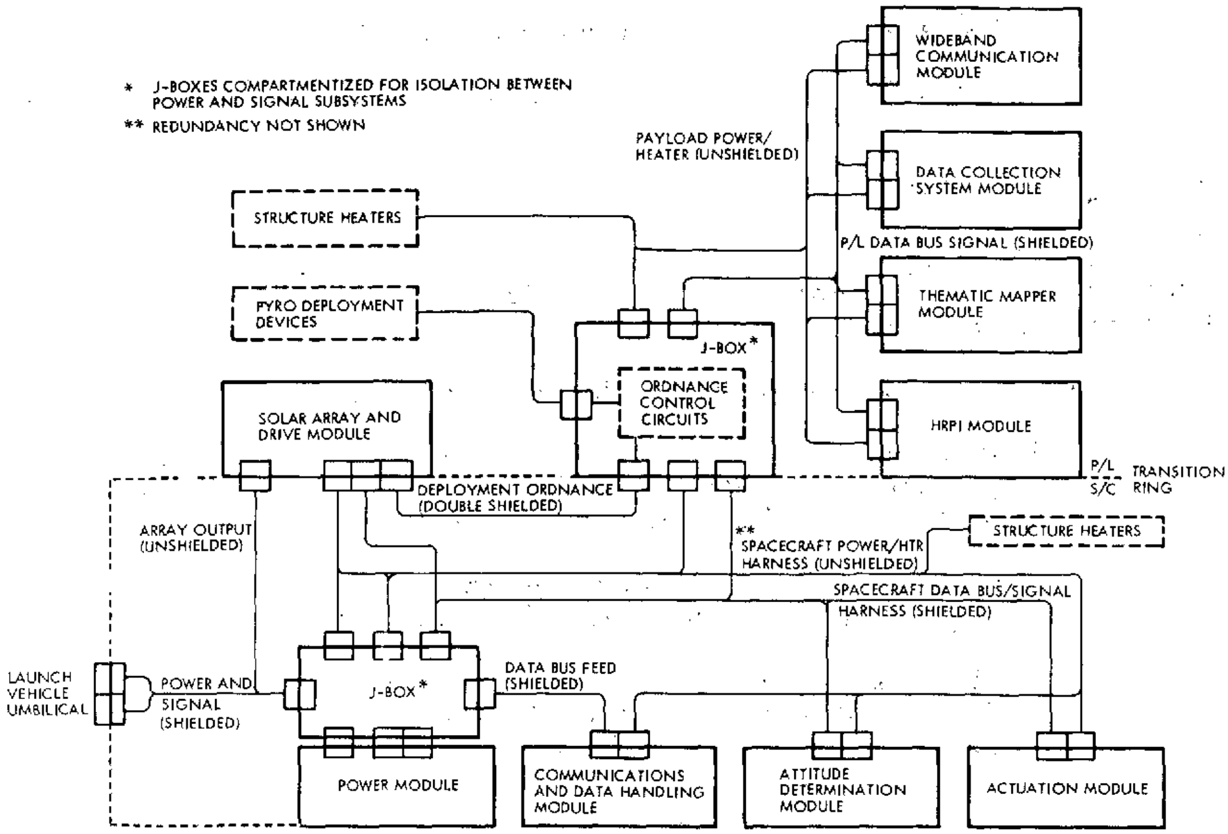

Figure 1. Baseline Modular Block Diagram

Launch Vehicle	Interface Specification
Thor/Delta	IFS-T/D - TBD
TIIIB	IFS-TIIIB - TBD
TIIID	IFS-TIIID - TBD
TIIIC	IFS-TIIIC - TBD
TIIIE/Centaur	IFS-TIIIE/Centaur - TBD
Space Shuttle	IFS-SS - TBD

The Observatory/launch vehicle adapter, normally, shall mate with and support the Observatory at the transition ring during launch and retrieval operations. Where weight is critical for some missions employing expendable boosters, the procuring agency may authorize the attachment of the adapter to the aft spacecraft structures providing spacecraft structural loads are not exceeded.

The conceptual arrangements for mating the Observatory to expendable launch vehicles or the Space Shuttle are illustrated in

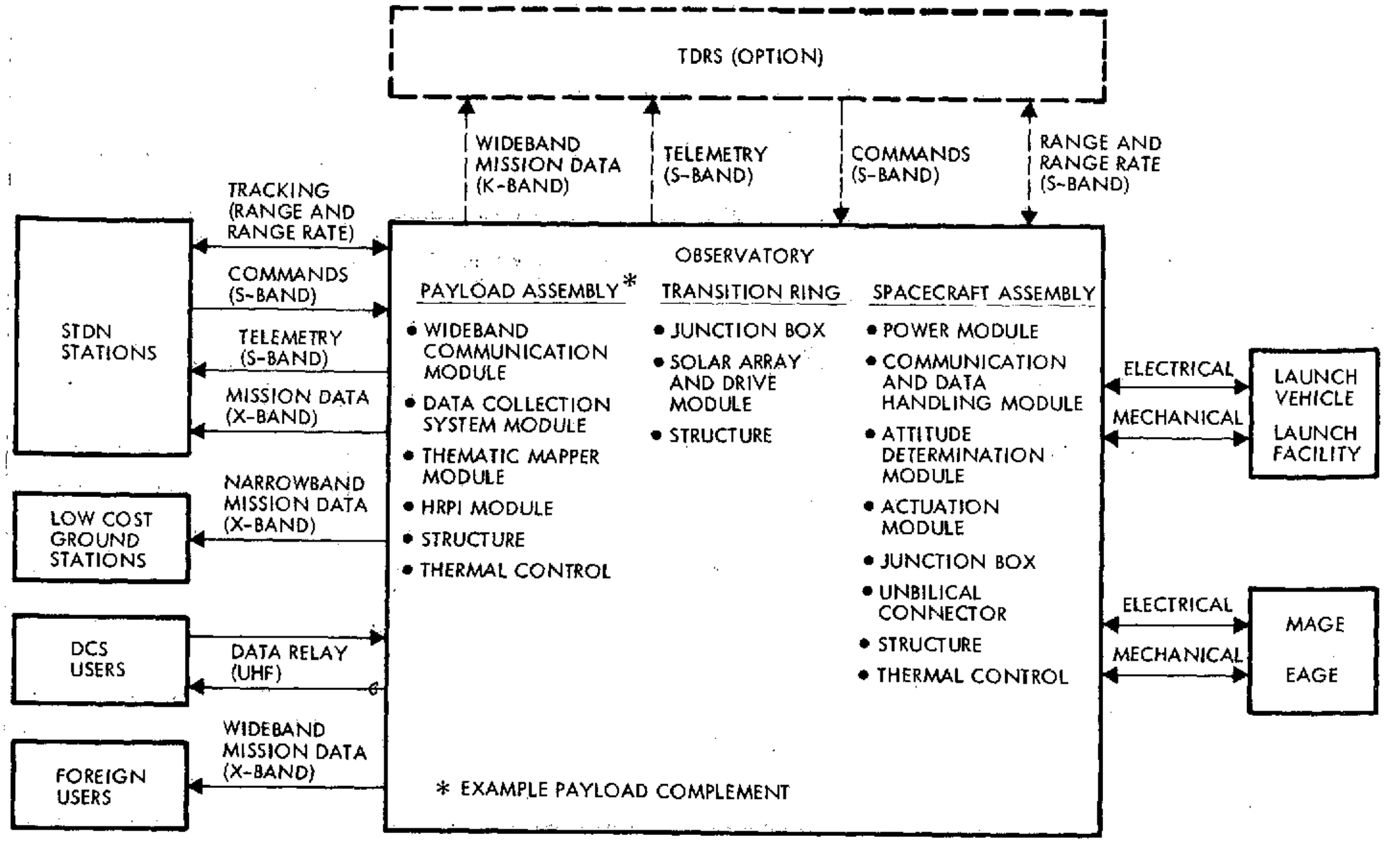

Figure 2. Interfaces Between Observatory and Other System Segments for Example EOS Mission

Figure 3. The Observatory design shall permit mating with the Space Shuttle during on-orbit service operations in the manner specified in IFS-SS-TBD.

3.1.2.1.1.1 Expendable launch vehicle fairings. It is intended that space-qualified expendable launch fairings be used where applicable. There shall be no contact between the Observatory and the fairing under the dynamic environment experienced during the launch phase.

3.1.2.1.1.2 Fairing jettison. The Observatory design shall be compatible with the heating and dynamic pressure environment existing at the time of fairing jettison.

3.1.2.1.1.3 Satellite environment. The Observatory shall be designed so that its performance is not degraded by exposure to the launch vibration, acoustic, shock, and thermal environments as specified in the launch vehicle interface specifications identified in paragraph 3.1.2.1.1.

3.1.2.1.2 Observatory/launch facility. The interface between the Observatory and the launch facility shall be as defined in IFS-LF-TBD. The launch facilities will be capable of receiving, inspecting, checking out, mating, and servicing the Observatory. Observatory-peculiar AGE shall be provided by the Observatory integrating contractor.

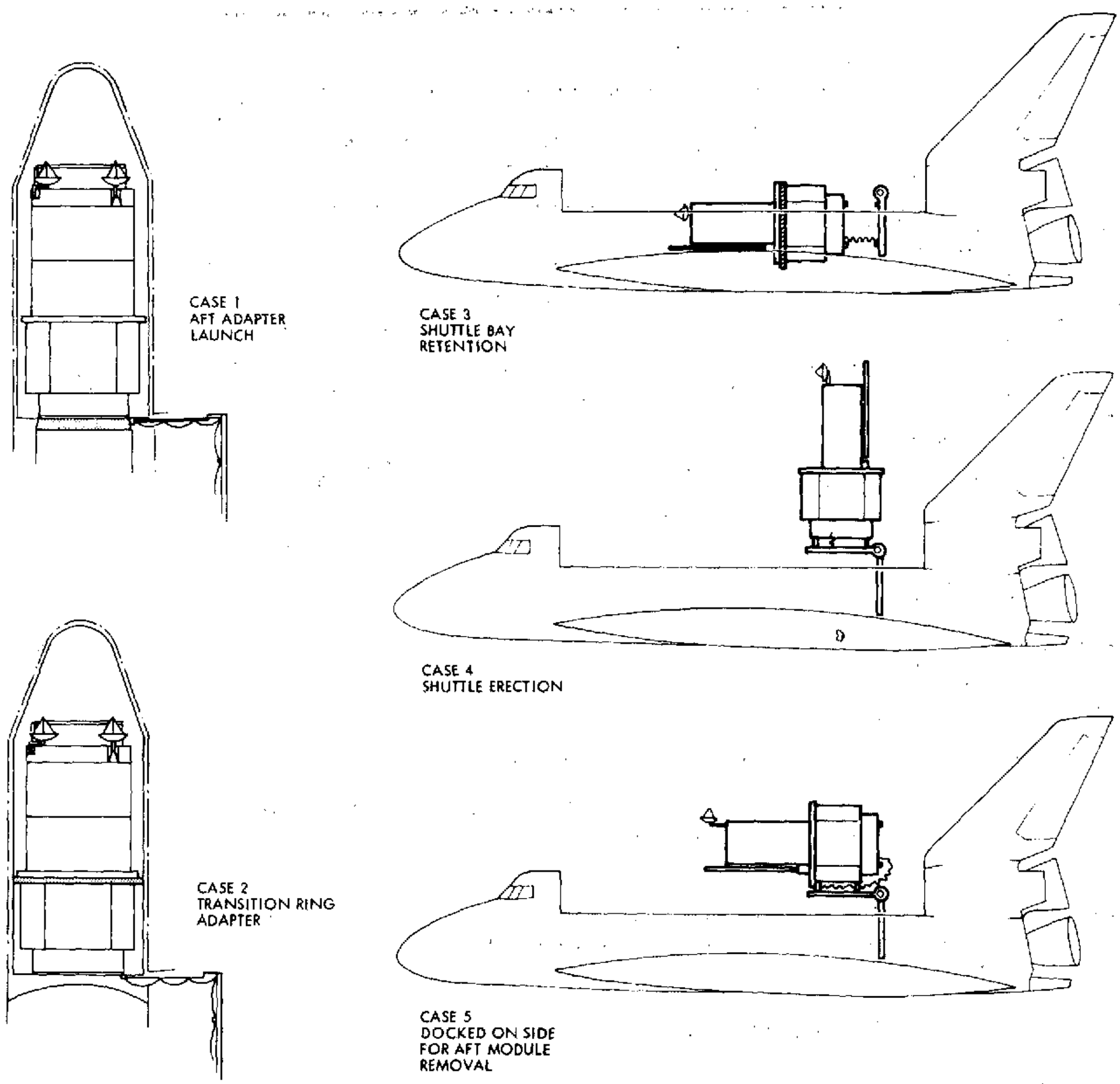

Figure 3. Observatory/Launch Vehicle Mating Concepts

3.1.2.1.3 Observatory/STDN stations. The Observatory shall interface with the STDN stations and be compatible with the unified S-band system for tracking, command and telemetry as defined in GSFC document X-560-63-2. The Observatory shall also be capable of transmitting wideband payload data to selected STDN stations. These RF links shall have the following characteristics:

(a) Uplink (USB):

Frequency:	TBD MHz (2050 to 2150 MHz range)
Data rate:	2000 bits/sec

Error rate:	$<1 \times 10^{-10}$
Antenna polarization:	Singly linearly polarized or dual right and left circular
Command modulation:	PCM/PSK/FM/PM

(b) Downlink (USB):

Frequency:	TBD MHz (2200 to 2300 MHz range)
Narrow band telemetry rate:	Selectable 64, 32, 16, 8, 4, 2, 1 kbit/sec
Narrow band modulation:	Split phase PCM/PMPSK on subcarrier
Medium rate telemetry:	512 kbit/sec maximum
Medium rate modulation:	Split phase PCM/PM

(c) Wideband Downlink (Mission Data):

Frequency:	TBD MHz (X-band)
Data rate:	256 Mbits/sec
Error rate:	10^{-5}
Maximum power flux density at earth:	140 dBw/m^2/4 kHz (zenith) 150 dBw/m^2/4 kHz (horizon)
Modulation:	QPSK or MSK

3.1.2.1.4 Observatory/low cost ground station. The Observatory shall be capable of transmitting wideband payload data to the low-cost ground station (LCGS). This link shall have the following characteristics:

Frequency:	TBD MHz (X-band)
Data rate:	20 Mbits/sec
Error rate:	10^{-5}
Maximum power flux density at earth:	140 dBw/m^2/4 kHz (zenith) 150 dBw/m^2/4 kHz (horizon)
Modulation:	BPSK

3.1.2.1.5 Observatory/foreign users. The Observatory shall be capable of transmitting payload data to foreign users when programmed to do so by STDN command. This link shall have the same characteristics as specified in paragraph 3.1.2.4.

3.1.2.1.6 Observatory/data collection system users. The Observatory shall be capable of relaying signals from remote sensors to data collection system (DCS) user stations. These RF links will operate at UHF and have the following characteristics: TBD

3.1.2.1.7 Observatory/AGE. The aerospace ground equipment, as defined in SP-113, shall be electrically and mechanically compatible with the Observatory design.

3.1.2.2 Internal interfaces. The Observatory shall be modular in design such that it is adaptable to a variety of missions by assembly of a set of mission-peculiar modules to a standard spacecraft structure. The payload structure may be mission-peculiar depending upon the payload configuration. The module internal configurations shall be specified for each mission. The modular design concept for an example EOS mission is illustrated in Figure 4. Functional block diagrams for the minimum redundancy and nominal redundancy configurations (see 6.1 for definitions) of an EOS mission example Observatory are illustrated in Figures 5 and 6. Module redundancy may be varied to satisfy specified mission-peculiar mean mission duration requirements.

Module interfaces with the Observatory structure/electrical assembly shall be standardized to the maximum extent practical and shall be controlled by ICD 10.7.

The module mechanical and electrical connections shall be designed to enable on-orbit replacement by the special module exchange mechanism to be provided by the Space Shuttle system.

Primary power shall be distributed from the electrical power module to the spacecraft and payload modules on two independently controlled main buses. The primary bus voltage at the connecter on each using module shall be 28 ± 7 volts.

Primary power for the structure and module heaters shall be distributed from the electric power module to using modules on an independentally controlled bus.

Commands received by the communication and data handling module shall be distributed to all modules via remote decoder units located in the user modules.

Data transfer between the communication and data handling module and all modules shall be via remote multiplexers located in the user modules.

Communications between the communication and data handling module and the remote decoder and multiplexer units shall be via a full

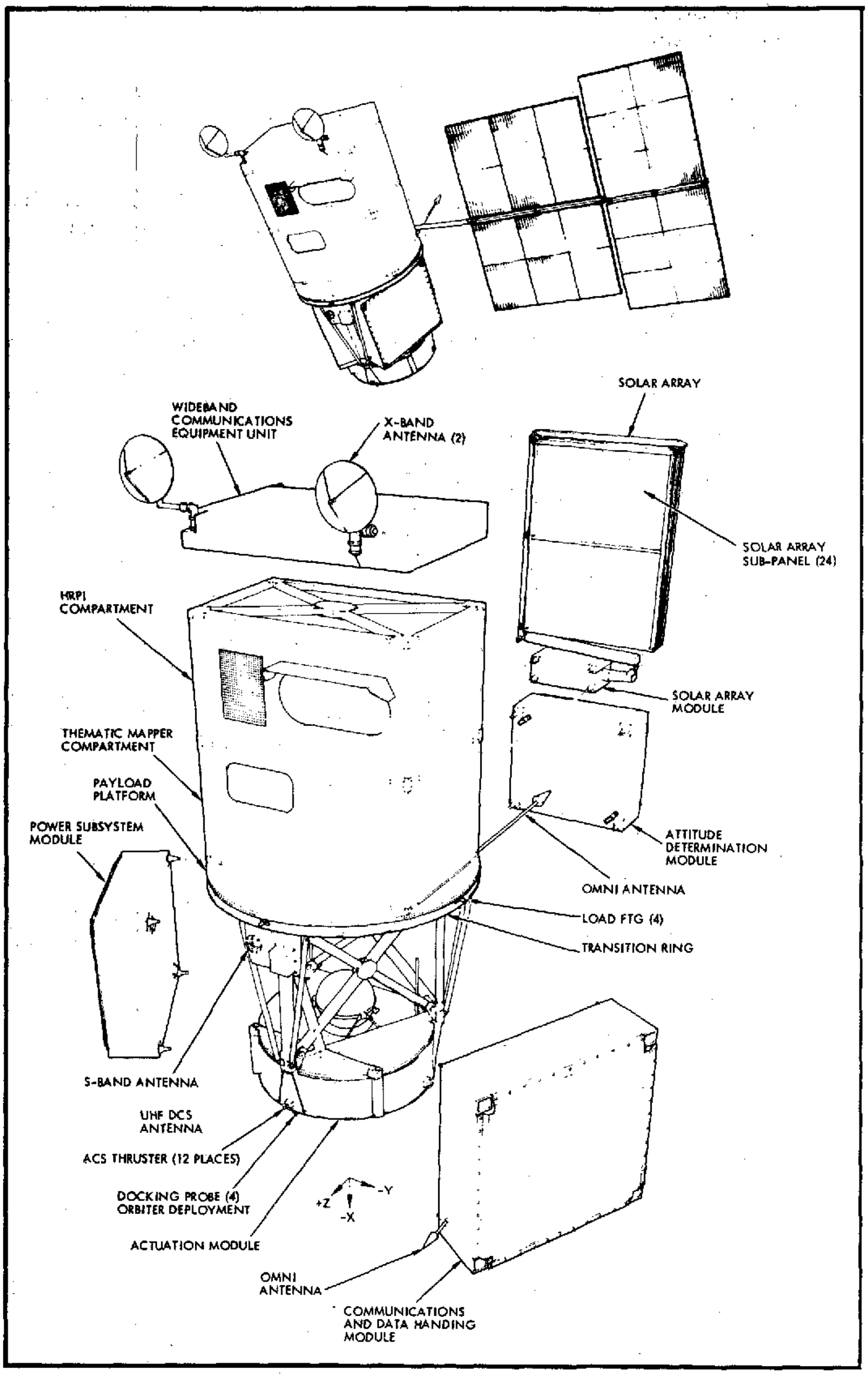

Figure 4. Modular Configuration Concept for Example EOS Mission

duplex party line bus. The data bus shall be physically separated from high level buses to the maximum practical extent.

The on-board computer in the communication and data handling module shall communicate with all spacecraft and payload modules via the data bus.

The spacecraft clock located in the communication and data handling module shall be available via the data bus for timing all other Observatory functions as required.

The attitude determination module shall have the capability of accepting error signals derived by a payload instrument as a mission peculiar option.

3.1.3 Major component list

3.1.3.1 Spacecraft. Major components which comprise the equipment configuration of the spacecraft shall be:

(a) Solar array and drive module

(b) Spacecraft structure assembly

(c) Electric power module

(d) Communication and data handling module

(e) Attitude determination module

(f) Actuation module

3.1.3.2 Payload. The payload elements are mission peculiar and shall be specified separately for each mission procurement. However, the payload mechanical and electrical interface shall be compliant with the requirements specified herein and defined in ICD EOS 10.5.

3.1.4 Government-furnished module equipment. Certain items of module equipment which are identical or nearly identical in function will be procured or fabricated by the integration contractor and provided as GFE to the remaining module contractors. This equipment shall include:

Item	Drawing No.
Data interface unit	TBD
Power conditioning unit	TBD

All module structures and integration of the modules will be provided by the integrating contractor.

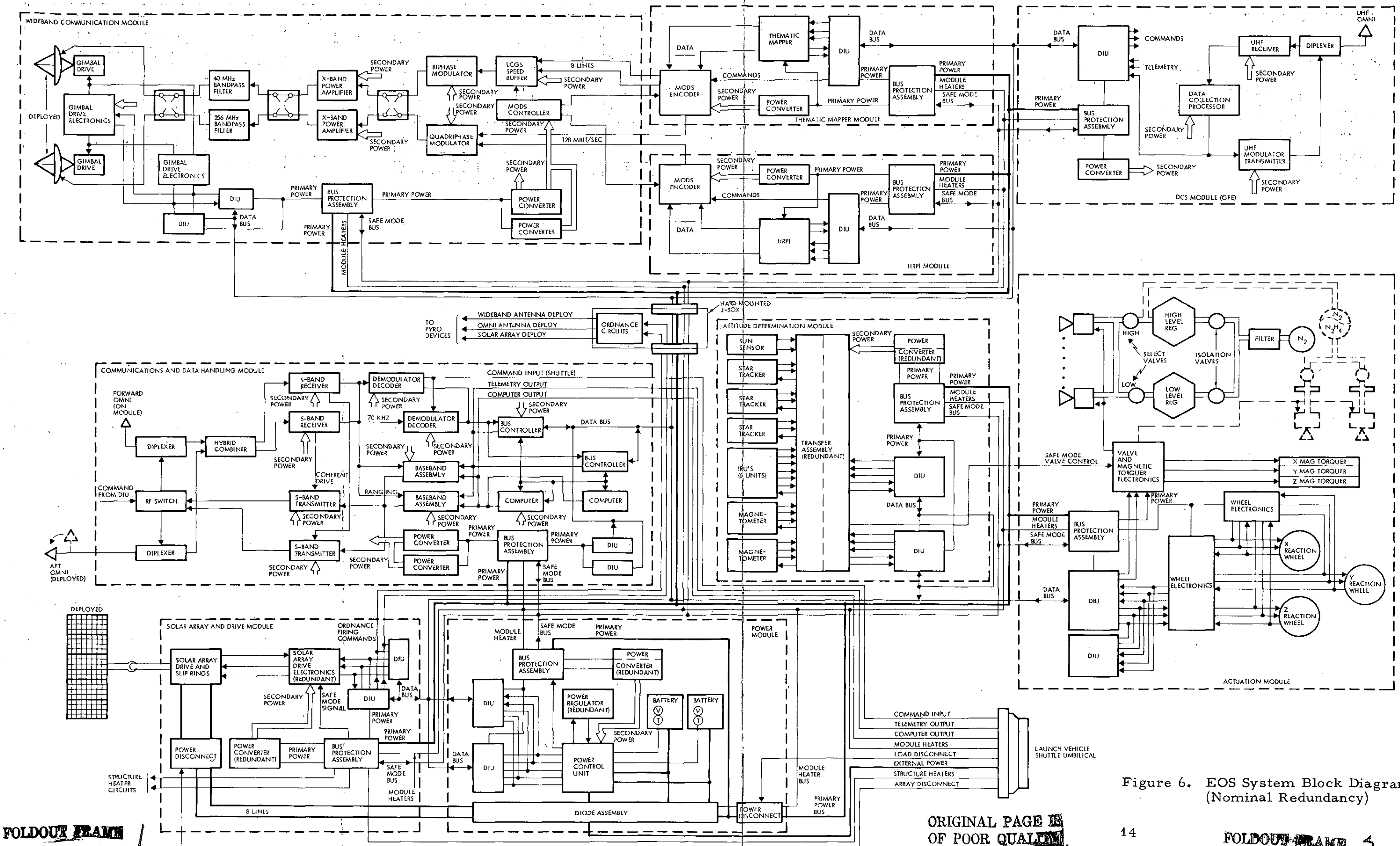

Figure 6. EOS System Block Diagram (Nominal Redundancy)

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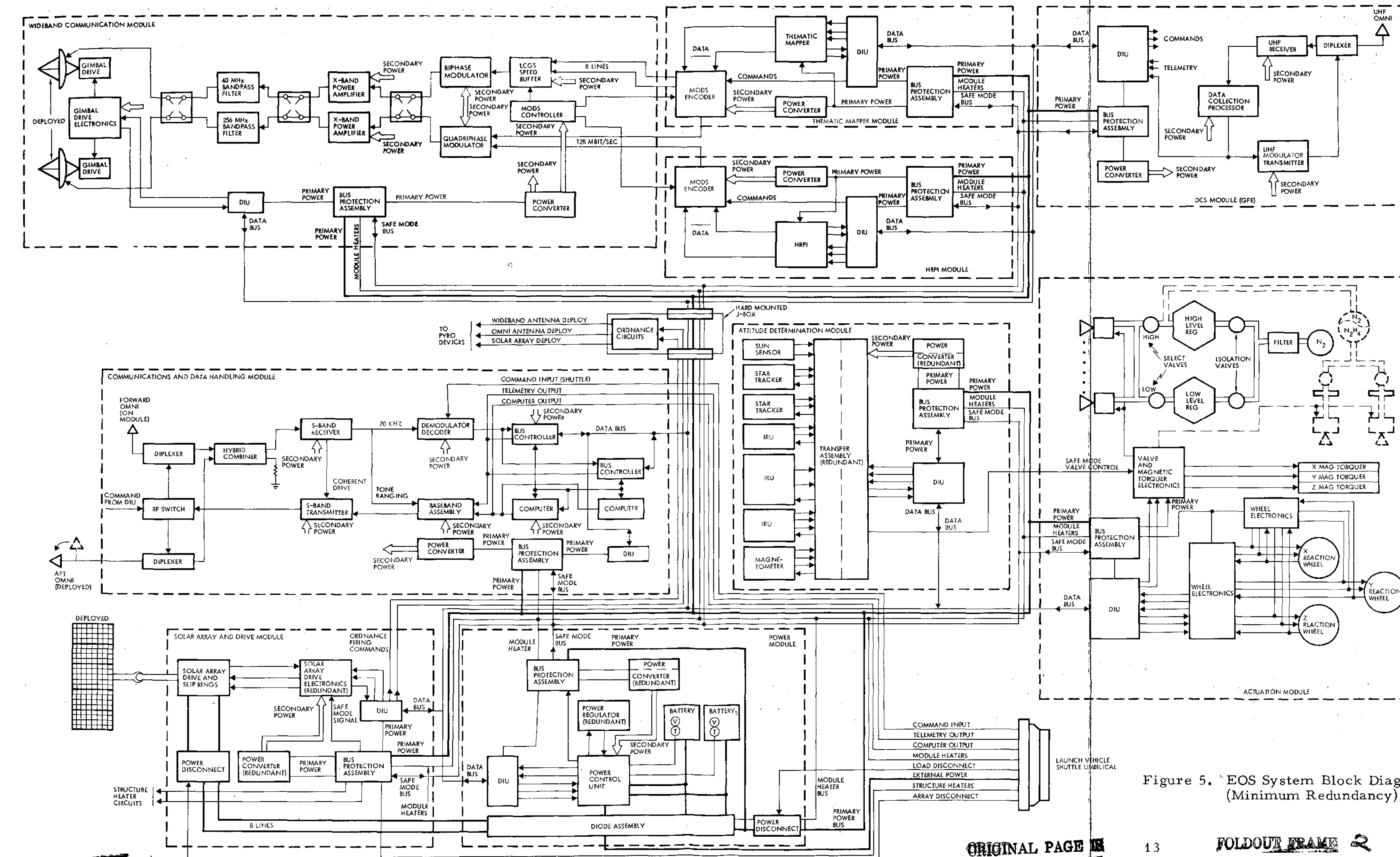

Figure 5. EOS System Block Diagram (Minimum Redundancy)

3.2 Characteristics

3.2.1 Performance. The Observatory shall be capable of being launched and achieving and maintaining the final mission orbit in the manner specified in SPM-XX.

3.2.1.1 Observatory separation

3.2.1.1.1 Observatory/launch vehicle. The Observatory shall provide for ordnance-actuated release from the Observatory/launch vehicle adapter when launched by an expendable launch vehicle. Signals to initiate the separation sequence will be provided by the launch vehicle. Separation springs shall impart a velocity relative to the launch vehicle of at least 1 ft/sec. Tipoff rates shall remain below TBD.

3.2.1.1.2 Observatory space shuttle. When launched by the Space Shuttle the separation sequence shall be as provided by the Space Shuttle system under the supervision of the shuttle crew.

3.2.1.2 Observatory attitude determination and control. The performance of the attitude determination and control functions of the Observatory during the data taking orbit period shall be as specified in Tables 1, 2, and 3.

The design for attitude determination and control shall meet the requirements specified in SP-1114 and SP-1115. During ΔV burns the Observatory shall be oriented such that the ΔV thruster is aligned within ±2 degrees of the desired thrust direction.

Table 1. Attitude Estimation Requirements*

	Earth Pointing		Stellar Inertial** or Sun Pointing (per axis)
	Roll, Pitch	Yaw	
Bias	15 arc-sec	25 arc-sec	0.05 arc-min
Drift	0.009 deg/hr	0.04 deg/hr	0.004 deg/hr over 1 hr
Drift rate deviation	0.005 deg/hr over 2 minutes	0.015 deg/hr over 2 minutes	0.004 deg/hr for 30 sec≤t≤1 hr
Jitter	0.30 arc-sec rms	1.0 arc-sec rms	2 arc-sec rms for times up to 1 hr

*Accuracy of estimate of attitude of star tracker mounting surface.
**When separate attitude estimation module is used.

Table 2. **Attitude Control Requirements***

	Earth Pointing		Stellar Inertial** or Sun Pointing (per axis)	Stellar Inertial*** Using Experiment (per axis)
	Roll, Pitch	Yaw		
Bias	7.5 arc-sec	12.5 arc-sec	0.25 arc-min	0.01 arc-sec
Drift	0.005 deg/hr	0.02 deg/hr	0.002 deg/hr over 1 hr	Not applicable
Drift rate deviation	0.0025 deg/hr over 2 minutes	0.007 deg/hr over 2 minutes	0.002 deg/hr for 30 sec≤t≤1 hr	Not applicable
Jitter	0.15 arc-sec rms over 2 minutes	0.5 arc-sec rms over 2 minutes	1 arc-sec rms for times up to 1 hr	0.004 arc-sec rms for times up to 1 hr

*Refers to control using ideal sensors of star tracker mounting surface.
**When separate attitude estimation module is used.
***When the experiment becomes the attitude sensor.

Table 3. Thermal/Structural Bending Requirements*

	Earth Pointing		Stellar Inertial/ Sun Pointing
	Roll, Pitch	Yaw	
Bias	30 arc-sec	30 arc-sec	0.25 arc-min
Drift	0.009 deg/hr	0.04 deg/hr	0.002 deg/hr
Jitter	0.3 arc-sec rms	1.0 arc-sec rms	1 arc sec rms

*Refers to error in the estimate of the attitude of the instrument mounting surface relative to that of the star tracker. A fixed bias in yaw can be compensated by an attitude yaw command.

3.2.1.3 Safe mode. A backup safe mode shall be provided which automatically orients the spacecraft for maximum sun illumination of the solar array. This mode may be initiated by an on-board signal indicating an unsafe low power condition or any other condition deemed threatening to the survivability of the Observatory. During the safe mode the solar array shall be driven to the position where the normal to the array surface is in the Y-Z plane (see Figure 4) and pointing toward the sun. Pointing in this mode shall be by means of the sun sensor and the low level thrusters. Special purpose safe mode logic electronics shall replace the on-board computer attitude function. Gyros, reaction wheels, and magnetic torquers shall be disabled. Return from the safe mode to normal control shall be by ground command.

3.2.1.4 Command and telemetry. The Observatory shall transmit telemetry data and receive commands via the unified S-band links with the STDN stations or alternatively, if specified for the mission, via the S-band links with the TDRS relay. The Observatory shall be capable of transmitting telemetry and receiving commands via the STDN links in any on-orbit attitude orientation when in line of sight with an STDN station. The satellite shall be capable of accepting and executing commands either in real time or stored and executed at a later time as commanded.

3.2.1.5 On-board data handling. On-orbit control functions shall be handled by the computer in the communications and data handling module as specified in SP-1112. The software for the on-board computer shall be compliant with the requirements of SP-114.

3.2.1.6 ΔV operation. The Observatory shall provide a ΔV capability for orbit transfer, initialization and maintenance as required by each specific mission. The minimum ΔV increment shall be TBD ft/sec. The pointing requirements will be relaxed during orbit adjust periods as specified in paragraph 3.2.1.2; however, the close control requirements for data taking shall be met within 30 minutes following the end of any orbit-adjust period.

3.2.1.7 Electric power. The primary power for Observatory operation shall be derived from a solar array. Batteries shall support the electrical load requirements during eclipse and peak loading as required in sunlight. The solar array and battery complement shall be modular in design to support the power and duty cycle requirements to be specified for each mission. The maximum required solar array output shall be TBD watts with normal solar incidence. The maximum battery complement capacity (full discharge) shall be TBD ampere hours. The primary power bus shall be redundant and shall be regulated to within 28 ± TBD volts. Characteristics of the primary power shall be as specified in SP-1113. A separate bus shall be provided for the electrical heaters.

3.2.1.8 Thermal control. The thermal control design of the Observatory shall conform with the following requirements:

- To the extent possible, spacecraft and payload modules shall be thermally decoupled from the spacecraft and payload structures.

- Design flexibility and growth margin shall be provided to accommodate a wide range of experiments and various earth orbit missions.

- Structural temperature gradients and fluctuations shall be limited to preserve structural alignments.

- Design shall be capable of accommodating module replacement via Shuttle.

3.2.1.9 Telemetry. Telemetry data shall be primarily limited to those functions necessary for control and operation of the Observatory during mission life. This telemetry shall include but not be limited to functions such as bus and power converter voltages, the status of all commandable modes, temperatures, receiver signal quality.

3.2.1.10 Useful life. Observatory useful life shall be as specified in paragraph 3.2.3.2.

3.2.1.11 Storage life. The spacecraft shall have a minimum storage life of 3 years. Storage life critical components may be refurbished. Components which require refurbishing shall be identified in the appropriate module specification.

The payload shall have a minimum storage life of TBD years.

3.2.1.12 Expansion capability. A capability shall be provided, within the limits of Observatory envelope dimensions, module structure size and power available, for expanding the baseline configuration to include the addition of redundant units or the addition of new components to perform additional functions.

3.2.2 Physical characteristics

3.2.2.1 Mechanical

3.2.2.1.1 Envelope. The Observatory configuration shall conform with the envelope dimensions shown in ICD 10.6.

3.2.2.1.2 Weight. Maximum Observatory weight for structure and attitude control and determination sizing shall be TBD pounds.

Total Observatory weight shall not exceed the listed weight allocations:

	Minimum Redundancy	Nominal Redundancy
Spacecraft (including propellant and expendables)	1526	1638
Payload	TBD	TBD
Total	TBD	TBD

3.2.2.1.3 Attach-points

(a) Spacecraft. The attach-points between the various modules and the spacecraft shall be as shown in ICD 10.7.

(b) Payload. The attach-points between the various payload instruments and the payload structure shall be as shown in ICD 10.8.

(c) Observatory. The attach-points between the spacecraft and the payload shall be as shown in ICD 10.6.

3.2.2.1.4 Module characteristics. Modules shall be designed to enable replacement during prelaunch operations and by the module exchange mechanisms provided by the Space Shuttle in the on-orbit service mode. Electrical connectors and mechanical attachments shall incorporate self aligning features which enable mate and demate operations with translation misalignments up to TBD inches. The module electrical connections shall be automatically made when mechanical attachment is performed by either automatic servicing on the shuttle or by manual ground devices.

3.2.2.1.5 Connectors

3.2.2.1.5.1 Observatory/launch vehicle umbilical. Only one Observatory umbilical connector shall be provided to mate with the launch system for use during launch operations. The umbilical connector shall be mounted on the spacecraft structure with sufficient accessibility and clearances such that, in addition to normal use with expendable launch vehicles, a mating connector on a Space Shuttle can be installed or removed by a remote manipulator. As a minimum, the functions indicated on Figure 5 shall be required.

3.2.2.1.5.2 Module/structure interface connector. The module/structure interface connector shall be provided by the system contractor and shall be mounted on the inside face of the module. It is required that the connector positions be maintained as specified in ICD's 10.7 and 10.8 to ensure interchangeability of modules.

3.2.2.1.6 Equipment expansion. The components shall be arranged in the spacecraft modules such that the expansion capability requirements of paragraph 3.2.1.12 can be accommodated with minimum impact on the baseline configuration design.

3.2.2.2 Electrical

3.2.2.2.1 Power. Total power required for the Observatory shall not exceed TBD watts. Allocation of this power is as follows:

	Minimum Redundancy (watts)	Nominal Redundancy (watts)
Spacecraft Payloads Contingency	TBD	TBD

3.2.2.2.2 Commands. Commands for controlling the Observatory shall be as listed in ICD 10.3.

3.2.2.2.3 Telemetry. Telemetry measurements for the Observatory shall be as listed in ICD 10.4.

3.2.3 Reliability

3.2.3.1 Observatory reliability. The Observatory reliability requirements will be mission peculiar and shall be as specified in SPM-XX for each mission.

3.2.3.2 Spacecraft reliability. The spacecraft shall be capable of performing as specified for a period of at least TBD years in orbit. Expendables shall be sized for TBD years. This period shall be in addition to the storage requirements of 3.2.1.11. The reliability of the minimum redundancy configured spacecraft shall be a probability of success of not less than TBD for TBD years of orbital operation. The reliability of the nominal redundancy configured spacecraft shall be a probability of success of not less than TBD for TBD years of orbital operation. Orbital operation includes the prior successful survival of launch, coast and deployment on-station. Demonstration of compliance with these requirements shall be through reliability analysis using methods called out in EOS-4.1.

3.2.3.3 Payload reliability. TBD.

3.2.4 Maintainability

3.2.4.1 General. The Observatory shall be designed in accordance with the requirements of MIL-STD-1472A, paragraph 5.9.

3.2.4.2 Access. The Observatory shall be designed to facilitate accessibility and removal and replacement of equipment items during in-plant manufacturing and test operations.

3.2.4.3 Non-interchangeability of connectors. When there are adjacent connectors of like type they shall be physically keyed in such a manner that their interchangeability is precluded.

3.2.4.4 Alignments. Where exact alignments are required during Observatory assembly, alignment references shall be clearly marked and visible for mating operations.

3.2.4.5 Attach points for lifting. All items which require use of a lifting sling or fixture for installation shall have attach points for secure handling.

3.2.4.6 Checkout and test. Functional modules shall be designed to provide a capability to identify failed components during ground checkout and test operations.

3.2.5 Environmental conditions

3.2.5.1 Induced environments. The Observatory shall be designed to withstand or be protected against the following induced environments:

(a) Vibration (transportation, handling, and launch/ascent)

(b) Acoustics (launch/ascent)

(c) Shock (transportation, handling, fairing separation, pyrotechnics).

The quantitative definition of these environments shall be as specified in SP-115.

3.2.5.2 Natural environments. The Observatory shall be designed to withstand or be protected against the following natural environments:

(a) Temperature

(b) Humidity

(c) Pressure

(d) Sand and dust

(e) Fungus

(f) Rain

(g) Solar radiation

(h) Albedo radiation

(i) Earth-emitted radiation

(j) Micrometeoroids

(k) Trapped radiation

The description and/or quantitative definition of these environments shall be as specified in SP-115.

3.2.6 Transportability. The Observatory and the equipment modules shall be capable of being shipped in specially designed containers suitable for transportation by air or road in accordance with the natural, transportation and handling environmental requirements of paragraph 3.2.5.

3.3 Design and construction

3.3.1 Parts, materials and processes

3.3.1.1 Selection of parts, materials, and processes. Selection of parts, materials, and processes (PMP) shall be to the requirements identified in MIL-STD-143 Group 1. Such selections shall be identified as standard PMP. When PMP selection cannot be made within this requirement, the item is nonstandard and justification must be made in accordance with MIL-STD-749. Control of this action shall be effected by the EOS parts, materials, and processes control board (PMPCB), with the PMPCB approving all selections, both standard and nonstandard, in accordance with EOS-4.1.

3.3.1.2 Program authorized parts list. All selections of electronic parts shall be identified and authorized by EOS-3.3-1. This list will reflect all PMPCB electronic part selections.

3.3.1.3 Program authorized materials list. All selections of materials shall be identified and authorized by EOS-3.3-2. This list will reflect all PMPCB material selections.

3.3.1.4 Program authorized processes list. All selection of processes shall be identified and authorized by EOS-3.3-3. This list will reflect all PMPCB process decisions.

3.3.1.5 Dissimilar metals. To avoid electrolytic corrosion, dissimilar metals shall not be used indirect contact unless protection against corrosion has been provided in accordance with MIL-E-8983 and MIL-STD-454, Requirement 16.

3.3.1.6 Magnetic materials. Magnetic materials shall be used only if necessary for equipment operation. Those magnetic materials which are used shall have minimum permanent, induced, and transient magnetic fields.

3.3.1.7 Fungus-inert materials. Materials which are fungus-inert, in accordance with MIL-E-8983 and MIL-STD-454, Requirement 4, shall be used.

3.3.1.8 Flammable toxic and unstable materials. Flammable, toxic, and unstable materials shall not be used.

3.3.1.9 Finish. The surface of each component shall be adequately finished to prevent deterioration from exposure to the specified environments that might jeopardize fulfillment of the specified performance. The finish shall also meet the applicable bonding requirements. Thermal properties of the finishes used on the component shall be compatible with the requirements given in detail in applicable equipment specifications. The requirements for finishes identified in MIL-E-8983 shall be implemented.

3.3.1.10 Outgassing. Low outgassing polymeric materials shall be used where sensitive thermal control and other surfaces are in direct line of sight and where temperature differences can exist between such surfaces.

3.3.2 Electromagnetic radiation

3.3.2.1 EMC/EMI requirements. The Observatory, and all modules, units, equipment components and/or wiring harnesses comprising a part thereof, shall be designed for compliance with the electromagnetic compatibility requirements of NASA/JSC specifications SL-E-0001 and SL-E-0002, as modified or amended by EOS-3.3.4 and EOS-3.3-5.

3.3.2.2 Electrical systems grounding. A hybrid electrical systems grounding policy shall be implemented for the spacecraft and payload. This hybrid will be a judicious mixture of single-point and multiple-point grounding methods selected on a module basis. In general, the primary DC power and party line data bus distribution shall have single-point grounds. Also, single-point grounding shall be employed for low level analog sensor circuits and high current control circuits. Module circuitry wholly contained within a given module shall, with some exceptions, employ multiple-point grounding using the module radiator panel and support structure as the ground reference plane and return path for secondary power and signal currents within the module. On a module or circuit basis, the following grounding rules shall be employed:

3.3.2.2.1 Primary DC power. The primary DC power distribution shall be fully isolated from chassis, case, and structural grounds at all source and load terminals, except for one designated single-point ground to be located in the power module. Power conversion units, or subunits, with DC isolation shall be provided at each primary power user terminal in order to maintain a DC isolation between the primary input and secondary outputs of at least 1 megohm.

3.3.2.2.2 Heater power. The thermal power distribution network shall be fully isolated from chassis, case and structural grounds at all source and load terminals, except for one hardware connection to the primary DC power ground point in the power module.

3.3.2.2.3 Data bus. The command and telemetry data bus shall be differentially driven and resistively balanced to structural ground in the communications and data handling module. This module shall be transformer-coupled and DC-isolated at all remote terminals.

3.3.2.2.4 Secondary DC power. Secondary DC power distribution networks, at the module level and below, shall employ multiple-point grounding with return paths through the module radiator panel and support structure. These circuits shall be grounded to chassis/frame/structure at the secondary transformer winding in the power converter and at each user element.

3.3.2.2.5 Secondary AC power. Secondary AC power networks shall be single-point grounded either at the secondary winding of the power inverter or at the load, whichever is shown by circuit analysis to be most beneficial to compatible module operation. If possible, secondary AC power should be balanced to ground. Structural return paths shall not be used.

3.3.2.2.6 High level signal/control circuits (intramodule). Logic, bilevel, or analog signal and/or control circuits which operate at voltage levels of 5 volts, or greater, and which are wholly contained at the module level shall be multiple-grounded at both source and load end of each circuit branch with return paths through the module radiator panel and support structure. However, if the case of the load unit, such as a thruster valve, cannot be readily provided with a low impedance electrical bond to structure, single-point grounding may be used for that circuit with the ground-point at the line driver element.

3.3.2.2.7 High level signal/control circuits (intermodule). High level signal and/or control circuits which originate in one module and terminate in another shall be single-point grounded, normally in the driving module. Where two or more subcontractors are effected, the location of the circuit ground point shall be coordinated by the systems integration contractor.

3.3.2.2.8 Low level analog circuits. Any analog sensor circuits operating at less than 5 volts, which are shown by circuit analysis or test to be sensitive to circulating currents in the module or spacecraft structure, shall be single-point grounded either at the source or load end, whichever is most appropriate for the particular circuit under consideration. Wherever possible, balanced differential circuitry shall be used for circuits having low level sensors with high gain amplifiers. If the circuit transcends a module interface, the location of the circuit ground point shall be coordinated by the systems integration contractor.

3.3.2.2.9 Thermistor monitoring circuits. Thermistor circuits used to monitor structure temperatures shall be single-point grounded in the signal conditioning assembly.

3.3.2.2.10 Pyrotechnic circuits. Prior to receipt of a valid command, the firing device shall ground both sides of the pyrotechnic circuit. The circuit between the firing device and the initiator bridge wires shall be undergrounded, twisted shielded pairs. The shields shall provide a minimum of 40 dB isolation to external RF fields as required by AFWTRM 127-1.

3.3.2.2.11 Wire shields. With the exception of primary DC power and thermal wiring harnesses, external shields shall be provided for all exposed interconnecting wiring between modules, assemblies, units and/or components of the Observatory.

3.3.2.3 Electrical bonding. The electrical bonding configuration of the Observatory shall be designed to provide maximum electrical conductivity across all mechanical joints between metallic members except where DC isolation is a design requirement. The standardized bonding provisions of MIL-B-5087, shall be implemented. Also, interfacing metals shall be suitably treated with conductive protective coatings to prohibit the deteriorization of electrical bond joints through electrolytic or galvanic action. The design goal of the electrical bonding configuration is to provide a low impedance, electrically continuous, homogeneous ground reference plane at the module level.

3.3.2.3.1 Structural bonds. All metallic structural bonds. All metallic structural members of the Observatory modules shall be electrically bonded to each mating member to form a continuous, equipotential ground plane. The maximum DC impedance across any individual mechanical joint shall be 2.5 milliohms with an overall design goal of 10 milliohms, or less, between any two points on the module structure and radiator panel.

3.3.2.3.2 Equipment mounting surfaces. Surfaces on the module radiator panel and support structure which are intended for unit, equipment or component mounting shall be free of paint, anodize or other non-conductive finishes. Each component chassis shall be directly bonded to the mounting surface on the structure by contact pressure across the entire footprint of the component. The maximum DC impedance between the baseplate of any electrically active unit, equipment or component and the mating structure shall be 2.5 millohms.

3.3.2.3.3 Electrical connectors. All interface electrical connectors, both plug and receptacle, which form a part of the unit and cable RF shielding system shall have electrically conductive body shells, free of non-conductive finishes. They shall have provisions for terminating (ground) the shields on the interconnecting electrical harness. The maximum DC impedance between the shield termination point and the baseplate of the parent unit shall be 2.5 milliohms.

3.3.2.3.4 Electrostatic bonds. Electrically passive components and appurtenant metallic structures which are attached to the module or Observatory structure through thermal isolators shall be provided with electrostatic bonding jumpers having a DC impedance of 1.0 ohm, or less.

3.3.2.3.5 Module/spacecraft interface. No intentional power, signal or control return currents shall pass through the module to spacecraft interfaces via the mating structural elements therefore a direct RF bonding requirement will not be specified; however, to avoid electrostatic and unintentional RF potentials across this interface the DC impedance shall be minimized but in no case shall it be greater than 1.0 ohm.

3.3.2.4 Interconnect wiring/harness. The interconnect wiring between units, equipments or components within the modules and between modules of the Observatory shall be categorized into the functional groups described in paragraph 3.3.2.2. Wherever practical, unshielded primary DC and heater power cables shall be individually routed and physically separated from the shielded, signal/control cables. If different functional groups must share common interface connectors, pin assignments shall be chosen to minimize circuit to circuit coupling. Under no circumstances shall pyrotechnic circuits share common connectors or cable routing paths with other functional wire groups between the firing device and the initiator bridge wire. The following wire types shall be used for the indicated function groups:

(a) Twisted pairs, unshielded

- Primary DC power
- Heater power

(b) Twisted pairs, shielded

- Data bus
- Secondary AC power
- Intermodule signal/control circuits
- Thermistor monitors
- Low-level analog circuits

(c) Twisted pairs, double shielded

- Pyrotechnic circuits

(d) Single conductor, shielded

- Secondary DC power
- Intramodule signal/control circuits

(e) Coaxial cable

- RF circuits

3.3.3 <u>Nameplates and product marking</u>. Observatory components shall be marked for identification in accordance with MIL-STD-130. Marking shall include:

(a) Item name

(b) Integrating contractor's part number (if different from manufacturers part number)

(c) Manufacturer's part number

(d) Manufacturer's serial number

(e) Date of initial acceptance

(f) Manufacturer's name

(g) Contract number

(h) Actual weight in pounds to at least ±0.1 lb.

Character height shall be at least 0.09 inch minimum where possible.

3.3.3.1 Fluid and gas lines. Fluid and gas tubing shall be identified and marked in accordance with MIL-STD-1247, except for lines which do not have adequate space for such marking.

3.3.3.2 Pyrotechnics. All propellant grains, ignitors, squibs or ordnance charges shall be classified and marked in accordance with procedures as defined in AFWTRM 127-1.

3.3.4 Workmanship

3.3.4.1 Workmanship standards. The Observatory and components shall be designed and constructed in compliance with Requirement 9 of MIL-STD-454.

3.3.4.2 Personnel certification. Personnel involved in assembly, soldering, welding or other activity requiring special skills shall be certified as to their capability to perform such operations effectively in accordance with the procedures specified in EOS-4.1.

3.3.4.3 Process certification. Machines, equipment and procedures used in selected process operations shall be certified to perform their required functions in accordance with the procedures specified in EOS-4.1.

3.3.5 Interchangeability. Observatory design shall meet the requirements of MIL-STD-100 for interchangeability and replaceability. In general, all modules of similar function shall mate interchangeably with their designated structural and electrical attachment provisions on the spacecraft and transition ring structural of electrical assembly.

3.3.6 Safety

3.3.6.1 General. The Observatory shall be designed in accordance with the requirements of MIL-STD-882 and AFWTRM 127-1 as implemented by EOS-4.1 and EOS-3.3-7 to:

(a) Limit or eliminate inherent personnel hazards as described in Table 4.

(b) Operate in a manner that will limit or preclude single failures, malfunctions, premature operation or personnel error which might result in damage to equipment or injury to personnel.

Table 4. Inherent Personnel Hazards and Controls

Hazard	Source of Hazard	Hazard Effects	Controls
RF radiation	Spacecraft antennas	Hazardous effects as defined in the Safety Design Criteria.	Control power density to preclude exposure to radiation exceeding TBD watts/m^2
Pneumatic explosion of fragmentation	Propellant tanks	Hazardous effects as defined in AFSC DH 1-6	Relatively low pressure and volume. Proof tests, loading procedures, protective clothing. Design per AFWTRM 127-1.
Flammable/ explosive	EED's	EED's ICC Class C	Design per AFWTRM 127-1 and MIL-STD-1512. Handling and storage per AFM 127-100 and AFM 71-4.
Flammable, corrosive and toxic	Liquid propellant hydrazine (N_2H_4)	DOT corrosive liquid Military Class I, Group IIIC Threshold limit value 1 ppm by volume. Flammable limits 4.7 to 100 percent.	Protective clothing, special handling equipment, and loading procedures, transportation, and storage per AFWTRM 127-1, 160-39. Design - of propellant tanks, tubing valves, components per AFWTRM 127-1.

3.3.6.2 Pyrotechnic devices

The Observatory design shall include appropriate provisions for pyrotechnic devices. Electroexplosive devices (EED's) and associated firing harnesses shall be designed in accordance with MIL-STD-1512 and applicable criteria in AFWTRM 127-1, and the requirements as specified herein. RF shielding caps shall be provided on EED's during shipment, storage, handling and installation up to the point of electrical connection in the Observatory. During all handling modes, before installation in the Observatory, EED's shall be installed in safeguard devices designed to safely contain the squib in case of inadvertent initiation.

3.3.7 Human Engineering

3.3.7.1 General. The Observatory shall be designed in accordance with the requirements of MIL-STD-1472A.

3.3.7.2 Observatory control. Observatory command execution shall support operator-controlled ground-initiated command capability by providing:

(a) Unambiguous routing of commands to the proper equipments.

(b) On-board lock-out and fail-safe features in the event that improper commands are received at the Observatory.

3.3.7.3 Observatory information. The design of Observatory telemetry shall support ground processing and operator data interpretation such that:

(a) Equipment on/off status resulting from ground-transmitted commands can be presented unambiguously from telemetry data.

(b) Telemetry format provides ease of ground processing and subsequent operator interpretation.

(c) Anomalies can be readily analyzed from processed telemetry data.

3.4 Documentation

3.4.1 Specifications and drawings. Observatory specifications and and drawings shall be prepared in accordance with EOS-3.3-8.

3.4.2 Test plans and procedures. Test plans and procedures shall be in accordance with EOS-4.2.

3.5 Logistics

3.5.1 Maintenance at the launch site. Maintenance of the Observatory at the launch site shall be accomplished as follows:

(a) Servicing and loading of cold gas and liquid propellants

(b) Servicing of batteries

(c) Checkout at the system level at the Observatory Assembly Building.

(d) Removal and replacement of failed modules

3.5.1.1 Servicing and testing after mating. Servicing and testing of the Observatory after mating shall be minimized.

(a) Subsequent to fairing encapsulation, only critical AGE connections shall be permitted.

3.5.1.2 Installation of electro-explosive devices. Class "B" or "C" EED's shall be installed either at the explosive safe area or prior to shipment of the Observatory from the factory. If the Observatory is shipped with EED's installed, proper shielding and shielded connectors shall be used.

3.5.2 Supply. Observatory spare components shall be provided from available factory extra units and from subcontractor delivered spares. Hydrazine will be obtained as GFE.

3.5.3 Facilities and facility equipment. TBD

3.6 Personnel and training

3.6.1 Personnel. Not applicable.

3.6.2 Training. Not applicable.

3.7 Major component characteristics

3.7.1 Instrument payload. The instrument payload requirements will be mission dependent and will be separately specified for each mission. The instrument requirements and characteristics for an example low earth orbit, earth observation satellite (EOS) mission are contained in the following documents:

(a) Report 2, Instrument Contraints and Interface Specifications, TRW Document 22296-6001-RU-01.

(b) Wideband Communications Module Specification, SP-1124.

The wideband communication module may be applicable to a number of missions and the specification noted in (b) above shall be complied with to the extent specified in the mission requirements documentation.

3.7.2 Communications and data handling module. The communications and data handling module shall perform and have the characteristics as specified in SP-1112.

3.7.3 Power module. The power module shall perform and have the characteristics as specified in SP-1113.

3.7.4 Attitude determination module. The attitude determination module shall perform and have the characteristics as specified in SP-1114.

3.7.5 Spacecraft structure/thermal assembly. The spacecraft structure/thermal assembly shall perform and have the characteristics as specified in SP-1111.

3.7.6 Actuation module. The actuation module shall perform and have the characteristics as specified in SP-1115.

3.7.7 Solar array and drive module. The solar array and drive module shall perform and have the characteristics as specified in SP-1116.

3.8 Precedence. The requirements of SPM-XX shall have precedence relative to requirements set forth in this specification.

Other military specifications, standards, manuals, and non-government documents listed in Section 2, shall be applicable to the extent specified in this specification.

4. QUALITY ASSURANCE PROVISIONS

4.1 General. The quality assurance program controls shall be in accordance with the requirements of MIL-Q-9858 as augmented by EOS-4.1.

4.1.1 Responsibility for inspection and test. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified herein. Except as otherwise specified, the supplier may use his own or any commercial facilities acceptable to the government. The government reserves the right to perform any of the inspections set forth in the specifications where such inspections are deemed necessary to ensure that supplies and services conform to prescribed requirements.

4.2 Quality conformance inspections

4.2.1 Category I tests

4.2.1.1 Development tests. Development tests shall be conducted to verify the module performance parameters, proper interfacing between components of the module, and proper interfacing between the module and other spacecraft modules. The development tests shall be conducted at the unit and/or integrated module levels. The type of developmental evaluation or test to be applied to verify satisfaction of the requirements defined in this specification are outlined in Table 5. The tests shall be conducted in accordance with EOS-4.2 and EOS-4.6.

As each payload instrument is integrated, interface signal characteristics shall be measured to verify design margins and compatible operation with the spacecraft data bus and power bus. These tests shall be conducted in accordance with EOS-4.2.

4.2.1.2 Qualification tests. Module qualification shall be accomplished by means of qualification tests on the individual units and/or as a normal consequence of having the units installed in the qualification test space vehicle during its qualification testing. The

Table 5. Verification Cross Reference Index

VERIFICATION CROSS REFERENCE INDEX

LEGEND: N/A - Not Applicable; I - Inspection; A - Analysis; S - Similarity; T - Test

SECTION 3 REQUIREMENTS	Not Applicable	Acceptance			Qualification				SECTION 4 VERIFICATION REQUIREMENTS
		I	A	T	I	A	S	T	
(TBD)									

SAMPLE

types of qualification evaluation or test to be applied to verify satisfaction of the requirements of this specification are outlined in Table 5. Qualification test verification methods and requirements shall be as defined in EOS-4.2.

4.2.1.2.1 Components. As a minimum, the following types of component qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, are measured before and after the environmental tests.

- Random vibration. Each component will be subjected to a random vibration from 20 to 2000 Hz in all three axes while electrically powered in the maximum normal load condition. Performance will be continuously monitored to detect failures (either hard or trends).

- Thermal vacuum. Thermally cycle the component beyond expected orbital temperatures while at an orbital vacuum condition. The component is operating continuously and detailed performance parameters are measured at each temperature extreme.

- Electromagnetic interference (EMI) and susceptibility (EMS). Engineering EMI and EMS data are measured to provide diagnostic information for the module EMI/EMS qualification.

4.2.1.2.2 Module. As a minimum, the following types of module qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, and electrical interface characteristics are measured before and after the environmental tests.

- Acoustics. Acoustics testing will be performed with the module mounted in a receptacle which provides acoustics shielding similar to the Observatory structure. The module will be heavily instrumented to verify design adequacy of interconnections (electrical and mechanical) between the module and other Observatory elements. The test data will be used to confirm estimates of the vibration environment for components mounted on radiators and establish procedures for acceptance testing. The module will be electrically powered and continuously monitored via the data bus to detect failures and out-of-tolerance performance trends. The solar array will be excluded from this test.

- Thermal vacuum. The objectives of the module thermal qualification test are to determine: 1) maximum and minimum operating temperatures for the module for worst-case environments and operating duty cycles, 2) temperature levels and temperature variations of the module adjacent to the interface fittings, and 3) heater power requirements for cold-case conditions.

  The test will be conducted in a thermal-vacuum chamber with liquid nitrogen cold walls. The module will be mounted on a fixture simulating the spacecraft frame. The properties of this fixture shall permit its temperature to be held constant at any level between 0 and 100°F.

  A heating source for the radiators will approximate the absorbed flux of the external environment. This can be done with electrical heaters, infrared lamps, or other techniques where the absorbed heating can be determined accurately.

  - Power bus and data bus will be tested in excess of their operational limits to determine design margins and compliance with the interface specification.

  - Detailed performance data will be measured to determine module specification values.

  - Thermistor/heater control and calibration will be determined.

- EMI and EMS. EMI measurements will determine the levels radiated on each module electrical interface and verify there is adequate design margin to ensure non-interference with other modules. EMS measurements will verify that each module has adequate susceptibility margin to prevent performance degradation resulting from other module allowable radiation levels. The general test methods of MIL-STD-462, as modified or amended by EOS - 3. 3-5, shall be used during this test program except that, wherever practical, spectrum analyzers or other forms of rapid display and analysis equipment may be used in lieu of equipment requiring manual tuning and data collection.

4.2.1.2.3 Observatory. As a minimum, the following types of observatory qualification tests shall be included:

- Functional. System performance parameters will be measured before and after the environmental tests. The functional tests consist of: 1) integrated system test (IST), 2) detailed subsystem tests, 3) detailed instrument tests, 4) power load determination (before only), 5) deployment tests, 6) alignments determination, 7) leak test, and 8) solar array illumination. In addition, weight and center-of-gravity design is measured.

- Low-frequency sine vibration. The Observatory will be subjected to sine vibration at levels less than 1.1 times limit load over the range of 5 to 100 Hz in all three axes. The Observatory will be heavily instrumented to verify the analytical model used for loads prediction and design adequacy of primary and secondary structure and module connections. The Observatory will be electrically powered in the launch mode. RF telemetry data will be monitored continuously to verify the electrical system performance and design adequacy of the electrical interconnections between modules.

- Acoustics. The Observatory will be subjected to an acoustics test while mounted vertically on an integration and test pedestal. Extensive instrumentation will be used to: 1) verify design adequacy of the solar array and other non-module components, 2) confirm estimates of the vibration environment for components mounted on walls other than radiators, and 3) confirm adequacy of the module receptacle used for module-level acoustics test. The Observatory will be electrically powered in the maximum normal load condition and continuously monitored via the RF telemetry link to detect performance degradation.

- Shock. All ordnance (separation system, pin pullers on the array, antennas, and module supports) will be fired to verify design adequacy of all Observatory components. The Observatory will be electrically powered in the appropriate mode during the firing. RF telemetry will be monitored to detect performance degradation.

- Thermal vacuum. The Observatory thermal control test has the basic objectives to evaluate the modular testing concept and the thermal control system. Evaluation of modular testing requires correlation of system and module test results. Evaluation of the thermal control system entails the following: 1) structure thermal control, 2) module/structure interaction, and 3) required heater power.

  The thermal vacuum test will be conducted in a thermal vacuum environmental chamber with an LN_2 cold wall. The -Z side will be irradiated with a heat source that can be accurately defined (solar simulation will not be necessary). The +Z side will face the cold wall; no attempt will be made to simulate the external energy input (earth emission, albedo, and solar). This test method will allow an accurate thermal definition of the chamber environment. The test conditions will include a cooldown phase to evaluate heat leaks, a steady-state phase to evaluate heater requirements, and a transient phase to evaluate interface interactions.

Throughout the test, detailed temperature data will be measured to verify the thermal analytical model. Design adequacy of the thermal insulation, heater control, and thermistor placement will be determined. In addition, thermistor calibration and methods for thermal evaluation and control by the ground station will be analyzed. At each thermally-stabilized level, integrated systems test and sensor (ADM and instrument) aliveness tests will be performed. Telemetry data will be monitored continuously to verify design margins of all subsystems.

- Electromagnetic compatability. Noise levels will be measured on critical signals (including interfaces) utilizing the module test connectors. This will verify module design adequacy and validate the EMI/EMS interface criteria established for module-level qualification.

4.2.1.2.4 Qualification by similarity. With concurrence of the procuring agency, qualification by similarity shall be acceptable at the component level where the component design or mounting has not significantly changed and the component has been previously qualified under applicable environmental conditions.

4.2.1.2.5 Inspection sequence. The inspection sequence shall be governed by the following:

- Examination of the module shall be performed prior to functional testing.
- Functional tests shall be performed prior to, during, where appropriate, the following environmental testing. The functional testing conducted at the conclusion of an environmental test may serve as the functional testing to be performed prior to the next environmental test.
- Environmental testing may be performed in any sequence.

4.2.1.2.6 Failure criteria. The module shall exhibit no failure, malfunction or out-of-tolerance performance degradation as a result of the examinations and test specified herein. If a failure occurs during the performance of any test, the test shall be suspended and the discrepancy, failure reporting, analysis, and corrective action procedures as set forth in EOS-4.1 shall be followed.

4.2.2 Test conditions. Unless otherwise specified, the conditions under which all inspections are accomplished shall be specified below:

4.2.2.1 Atmospheric and environmental. All examinations and tests shall be conducted under the local prevailing pressure at the time of test, at a temperature of 75 ±5 °F and at a relative humidity of 55 percent or less.

4.2.2.2 Deviations of atmospheric conditions. When tests are performed with atmospheric conditions substantially different from the specified values, proper allowances for changes in instrument readings shall be made to compensate for the deviation from the specified conditions.

4.2.3 Equipment warmup time. The equipment warmup time shall be less than 1 minute.

4.2.4 Temperature stabilization. Temperature stabilization shall have been achieved when temperature of the equipment mounting structure varies not more than 5 °F during a period of 15 minutes.

4.2.5 Measurements. Measuring instruments used to determine functional parameter values (such as voltage, frequency, current, flow, pressure, etc.) shall indicate true values with an accuracy determined by the tolerance allowed for the parameter variation itself, such that the measuring instrument shall not introduce an uncertainty greater than 10 percent of the allowable variation of the measured parameter. However, no such measurement accuracy shall be required to exceed 0.5 percent of the required value of the parameter unless otherwise specified.

Test instrumentation and equipment shall comply with the requirements of MIL-C-45662.

Except as specifically noted in the inspection methods, tolerances for environmental test conditions shall be defined as follows:

Temperature: ±5 °F

Barometric pressure: ±10 percent

Relative humidity: +5, -10 percent

Time: ±5 percent

Vibration amplitude (sine): ±10 percent

Vibration frequency: ±2 percent

4.3 Category II tests. (Not applicable)

4.4 Analyses. The following requirements of Section 3 shall be verified by review of analytical data:

Reliability. To be verified by analysis in accordance with Section 3.2.3 of EOS-4.1.

Safety. To be verified by analysis in accordance with Section 3.3.6 of EOS-4.1.

5. PREPARATION FOR DELIVERY

5.1 General. The Observatory shall be protected from degradation by environments anticipated during shipment, handling, and storage. Standard commercial packaging practices are acceptable provided they fulfill these requirements.

6. NOTES

6.1 Definition of spacecraft configuration.

6.1.1 Minimum redundancy configuration. The minimum redundancy configuration is defined as the spacecraft configuration which contains the minimum redundancy of units necessary to ensure that no plausible single-point failure will prevent Observatory retrieval by the Space Shuttle system. For purposes of this specification this configuration is identified as the baseline spacecraft configuration.

6.1.2 Nominal redundancy configuration. The nominal redundancy configuration is defined as the spacecraft configuration which includes standby redundant units for most of the electronic assemblies to provide a "typical" redundancy level for long-life spacecraft.

TITLE

PRIME ITEM DEVELOPMENT SPECIFICATION
FOR THE

CENTRAL DATA PROCESSING FACILITY

DATE 20 SEPT 1974 NO. SP-312

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

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SPECIFICATION SP-312

CENTRAL DATA PROCESSING FACILITY

1. SCOPE

1.1 Scope. This specification establishes the performance, design, test, and qualification requirements for a Central Data Processing Facility (CDPF) for the Earth Observatory Satellite (EOS) System.

1.2 Purpose. The purpose of this specification is to define specific efforts and items of equipment for NASA. This effort and equipment is to be applied toward the procurement of a CDPF which will process recorded telemetry data into formats compatible with user requirements. The station is an integral element of the EOS ground system as described in EOS System Specification SP-1. When incorporated into the ground system the facility will provide a data services laboratory for generating user products from outputs of the image processing subsystem, an information management system for controlling and scheduling operations, and an image processing subsystem whose outputs are corrected image positive/negative transparencies, high density digital tape (HDDT's), computer compatible tape (CCT's), and incorrected HDDT's.

2. APPLICABLE DOCUMENTS

2.1 Government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

SPECIFICATIONS

NASA

SP-1	EOS System Specification

Military

MIL-B-5087B	Bonding, Electrical, and Lightning Protection for Aerospace Systems

STANDARDS

Military

MIL-STD-130D	Identification Marking of U. S. Military Systems
MIL-STD-1472A	Human Engineering Design Criteria for Military Systems Equipment and Facilities

2.2 Non-government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. If no revision or data is shown, the latest released issue of the applicable document shall apply. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

EOS-3.3-8	Configuration Management Plan
EOS-4.1	System Effectiveness Program Plan

3. REQUIREMENTS

3.1 Item definition. The CDPF receives wideband sensor data from three receiving stations in the form of wideband tape recordings. Auxiliary spacecraft and calibration data is supplied by the NASA Orbit Determination Group (ODG) and the Control Center (CC). The CDPF processes and transmits user and LCGS requests for coverage to the CC mission planning function, and transmits output film and magnetic tape products to NASA investigators and to the EROS data center for dissemination to other users. The CDPF also provides a 70-day archive of tape and film products for NASA use. Deep archiving (>70 days) is relegated to EROS.

The CDPF functional structure can be further broken down as shown in Figure 1. The CDPF is composed of three operating subsystems:

(a) Data Services Laboratory (DSL)

(b) Information Management System (IMS)

(c) Image Processing System (IPS).

The DSL includes the management interface, user services, 70-day archiving, photographic production, and output product dissemination, and represents the user interface to the EOS data. The IMS includes the Control Center (CC) interface, the data management system, processing control, and represents the monitoring and control functions for internal CDPF data handling. The IPS includes the image correction and formatting functions and the output product generation.

The CDPF functions can be grouped into those which interact directly with the sensor data and those which do not. Most of the IMS and DSL (except for photo production and shipping) do not; dealing instead with summary characteristics of the data, such as catalogs and data routings. The existing ERTS IMS and DSL functions perform the same tasks as those required for EOS for approximately the same coverage cycle time and orbit. Data load increase is in the number of spectral bands and number of picture elements per scene, not in the number of scenes. Offloading of the CDPF user interface volume to LCGS and EROS provides the same or lower expected loading on the DSL as experienced in ERTS. Coverage

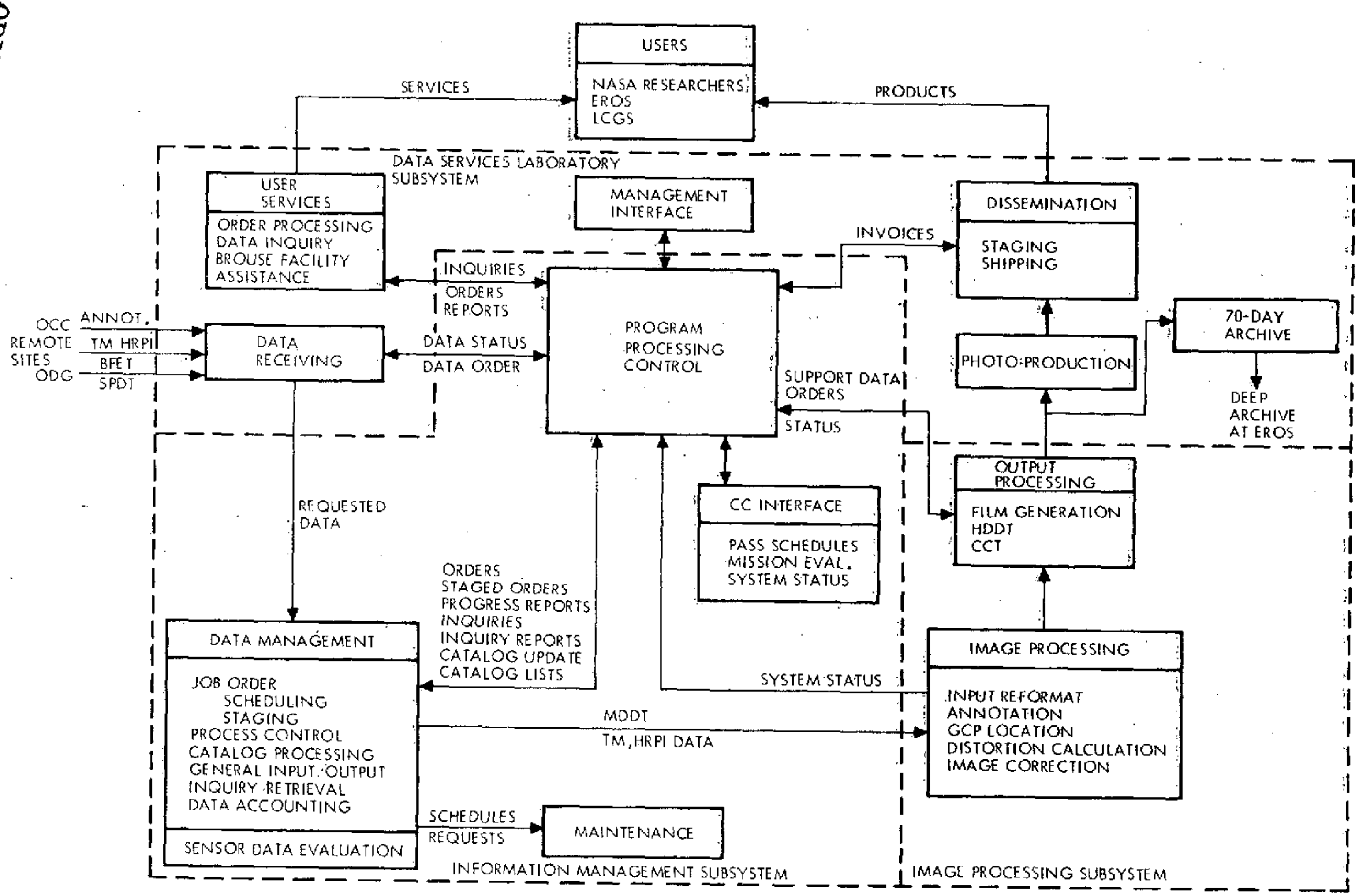

Figure 1. CDPF Functional Structure

requests in ERTS are for on-board tape recorder scheduling, while for EOS, scheduling is required for HRPI and LCGS coverage requiring a slightly greater requirement for mission planning. The mission planning function is contained within the Control Center with the DSL interfacing user coverage requests to the CC.

The current ERTS NDPF system shall be used as a baseline from which the EOS IMS/DSL software configuration is established. Most of the major information management functions and some of the computational subsystem functions shall be retained but augmented/modified as required to meet EOS requirements. Figure 2 shows an NDPF software overview with modification and replacement requirements designated.

The current NDPF functions are performed on a single XDS Sigma-5 computer with 84 K memory, 3 disk drive units, and 10 tape drive units. The photo production facility is sized to produce 860 MSS scenes/week and 860 RBV scenes/week with 36,000 positive/negative transparencies for each. The EBR film recorder is not considered applicable due to the increased performance requirements of EOS, but the photo production facility is adequate.

The baseline DSL/IMS shall use the existing Sigma-5 hardware with added core memory, all applicable ERTS IMS software, and the ERTS photo production facility.

3.1.1 Interface requirements. The CDPF baseline configuration is designed to handle 50 thematic mapper (TM) users and 200 HRPI scenes (the equivalent number of 50 TM scenes) every day. This corresponds to 22.5 minutes of sensor operation each day for the baseline sun-synchronous orbit of 717 km altitude. This represents an upper-bound on the processing load as there will be at most 35 TM scenes per day covering CONUS. There are at most three sensor data acquiring passes over CONUS each day, with the maximum pass duration being on the order of 10 minutes. The rest of the scenes will cover data taken at the Alaska ground station and other available scenes taken outside CONUS from the Goldstone and NTTF ground stations. Considering the three CONUS passes and the Alaska ground station data, there are an average of five passes each day during which sensor data is acquired. Table 1 summarizes the characteristics of the CDPF input sensor data.

There are three types of CDPF output products: high-density digital tapes (HDDT), computer-compatible tapes (CCT), and film. There is also the requirement for an archival product to be used for sensor data storage and retrieval. The HDDT products include both uncorrected and corrected sensor data. All of the CCT products are to be corrected. First-generation film products shall be positive and negative black and white images; the color positives and negatives are second-generation products. Photo processing will produce the additional black and white and color film products.

Output data product quality will be either uncorrected, reformatted sensor data, or it will be corrected to a quality that will satisfy the most demanding user requirement. Table 2 summarizes the data quality characteristics to be used for sizing the CDPF design.

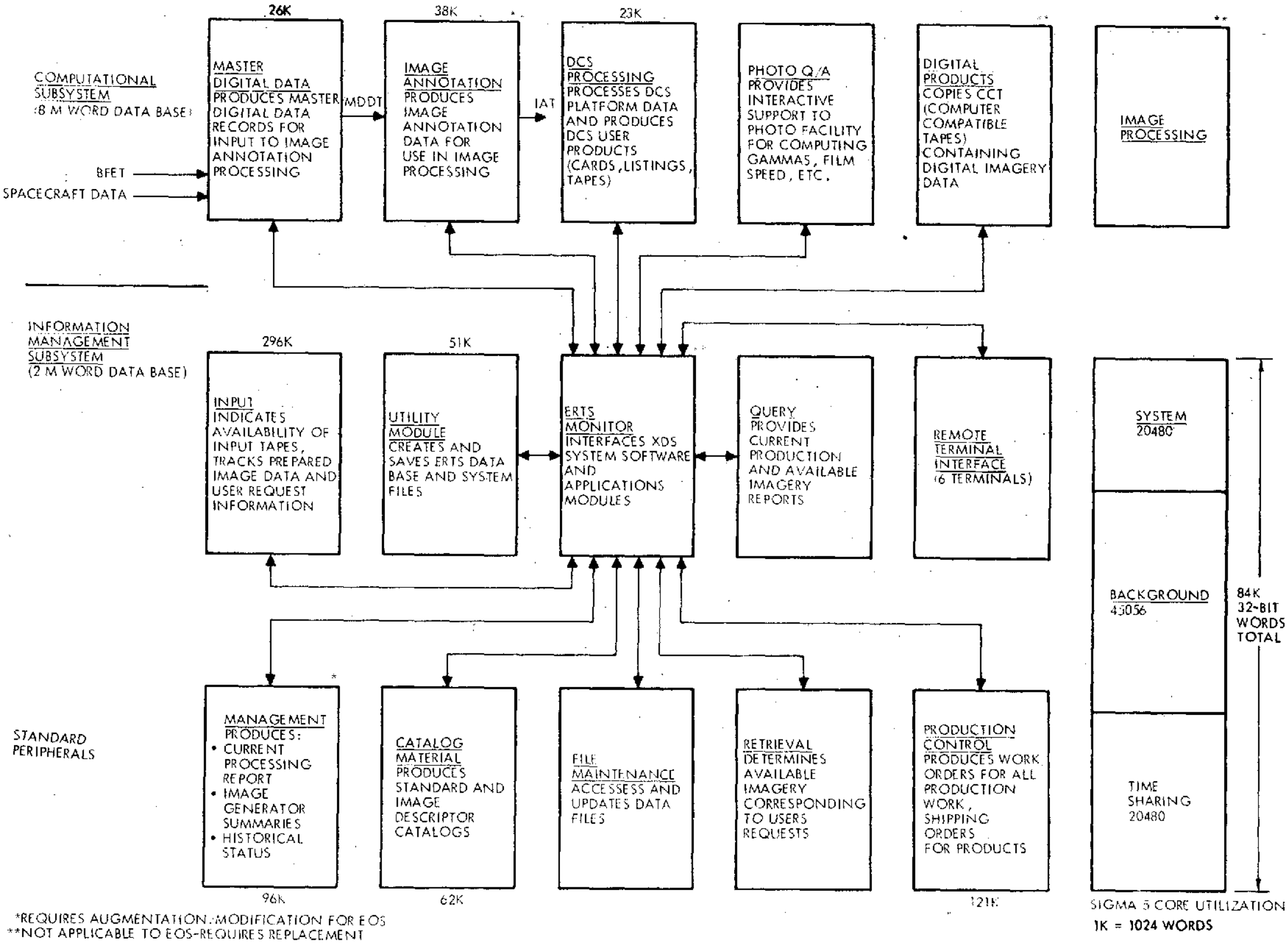

Figure 2. NDPF System Overview

Table 1. Characteristics of Input Sensor Data Used for Sizing CDPF Design

Input Sensor Product		
Average number of daily sensor data ground station passes		5
Average total duration of sensor data		22.5 minutes
Maximum station pass duration		10 minutes
Number of HDMR tapes for each pass (one for each sensor)		2
Average daily number of HDMR tapes		10
HDMR tape format: Each 14-inch, 9600-foot tape contains the data from one sensor and is recorded at 180 ips. The peak data rate is 128 Mbit/sec, and has a serial output. The average sensor data rates are 11.4 Mbytes/sec and 13.5 Mbytes/sec for the TM and HRPI, respectively, with 8 bits per byte representing each pixel.		
Input Sensor Data Daily Volume		
TM scenes (1.22 x 10 in. bits)		50
HRPI scenes (1.47 x 10 in. bits)		200
Total average daily sensor data input		2.69 x 10 in. bits
Input Sensor Data Format Parameters		
	TM	HRPI
Bits per pixel	8	8
Pixels per scan	8192	4800
Spectral bands (ground resolution)	No. 1-6 (30m), No. 7 (120m)	4 (10m)
Swathwidth	185 km	48 km
Detectors per scan	100	19,456
Pixels stored to form line	813K	58K
Pixels per scene	306M	92M

Table 2. Output Data Quality Characteristics Used for CDPF Design

Uncorrected Output Products:	Quality commensurate with that inherent in the input sensor data; only processing is reformatting.	
Corrected Output Products:		

Attribute	Sensor TM	Sensor HRPI
Pixel misalignment (3σ)	1/4 pixel	1/4 pixel
Band-to-band registration (3σ)	0.1 IFOV	0.3 IFOV
Position accuracy (1σ)	15 m	15 m
Relative radiometric accuracy (3σ)		
Visible Bands		
Tape	1.6%	1.6%
Film	5.0%	5.0%
Thermal Band		
Tape	$1^{o}K$	Not applicable
Film	$3^{o}K$	Not applicable

The principal user groups receive the output products: NASA principal investigators and EROS. The principal investigators are selected members of the scientific community who will require special and unique output product formats as well as the standard formats to satisfy their collective requirements. The output products that are shipped to EROS (or other similar facilities) will be used to satisfy the requirements of the general public and operational programs. The digital data interface to EROS will be HDDT's because of their high-data densities.

All the digital data contained on HDDT's shall be contained in three formats: uncorrected, band-interleaved corrected, and band-separated corrected. All HDDT formats shall have error correction coefficients and image annotation files as headers.

The equivalent of all the input sensor data shall be stored in each of the three formats. One copy of each of the formats will be sent to EROS and one copy of each will be retained by the CDPF for archiving. In addition, the equivalent of two copies of the uncorrected HDDT (with the error correction information in the header) will be sent to LCGS users for their unique processing requirements. This will result in a daily data volume of 1.1×10^{12} bits for uncorrected HDDT's and the same for the corrected HDDT's. Figure 3 shows the baseline GDHS interfaces.

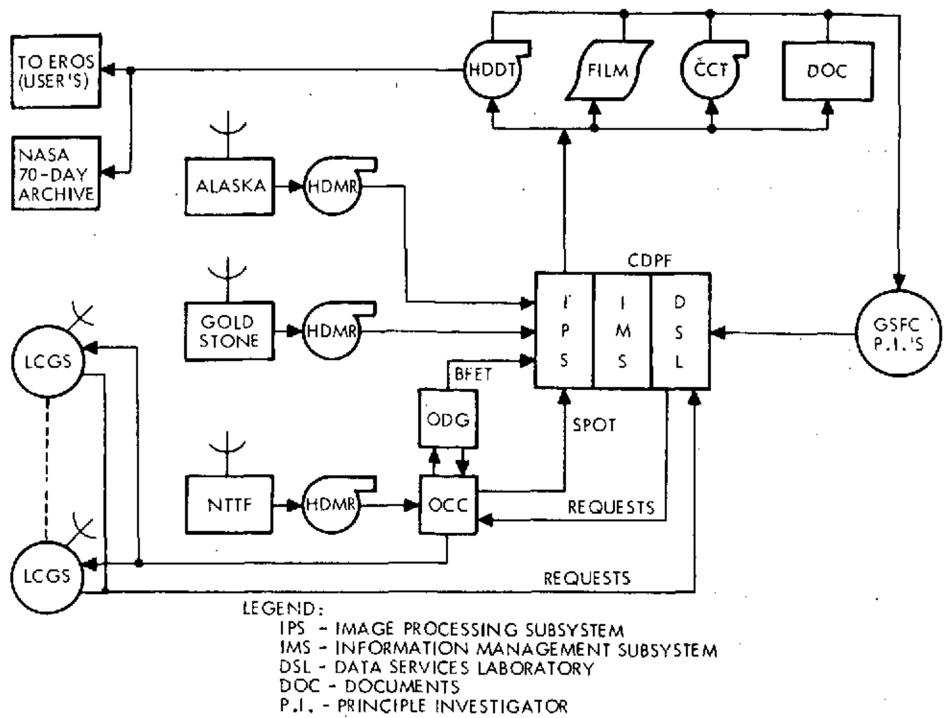

Figure 3. Baseline GDHS Interfaces

Two CCT technologies are used, 1600 and 6250 bits/inch (bpi). This will be the interface medium for digital data used by NASA principal investigators and a variety of formats are required. Standard formats shall include band-interleaved, band-separated, pixel-interleaved by band, 1/16 of TM swathwidth (512 pixels for standard displays), 1/4 swathwidth TM (for use with HRPI data), and 1/2 and 1/4 swathwidth HRPI formats. It should be recognized that by the very nature of work performed by the user community it is essential that there be enough flexibility to satisfy almost all formats anticipated. The users of larger data volumes will capitalize on the 6250 technology. One TM scene can be read onto four 2400-foot reels of 6250 bit/inch tape; the same medium can hold one HRPI scene. However, it requires at least nine 1600 bits/inch tapes to hold a TM scene and at least three tapes for a HRPI scene.

There is a requirement for both positive and negative first-generation black and white transparency film products. The products will consist of all scenes of each sensor. Color positive and negative transparencies are specified to be second-generation products. The film products are specified to have a 241 mm (9.5 inch) size. A number of output formats are required. Because the frame size of the TM is equivalent to the ERTS MSS frame size, there will be a continuation of the existing formats for enlargement to standard map scales, e.g., 1:1,000,000. The HRPI scene can be enlarged about four times the TM scene because of its reduced field of view and the fixed 241-mm film size. Table 3 summarizes the output product volume.

Table 3. CDPF Output Characteristics for Sizing Product Generation

Product	Average Daily Data Volume	Format	Comments
HDDT			Error correction and image annotation information used as header on all HDDT products
Uncorrected	1.1×10^{12}	Band-interleaved	Four copies of all data
Corrected	1.1×10^{12} bits	Band-interleaved and band-separated	Band-separated product annotated and formatted for film generation
CCT	1.5×10^{10} bits	1600 and 6250 bpi	Equivalent to 6 TM scenes; flexibility in generation to satisfy all formatting needs
Film		All 241 mm (9.5 inch) film size	Formats include enlargements to standard map scales including 1:1,000,000 1:1,500,000, and 1:250,000
First-generation black and white	2300 images		Positive and negative transparency of all error corrected scenes; 50 TM, 200 HRPI
Black and white positive transparency	5750 images		
Black and white negative transparency	11500 images		
Black and white positive prints	575 images		
Color positive transparency	58 images		
Color negative transparency	115 images		
Color positive print	80 images		

Archive Material: One copy each of the three HDDT formats ..... 8.3 x 10 in. bits (data volume) included in HDDT volumes above)

The archive material is one copy each of the three formats of the HDDT. These HDDT's represent master copies and are used to generate requested output products.

3.1.2 Major components. The major components of the CDPF consist of:

(a) The existing NDPF Data Services Laboratory (DSL) as modified by this specification.

(b) The existing NDPF Information Management System (IMS) as modified by this specification.

(c) The Image Processing System (IPS) which is specified in this specification and which replaces the existing NDPF image processing function. It consists of three subsystems to perform input reformatting of TM and HRPI information from the wideband sensor data recorder tapes (HDMR), distortion calculation and geodetic control point (GCP) determination, image correction, and output product generation. Each subsystem shall contain common computer equipment and peripherals to the maximum extent possible. In addition to the three processing subsystems, a fourth subsystem shall allow switching of HDMR and HDDT tape units between the other subsystems.

3.2 Characteristics

3.2.1 Performance

3.2.1.1 Processing subsystems

3.2.1.1.1 Common equipment. The following components are common to each of the three processing subsystems.

3.2.1.1.1.1 Tape subsystem. The tape subsystem shall consist of:

(a) An HDMR 3 x 3 matrix switch which will connect any of the three HDMR tape units to either of three HDMR interface units as shown in Figure 4. Both data and control shall be switched with no signal degradation. Switching shall be accomplished by either manual or automatic selection by any one of the three subsystem CPU's. The choice of the controlling CPU shall be made manually at the matrix switch. Automatic switching shall be inhibited whenever such inhibition is required to prevent data loss or degradation. A control line capable of generating an external interrupt shall be provided to the controlling CPU to identify periods when switching is enabled.

(b) An HDDT 20 x 10 matrix switch shall be provided with the same characteristics as the HDMR switch of paragraph 3.2.1.1.1.1.

(c) Three (3) HDMR tape units shall be provided equivalent in performance to the Ampex FR 2042 unit.

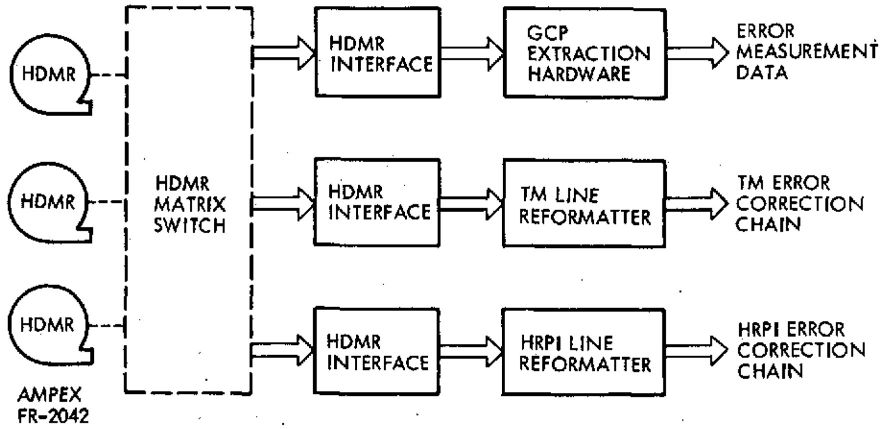

Figure 4. HDMR Subsystem

(d) Twenty (20) HDDT tape units shall be provided equivalent in performance to the Ampex FR 2028 unit.

3.2.1.1.1.2 Basic computational cluster. A basic cluster of equipment shall be provided to each of the processing subsystems as follows:

(a) CPU. A CPU with the following characteristics shall be provided:

Memory

Memory capacity: 16,384 to 262,144 32-bit words

Memory cycle time: 645 nanoseconds

Memory word size: 32 bits plus 4 parity bits

Maximum number of memory units: 8

Number of memory ports: 6

Basic processor

General registers: 64 in 4 blocks of 16

Data formats: Doubleword, word, halfword, and byte

Memory access modes: Unmapped (real addressing)
Mapped (virtual addressing)

Memory map page size: 512 words

Number of pages: 256

System control processor

Internal interrupts: 14

Maximum number of external interrupts: 48

Clocks: 4

Input/output (I/O) capabilities

Maximum number of I/O processors: 16

Number of channels per I/O processor: 16

Nominal maximum I/O processor bandwidth: 1 Mbyte/sec

Typical instruction execution times (microseconds)

Load immediate: 1.51

Load word: 1.72-1.94 (min-max)

Store word: 2.37-2.80

Branch: (Yes) 1.29
(No) 1.72

Logical AND: 1.72-1.94

Add immediate: 1.51

Add word (fixed): 1.72-1.94

Multiply word (fixed): 6.23

Add word (floating): 6.13 (typical)

Multiply word (floating): 9.14 (typical)

The CPU should be capable of operating with a basic Gibson Scientific Mix at over 380 K instructions/sec. As an integral part of the CPU, an I/O processor and 16 K of core dual port memory is provided. In addition to this, the basic unit shall also include 12 external interrupts, another 16 K of dual port memory (total = 32 K), and a second I/O processor.

(b) System control console. The system control console provides operator and maintenance control and communication with the CPU. The device shall be a standard ASC II, 10 cps teleprinter. The control console gives a hard copy of all dialog between operator and computer.

(c) Card reader. A 200-cpm card reader shall be provided. It shall be a photoelectric device that can read cards in binary or automatic mode. This device shall be used to input source programs, data cards, and operator job control information.

(d) Magnetic tape controller and two drives. A magnetic tape controller for two 9-track 125 inches/sec shall be provided. This unit will control the tape drive operation, provide single-track error detection and correction, and permit reading in both forward and reverse directions The two magnetic tape drives shall operate at 125 inches/sec with a density of 800 bits/in NRZI or 1600 bits/in phase encoded. Instantaneous transfer rate is 60,000 bytes/sec at 800 bits/in and 120,000 bytes/sec at 1600 bits/in. The drive system shall be a single capstan. Rewind time for a 2400-foot reel shall be 60 seconds or less.

(e) Line printer. A 300 line/min, fully buffered printer capable of printing 132 columns with a 64 ASC II character set.

(f) General-purpose controller interface. An interface to connect the general purpose controller to the computer shall be provided. This device shall allow communication as well as provide certain microcoder controlled functions.

(g) General-purpose controller. A general-purpose controller with the following features shall be provided:

- Three-bus architecture; 16-bit data path on each
- Eight general-purpose registers
- ALU
- Microprogrammable
- Parity checking
- Up to 128 I/O interfaces
- Bit/byte or 16-bit word manipulation
- Push/pull stack
- Eight interrupts.

Three of these general-purpose controllers shall be provided with a basic computational cluster.

3.2.1.1.2 Error correction subsystem. A hardware block diagram of the error correction subsystem is shown in Figure 5. The hardware that is peculiar to this subsystem is described in the following paragraphs.

3.2.1.1.2.1 Fixed-head disk storage. A fixed-head disk shall be provided on the error correction subsystem with the following characteristics:

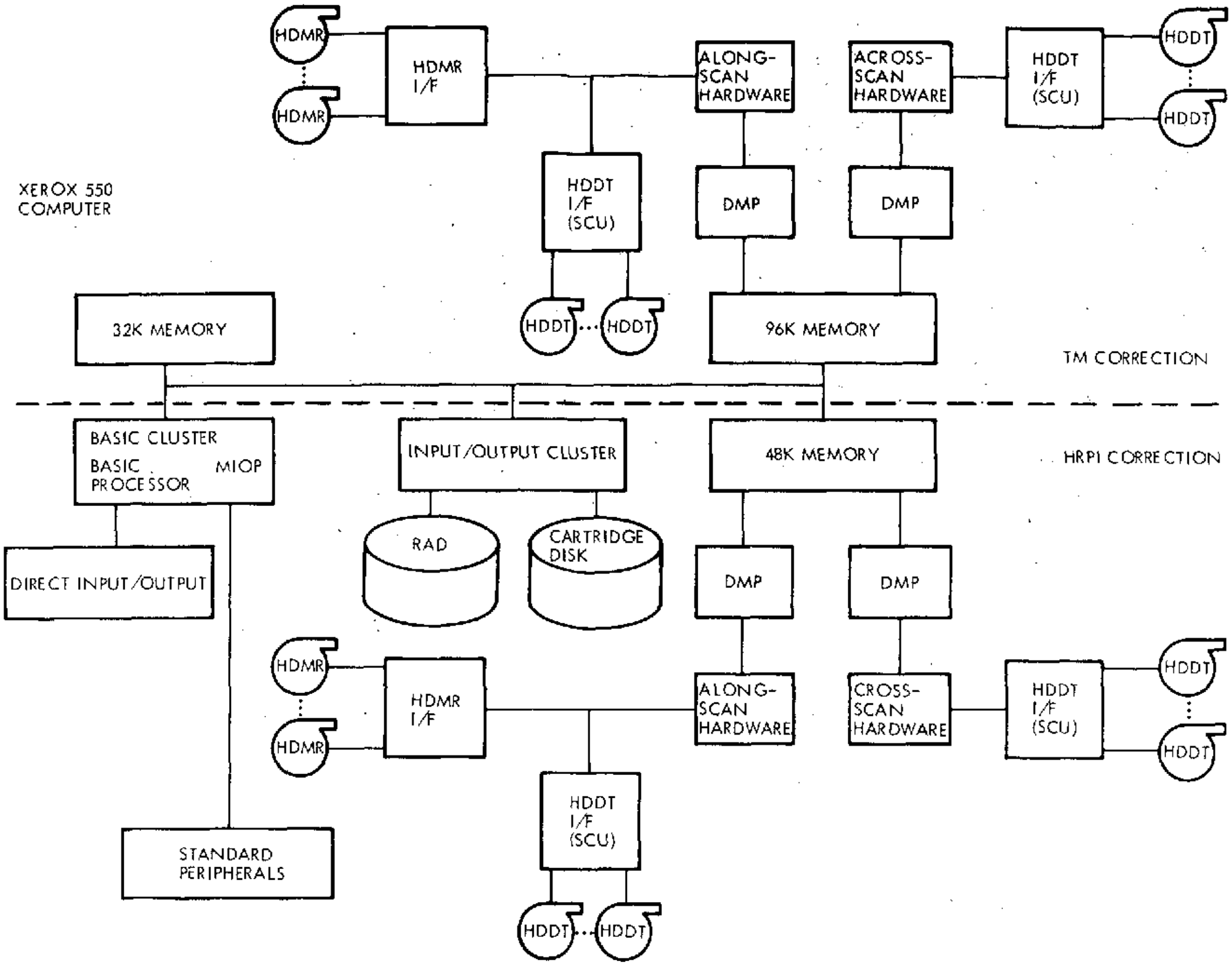

Figure 5. Error Correction Subsystem Hardware Complement

Operating characteristics

Storage capacity/unit: 2,883,584 bytes

Nominal access time

Average: 8.5 msec

Maximum: 17 msec

Nominal transfer rate

Single sector: 75,200 bytes/sec

Multiple sectors: 662, 500 bytes/sec

Recording format: 1024 bytes/sector, 11 sectors/track, 256 tracks/unit

3.2.1.1.2.2 Additional memory modules. A skew buffer bulk memory shall be provided to store sufficient along scan corrected data to allow the across scan skew corrections to be performed as an in-line process. The sizing of the buffer differs for the TM and HRPI. The configured memory for the baseline HRPI is 48 K 32-bit words and 96 K words for the baseline TM.

Each buffer memory is configured with a pair of dedicated ports for use solely by the image correction string attached to it. The dedicated ports are bussed together to form an input bus and output bus each attached to a direct memory processor. This separation of the input and output paths provides the minimum memory access interference for a continuous flow process such as image correction.

For this purpose, five additional 32 K modules of core memory shall be provided on the error correction subsystem and associated 2-port expansion units. This will be compatible with the basic computational subsystem requirements.

3.2.1.1.2.3 Additional input/output processors. An additional 16 input/output (I/O) channels will be required for the error correction subsystem. Each channel should be capable of transmitting a 32-bit data path. The I/O processor shall have requirements consistent with those found under the basic computational cluster description.

3.2.1.1.2.4 Additional cartridge disk storage and controllers. Two cartridge disk drives and the related controller shall be provided on the error correction subsystem. Each drive shall conform to the following specifications:

Recording format: 1024 bytes/sector, 7 sectors/track, 400 tracks/surface (+8 alternates)

Drive capacity (sectored): 5.7 million bytes (removable)

Nominal access time

Seek: 38 msec average (12 to 75 msec)

Rotational latency: 12.5 msec average (0 to 25 msec)

Transfer rates

Instantaneous (per sector): 312 K bytes/sec

Average (multiple sectors): 286 K bytes/sec

3.2.1.1.2.5 Additional general-purpose controller. One additional general purpose controller shall be provided. The characteristics shall be identical to those set forth in Section 3.2.1.1.2.1.

3.2.1.1.2.6 Direct memory access units. Four DMA units shall be provided for direct access to memory. These units shall comply with the requirements set forth in the basic computational cluster description.

3.2.1.1.2.7 HDDT interface. Four interfaces shall be provided between the HDDT matrix switch and the general purpose interface units of paragraphs 3.2.1.1.1.1 and 3.2.1.1.2.5.

3.2.1.1.2.8 Along-scan resampler. Two special purpose hardware units shall be provided to perform the first step of the sensor image correction processing. This processing includes receiving the raw sensor data from the line reformatter data bus, radiometrically calibrating each pixel, applying the along scan correction and placing the corrected pixel into the skew buffer for use by the across scan corrector (CSC). They shall interface to core memory via the direct memory access units specified in paragraph 3.2.1.1.2.6.

3.2.1.1.2.9 Across-scan resampler. Two special-purpose hardware units shall be provided to perform the final step of the image correction process. The CSC reads pixel data from four adjacent lines at a time from the skew buffer and generates one continuous output line. The processed output lines are passed directly to the output HDDT system via a parallel data bus.

The structure of the across scan resampler is similar to that of the ASC consisting of an input and output data interface, corrector module but no radiometric calibration module.

They shall interface to the core memories via the direct memory access units specified in paragraph 3.2.1.1.2.6.

3.2.1.1.2.10 HDMR interfaces. Two special-purpose hardware interfaces shall be provided between the HDMR matrix switch and the along-scan resampler and two of the general-purpose-interface units of paragraphs 3.2.1.1.1.2 and 3, 2.1.1.2.5.

3.2.1.1.3 Estimating, formatting, and film production subsystem. This subsystem shall consist of the common equipment and also the following unique items (Figure 6).

3.2.1.1.3.1 Additional memory modules. Three additional 32 K dual port memory modules shall be provided. These shall meet the same requirements as those in paragraph 3.2.1.1.2.2.

3.2.1.1.3.2 Additional I/O processors. Seven additional I/O processors shall be provided that meet the same requirements as those in paragraph 3.2.1.1.2.3.

3.2.1.1.3.3 Cartridge disk storage. Two cartridge disk controllers and four drive units shall be provided that meet the same requirements as paragraph 3.2.1.1.2.4.

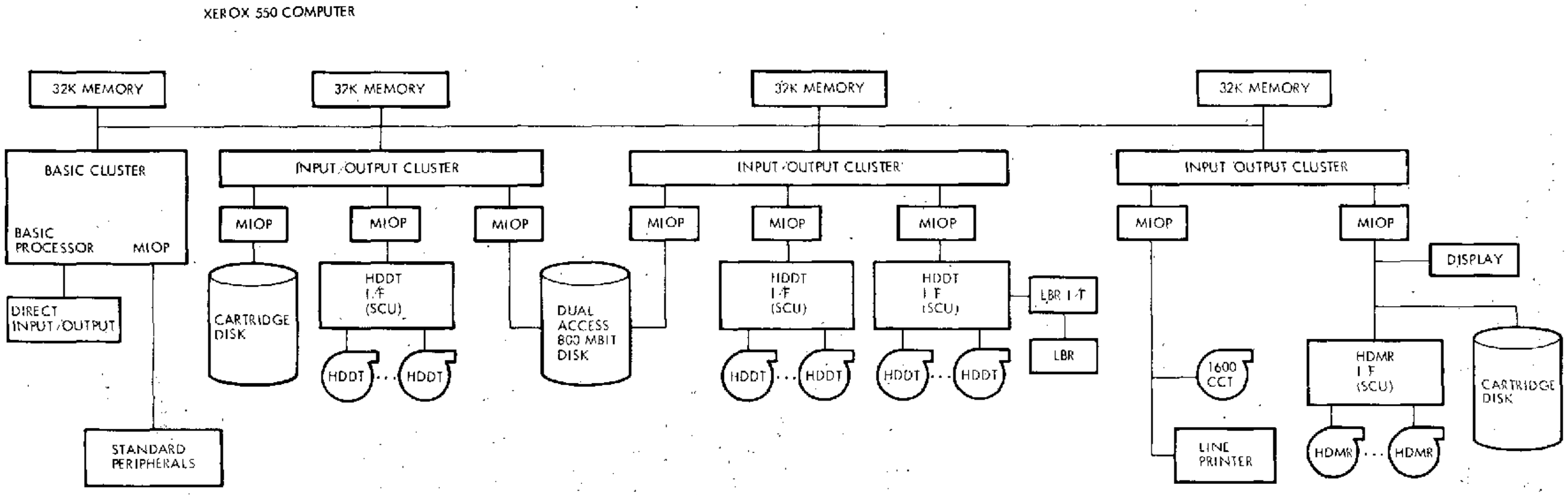

Figure 6. Estimation, Formatting, and Film Generation Subsystem Hardware Complement

3.2.1.1.3.4 Large capacity disk storage. Eight 100-Mbyte large capacity disk storage units shall be provided as an integral portion of the estimation, formatting, and film production subsystem and they shall have the following characteristics as a minimum:

Recording format	1024 bytes/sector 404 tracks/surface (plus seven alternates) 19 recording surfaces
Spindle capacity (unsectored)	100,000,000 bytes
Nominal access time	
Seek	30 msec average (10 to 55 msec)
Rotational latency	8.3 msec average (0 to 16.7 msec)
Transfer rate	
Instantaneous (per sector)	806,000 bytes/sec

3.2.1.1.3.5 Interactive CRT display. An interactive image display CRT is included for the purpose of control point library maintenance; that is, addition, deletion, and modification of control points in the control point library. The CRT contains an internal refresh memory to minimize CPU interaction requirements. The Comtal 8000 series is typical of the required display, containing sufficient storage for several 512 x 512 images of up to 256 gray level intensities. Hardware contrast enhancement is available (as well as color density slicing). A trackball or joystick, together with hardware-generated cursor, allows manual designation of image feature locations to the CPU. An entire 512 x 512 image can be replaced by the CPU in less than 2 seconds. A state-of-the-art fixed-head disk (data disk) is used as refresh store.

3.2.1.1.3.6 HDDT interface. Three HDDT interfaces as described in paragraph 3.3.1.1.2.7 shall be provided.

3.2.1.1.3.7 GCP/CALIB HDMR interface. A GCP/CALIB HDMR interface shall be an "intelligent" interface which accepts input data at a 20 Mbit rate and extracts TM and HRPI data points. The interface shall provide input data reformatting to line sequential, pixel ordered, band-interleaved-by-line data. The special-purpose hardware buffer shall be 1.6 Mbytes.

3.2.1.1.3.8 Filmwriter. The laser beam recorder shall accept 8-bit parallel data words and synchronizing signals from the interface units. Requirements and characteristics of this unit are as follows:

(a) Recording capability commensurate with continuous film transport; up to 70 images.

(b) Bandwidth must support a rate of 5.0 Mpixels/sec with up to 8 bits of grey level

(c) Output data quality resolution 50 percent MTF at 40 lp/mm, with f/32 optical size.

Relative radiometric accuracy	5%
Resolution element position accuracy (long term)	0.1%
Modulation transfer function	
at 8,192 pixels	50%
Spot wobble Transport jitter Scanning linearity Film distortion	Commensurate with sensor quality (0.1 pixels)
Dynamic range (grey level)	6 bit

(d) Laser

Spectral region	TBD
Aperture function	TBD

(e) Film

Size	241 mm
ASA rating	TBD
Processing (laboratory versus direct develop)	TBD
Transport types	Continuous

An intercept-driven line buffer would be an integral part of the LBR, with a capability of storing a number of lines of digital data. The interface between the LBR and the basic computational cluster shall also be included.

3.2.1.1.4 <u>Formatting and tape generation subsystem.</u> This subsystem is shown in Figure 7.

3.2.1.1.4.1 <u>Fixed-head disk storage.</u> A fixed-head disk storage unit equivalent to the one described in paragraph 3.2.1.1.2.1 shall be provided.

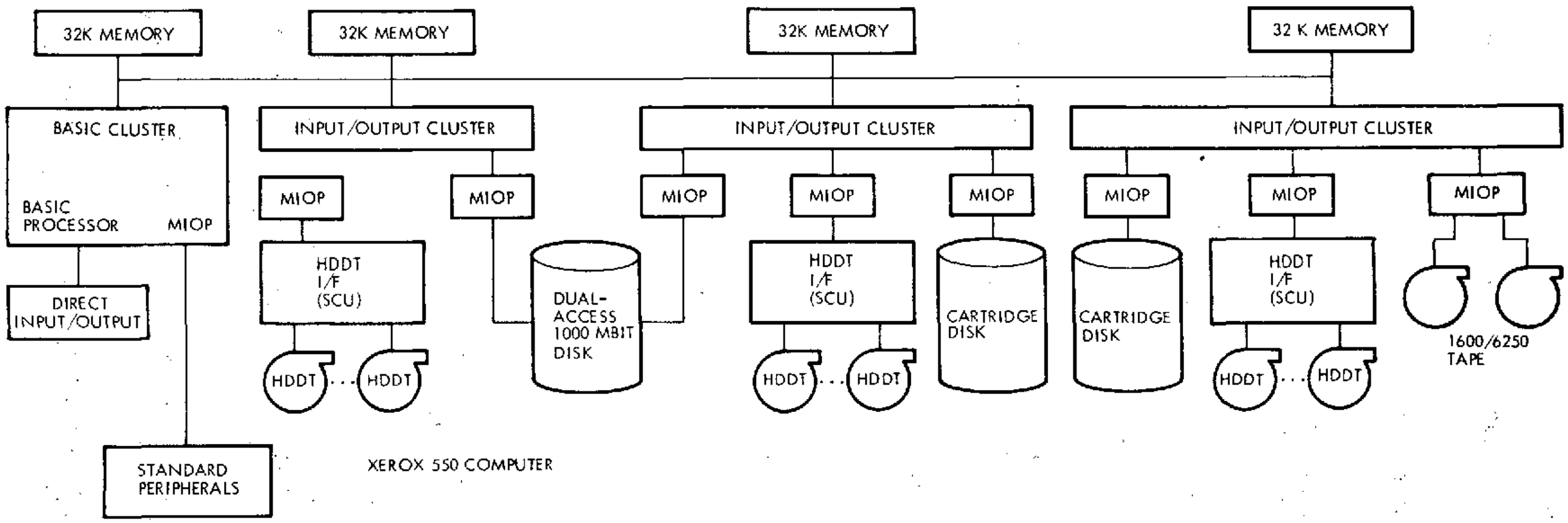

Figure 7. Formatting and Tape Generation Subsystem Hardware Complement

3.2.1.1.4.2 Additional memory modules. Three additional 32 K, two-port memory modules shall be provided that meet the requirements set forth in paragraph 3.2.1.1.2.2.

3.2.1.1.4.3 Input/out processor. Seven additional I/O processors as described in paragraph 3.2.1.1.2.3 shall be provided.

3.2.1.1.4.4 Cartridge disk storage. Two cartridge disk drives, the same as those specified in paragraph 3.2.1.1.2.4 shall be provided.

3.2.1.1.4.5 Large capacity cartridge disk drive. A large capacity cartridge disk system with a total capacity of 800 Mbytes shall be provided. This will be accomplished by providing eight drives and controllers as described in paragraph 3.2.1.1.3.4.

3.2.1.1.4.6 1600/6250 tape recorder. One combined 1600/6250 bits/in. unit: 2 drives; 400 Kbytes/sec at 250 ips for 1600 bpi; 780 Kbytes/sec at 125 ips for 6250 bpi; 45 sec rewind time.

3.2.1.1.4.7 HDDT interfaces. Three HDDT interfaces as described in paragraph 3.2.1.1.2.7 shall be provided.

3.2.1.2 Software subsystem. The following descriptions of the software requirements of the CDPF are not complete. Presented will be typical modules that will be required. An assumption here is made that the software existing on the NDPF will be compatible with the computers being provided for CDPF. The NDPF software is all written for the Xerox Sigma 5 series.

3.2.1.2.1 Real-time operating system. A multitasking real-time operating system similar to the Xerox CP-R shall be required. The following is a description of the types of functions this operating system must perform.

(a) Multiprogramming. Support shall be provided for up to 32 jobs (one background).

(b) Multitasking. Support shall be provided for up to 255 tasks.

(c) Dynamic real memory management. Services to manage pools of dynamic real memory among primary and secondary real-time tasks shall be provided.

(d) Task scheduling. Primary tasks shall be scheduled through the external interrupt structure. Secondary tasks shall be priority scheduled and dispatched at multiple dispatcher levels upon the completion of I/O, logical events, wakeup, and timer intervals, or upon demand by other foreground tasks.

(e) Intertask communication. Services shall be provided for foreground tasks to communicate directly through service calls, through shared files, and through shared segments between tasks.

(f) Complete memory protection. All secondary (including background) tasks shall be isolated virtually through memory access protection. Primary tasks shall be protected by memory write locks.

(g) Roll-in/roll-out. All secondary tasks, unless they request to be "locked" in memory, may be rolled out to make room for higher priority tasks.

(h) Mapped library management. Library routines may be shared between primary tasks (unmapped) or between secondary tasks (mapped). For mapped library management, all user (task) dependent data areas shall be mapped into the virtual memory assigned to each task so that reentrancy overhead is almost eliminated and data protection between tasks is ensured.

(i) File management. Dynamic cataloging of files (allot and delete) and also for file sharing through enqueue/dequeue services shall be provided. All files shall be addressed by name and can be read or written in several modes: compressed, sequential, and random.

(j) Device input/output. I/O may be either device-dependent or device-independent. The user may refer to I/O devices by symbolic designators or by I/O device address in order to perform I/O. All I/O services shall be FORTRAN callable.

(k) Input/output queueing. All I/O requests shall be queued on the basis of task priority as opposed to first-in/first-out (FIFO).

(l) Debug capability. The debug program can be used to debug either primary or secondary tasks and to set breakpoints, snapshots and dumps. These features are available from local and remote terminals. Multiple programs may be debugged concurrently from separate terminals.

(m) Terminal job entry. Remote terminals may be concurrently connected through a communication subsystem for purposes of job entry, program development, editing, and debugging.

(n) Program initiation. Programs shall be either resident or nonresident (called into execution from the disk). They shall be initiated by the operator, by request from the batch job stack, by a terminal, or by other real-time programs. They may be restarted from a suspended (wait) state by other tasks, by external events, by I/O completion or by a real-time clock.

(o) Reentrant routines. Reentrant service routines and sharable in-core public libraries eliminate the need for duplicate copies of frequently utilized routines.

(p) FORTRAN processor. A FORTRAN IV compiler shall be provided that conforms with ANS FORTRAN X 3.9.

3.2.1.2.2 Software requirements

3.2.1.2.2.1 NDPF existing software modifications. The following are descriptions that exist either in total or in part in the NDPF system. These are written in Xerox code and if compatibility is available, these can be modified according to the following summaries. (This is not a complete list; some modules are still TBD.)

NAME: Error Measurement Data Module

DESCRIPTION: Extract data from BFET, spacecraft performance data tape (SPDT), and metrology tapes to create error correction data (ECD) file by merging of data; HRPI calibration data may be on SPDT.

LIMITATIONS: This function might reside on CC off-line computer

PREVIOUS USAGE: Similar in function and scope to NDPF master digital data module located on Sigma 5.

FUNCTIONS/ROUTINES CALLED: Utility module, system module, file maintenance module

INPUT: BFET, spacecraft performance data, MET data

OUTPUT: Error correction data file onto disk pack (2 MB)

ALGORITHM STRUCTURE: Straight line

MEMORY SIZE (32 bit words): 32 K

Execute time: 30 minutes

Execute rate: 1 per day

DESIRED LANGUAGE CHARACTERISTICS:

- List processing
- Logical statements
- Data file management
- Algebraic operations (boolean, exponent, etc.)
- Machine language I/O

NAME: Error Estimation Module

DESCRIPTION: Process ECD file together with ground truth data to obtain distortion coefficients for error correction process; annotation data generated; calibration data formatted; ground truth file maintenance; updates catalog data base.

LIMITATIONS:

PREVIOUS USAGE: Annotation function exists at NDPF, a resident function on TRW's ERTS data processing system.

FUNCTIONS/ROUTINES CALLED: System module, utility, and file maintenance modules

INPUT: ECD file; ground truth data from reformatter

OUTPUT: ECD file (5 MB)

ALGORITHM STRUCTURE: Straight line

MEMORY SIZE (32 bit words): 70 K

Execute time: 91 seconds

Execute rate: 1/scene

DESIRED LANGUAGE CHARACTERISTICS:

- List processing
- Logical statements
- Data file management
- Common memory
- Vector/matrix operations
- Algebraic operations (boolean, exponent, etc.)
- Floating point operations
- Machine language I/O
- Fixed-float conversions
- Byte addressing

NAME: Photo Q/A Module

DESCRIPTION: Provides interactive support to photographic product facility for computing film speeds, gammas, etc. Provides data quality information for management report.

LIMITATIONS: None

PREVIOUS USAGE: Existing routine at NDPF.

FUNCTIONS/ROUTINES CALLED: System module, file maintenance, input module

INPUT: Photo parameters

OUTPUT: Photo parameters, update to Q/C data base

ALGORITHM STRUCTURE: Straight line

MEMORY SIZE (32 bit words): 20 K

Execute time: 2 minutes

Execute rate: 1/film image (estimated maximum)

DESIRED LANAGUAGE CHARACTERISTICS:

- List processing
- Logical statements
- Algebraic operations (boolean, exponent, etc.)
- Floating point operations
- Machine language I/O
- Fixed-float conversions

NAME: Production Control Module

DESCRIPTION: Produces estimated daily master data production report; work orders for master data generation, image generation, and digital product generation; and shipping documents and lists.

LIMITATIONS: None

PREVIOUS USAGE: Similar function exists at NDPF.

FUNCTIONS/ROUTINES CALLED: System module, utility and file maintenance

INPUT: Schedule job orders

OUTPUT: Work orders and shipping documents

ALGORITHM STRUCTURE: Straight line

MEMORY SIZE (32 bit words): 131 K

Execute time: 10 seconds

Execute rate: 6/day (estimated maximum)

DESIRED LANGUAGE CHARACTERISTICS:

- List processing
- Logical statements
- Data file management
- Common memory
- Message skeletons
- Bit manipulation
- Machine language I/O

NAME: Query Module

DESCRIPTION: Provides available imagery reports to browse facility user/management.

LIMITATIONS:

PREVIOUS USAGE: Similar function at NDPF.

FUNCTIONS/ROUTINES CALLED: System module, file maintenance, input module

INPUT: User query

OUTPUT: Inquiry response — see Table 4.

ALGORITHM STRUCTURE: Straight line

MEMORY SIZE (32 bit words): 128 K

Execute time: 3 seconds

Execute rate: 200/day

DESIRED LANGUAGE CHARACTERISTICS:

- List processing
- Logical statements
- Data file management
- Interrupt masking and handling
- Message skeletons
- Machine language I/O

Table 4. Query Response

Observation ID	Orbit No.	Station ID	Ephemeris Type	Data Quality	Scene Center Geographic Area		Time	Cloud Cover	Products	
					Lat	Lon			HRPI	TM
1000-00001	0001	G	R	G	121.55	36.21	1600	10% Cloud Cover	1234 FFFX	1234567 CCHCCCC
Mission 1 = A 2 = B 3 = C		Receiving Site G = Goldstone F = Fairbanks N = NTTF, Goddard	R = Refined P = Predicted	G = Good F = Fair P = Poor					H = HDDT available F = Film products available C = CCT available X = All products available	

NAME: Retrieval Module

DESCRIPTION: Identifies available imagery by browse facility user's descriptor input.

LIMITATIONS: None

PREVIOUS USAGE: Similar function exists at NDPF

FUNCTIONS/FOUTINES CALLED: System module, file maintenance, input module

INPUT: User query

OUTPUT: Inquiry response — see Table 5

ALGORITHM STRUCTURE: Straight line

MEMORY SIZE (32 bit words): 128 K

Execute time: 5 seconds

Execute rate: 200/day

DESIRED LANGUAGE CHARACTERISTICS:

- List processing
- Logical statements
- Data file management
- Interrupt masking and handling
- Message skeletons
- Machine language I/O

Table 5. Retrieval Processing – Display Image Output

OBSERVATION ID 1018-05453

Orbit number:	0196	Subsat point (longitude):	-80.83
Total cloud cover:	10	Subsat point (latitude):	41.83
Station	Goldstone	Picture center (longitude):	-80.84
Ephemeris type:	R	Picture center (latitude)	S41.79
Altitude:	492.32	Sun elevation:	33.70
Heading:	36.38	Sun azimuth:	40.62
Track:	48.32		
Microfilm roll:	4179		
Position in roll:	0591		

SENSOR QUALITY

HRPI				TM						
1	2	3	4	1	2	3	4	5	6	7
G	G	G	G	G	G	G	G	G	P	P

Image descriptors:	Forestry rivers
Number of products present:	13
Product:	P-CL
Date produced:	09/15/81
Product:	CCT
Date produced:	09/15/81
Band indicator:	11111111010
Last request date:	09/15/81
Band indicator:	01100000011
Last request date:	09/15/81

NAME: Utility and File Maintenance

DESCRIPTION: Utility software creates and saves disc data files; file maintenance software access, and updates disc data files

LIMITATIONS: None

PREVIOUS USAGE: Typical system routines – functions currently present at NDPF.

FUNCTIONS/ROUTINES CALLED: System module

INPUT:

OUTPUT:

ALGORITHM STRUCTURE: Straight line

MEMORY SIZE (32 bit words): 100 K

Execute time: 1/2 second

Execute rate: 1 K/day

DESIRED LANGUAGE CHARACTERISTICS:

- Data file management
- Interrupt masking and handling
- Common memory

NAME: Catalog Module

DESCRIPTION: Produces catalog masters; produces microfilm work orders; produces catalog montage master listing.

LIMITATIONS: Needs to be modified to reflect EOS mission payload

PREVIOUS USAGE: Currently exists at NDPF.

FUNCTIONS/ROUTINES CALLED: System module, utility and file maintenance module, input module

INPUT:

OUTPUT: See Table 6 showing sample catalog page

ALGORITHM STRUCTURE: Straight line

MEMORY SIZE (32 bit words): 90 K

Execute time: 2000 seconds (estimated maximum)

Execute rate: 1/day

DESIRED LANGUAGE CHARACTERISTICS:

- List processing
- Logical statements
- Data file management
- Message skeletons

Table 6. Sample Catalog Format

Observation ID	Microfilm Roll No. Position in Roll HRPI	TM	Date
1000-00001	00001/0001	00002/0001	08/01/81

Cloud Cover	Image Center Latitude	Longitude	Sun Elevation	Sun Azimuth	Data Quality HARI	TM
70	25.64N	99.35W	53.5	127.1	1234 GGGG	1234567 GGGGGGG

Available Product Types

Digital HDDT	CCT	Film B	C
X	X	X	X

NAME: Management Report

DESCRIPTION: Produces image generation report, historic shipping list, historic request statistics, photographic inventory, data quality statistics, provides user profiles, produces status-by-user reports, computes customer reaction statistics, reworks analysis module and LCGS pass request list.

LIMITATIONS: None

PREVIOUS USAGE: Similar function exists at NDPF.

FUNCTIONS/ROUTINES CALLED: System module, utility and file maintenance, input module

INPUT: Report code number to retrieve desired data

OUTPUT: Desired reports (off-line)

ALGORITHM STRUCTURE: Straight line

MEMORY SIZE (32 bit words): 180 K

Execute time: 900 (estimated maximum)

Execute rate: 1/day

DESIRED LANGUAGE CHARACTERISTICS:

- List processing
- Logical statements
- Data file management
- Common memory
- Message skeletons
- Machine language I/O

NAME: Input Module

DESCRIPTION: General man/machine interface: inputs user address entries, image assessment data, image descriptors, data received entries, data request entries, pass request entries, roll number entries, satellite coverage entries, data shipped entries, standing order entries; terminate work orders entries; remote batch call; and delete entries.

LIMITATIONS: None

PREVIOUS USAGE: Similar function exists at NDPF

FUNCTIONS/ROUTINES CALLED: Order staging, order scheduling, etc.; all major modules defined herein.

INPUT: None

OUTPUT: None

ALGORITHM STRUCTURE: Multiple branches

MEMORY SIZE (32 bit words): 304 K

Execute time : 2 seconds (estimated maximum)

Execute rate: 250/day

DESIRED LANGUAGE CHARACTERISTICS:

- List processing
- Logical statements
- Data file management
- Interrupt masking and handling
- Common memory
- Message skeletons
- Machine language I/O
- Explicit control of registers and indicators
- Direct control of indexing facilities
- Indirect addressing
- Byte addressing

NAME: EOS Monitor

DESCRIPTION: The collection of system functions which control, monitor, and interface all other modules — manages the computer resource. Possible Xerox operating system is CP-V.

LIMITATIONS: CP-V utilizes only CAL-1; certain transferable ERTS BTM routines will require modification

PREVIOUS USAGE: New software

FUNCTIONS/ROUTINES CALLED: None

INPUT: None

OUTPUT: None

ALGORITHM STRUCTURE: Table driven

MEMORY SIZE (32 bit words): 32 K

Execute time: 1/10 second (estimated maximum)

Execute rate: 20 K/day (estimated)

DESIRED LANGUAGE CHARACTERISTICS:

- List processing
- Logical statements
- Real-time control
- Data file management
- Interrupt masking and handling
- Common memory
- Message skeletons
- Algebraic operations (boolean, exponent, etc.)
- Bit manipulation
- Machine language I/O
- Fixed-float conversions
- Explicit control of registers and indicators
- Direct control of indexing facilities
- Indirect addressing
- Double precision
- Byte addressing
- Scheduling, queue management, memory management, etc.

3.2.1.2.2.2 New software requirements. TBD.

3.2.2 Physical characteristics

3.2.2.1 Mechanical. TBD

3.2.2.2 Electrical

3.2.2.2.1 Power. The CDPF shall be operated from electrical power derived from local commercial sources. The quality and quantity of this input power shall be as shown in Table 7.

High voltage transients due to lightning or other phenomena 2000 volts, duration 500 microseconds (maximum).

The transformer secondary connection from the commercial line shall be three-phase, four-wire, wye, with solidly grounded neutral.

3.2.3 Reliability. Compliance with reliability requirements shall be taken by prediction techniques in conformance with EOS-4.1. The allocated reliability for the CDPF baseline configuration operating under conditions specified herein for a TBD period is TBD.

3.2.4 Maintainability. Maintenance of the CDPF shall be minimal in terms of special tools, time of maintenance and expense per installed hour of equipment. The supplier shall prepare and deliver preventive maintenance and equipment calibration instructions that will permit the CDPF to operate without maintenance for (TBD) hours elapsed time.

Table 7. Power Input

Continuous:	TBD watts
Voltage:	208/120 Vrms
Phase:	3-phase (4 wire)
Steady-state voltage:	±5%
Steady-state frequency:	±3%
Voltage modulation:	1%
Frequency modulation:	1%
Harmonic content:	5% (maximum)
Crest factor:	1.414 ±10%
Phase unbalance:	5% (maximum)

3.2.5 Environmental conditions. The CDPF equipment shall be designed and constructed to withstand any combination of the following service conditions without mechanical or electrical damage or performance degradation below that specified in the detailed equipment specifications.

The environmental conditions shown in Table 8 shall apply for each of the following categories of equipment:

- Type I. On-site and protected – housed in equipment shelter
- Type II. Transportation – packaged

3.2.6 Transportability. Transportability requirements shall be considered in the design of the CDPF equipment such that it can be transported by all standard modes with a minimum of protection. Special packaging may be used, as required, to ensure that common carrier transportation does not impose design restrictions.

3.3 Design and construction. The contractor shall utilize to the maximum extent practicable, proven, existing equipment to meet the requirements and characteristics of Section 3.2 of this specification. Where existing equipment cannot meet performance, packaging or interface requirements, preference will be given to modification of existing equipment in order to permit collection of established performance history and accumulated part operating experience for all items comprising the CDPF. Use, application, and operating characteristics of the CDPF shall be documented, when so specified by contract. Additional material and technical information shall also be provided in support of any patent limitation, trade secret protection or compliance with any code or agency regulation of the U.S.A. governing operation.

3.3.1 Materials, processes, and parts. Materials, processes, and parts of the CDPF shall be of excellent quality and of highest grades obtainable commensurate with precision electronics equipment manufactured for scientific field use. Toxic, critically limited, and strategic materials shall not be used. Special processes not readily obtained by modern manufacturing and finishing techniques in commercial use shall also be avoided. Wherever possible, materials, processes, and parts will be of recent manufacture and proven capable of providing their required function or life by recent demonstration or test. The utilization of any material, process, or part in the equipment shall not relieve the supplier of the responsibility for complying with all acceptance requirements in Section 4 or provisions covering the quality and conformance of materials, processes, or parts furnished as spare articles. Materials or processes used as protective coatings or finishes for fungi and oxidation control shall be approved prior to application. Such materials to be applied by vendors shall be identified for evaluation by NASA.

3.3.2 Electromagnetic radiation. Electromagnetic interference requirements shall be as specified in this specification and as defined in the individual equipment specifications. The contractor shall ensure that

Table 8. Environmental Conditions

Environment	Type I Operating	Type I Non Operating	Type II Non Operating
Temperature	$+25^{\circ}C \pm 5^{\circ}C$	$+10^{\circ}C$ to $+55^{\circ}C$	$-10^{\circ}C$ to $55^{\circ}C$, $+15^{\circ}C$ solar radiation
Rainfall	N/A	N/A	100 mm/hr.
Relative Humidity	50%	90%	30-100%
Wind	N/A	N/A	
Salt Atmosphere	TBD	Same	TBD
Salt Spray	N/A	N/A	TBD
Blown Sand and Dust	N/A	N/A	TBD
Fungus	Non-nutrient materials or treated to inhibit growth in tropic region	Same	Same
Shock and Vibration	TBD	TBD	TBD

undesirable electromagnetic radiations from his equipment are so limited as to not induce malfunctions or degradation to the performance of his segment.

Provisions such as line filters, arc arrestors, fitted enclosures, and shielding shall be provided for all powered equipment, waveguides and cables to ensure that undesirable frequencies and spurious noise spikes are sufficiently attenuated through shielding and distance to be below the threshold sensitivity of the receiving equipment.

3.3.2.1 General EMC/EMI requirements. In addition to the normal environmental RF noise sources such as factory equipment machinery, and discrete RF signal sources, the various interference generators within the overall system itself must be considered as contributors to the EMI environment.

The broadband RF environment will consist of switching circuit, digital clock teleprinter, and other equipment generated interference of a broadband nature.

It is the intent of these requirements to describe design practices to guide all equipment suppliers which will minimize the levels of this RF noise environment conducted and radiated to other equipments.

Electrical and electronic equipment shall operate without degradation, not only independently, but also in conjunction with other such equipment which may be placed nearby. This requires that the operation of all such equipment shall not be adversely affected by interference voltages and fields reaching it from external sources, and also requires that such equipment shall not, in itself, be a source of interference that might adversely affect the operation of other equipments.

3.3.2.1.1 Facilities. The facility shall provide the basic ground plane for all equipment and subsystems installed. All equipment racks, consoles, and peripheral equipment shall be electrically bonded to the ground plane by direct metal-to-metal contact.

3.3.2.1.2 Racks

(a) All equipment racks or consoles are preferred to be RFI-shielded type provided with RF gasketing or spring finger contact strips at all doors and front panel mating surfaces.

(b) All rack mounted equipment front panels, blank panels, or control panels shall have clean (conductive) rack mating surfaces.

(c) All racks using AC power shall have power line filters (including the neutral) installed immediately behind the input power connector. The filters shall provide a minimum of 80 dB insertion-loss from 15 kHz through 10 GHz.

(d) The primary power safety wire shall be grounded to the rack immediately adjacent to the power input connector.

(e) AC power carried within a rack (usually twisted-pair wire) shall be physically separated from signal or DC power wiring.

(f) Connectors shall not employ any nonconductive protective coatings. Connector mounting areas on racks shall be free from paint or any other nonconducting material. Filler materials between connectors and racks shall not impair the conductivity between mating surfaces. Silver-loaded epoxy or similar materials should be used as a filler if a dissimilar metal problem is anticipated.

3.3.2.2 Grounding configuration. The CDPF shall employ a common electrical reference ground plane system. In order to develop an effective system ground plane, it is imperative that the system be built into well designed equipment racks that are electrically bonded together, and that all rack-mounted assemblies are electrically bonded to the rack structures by continuous metal-to-metal contact. The rack structures shall then be securely bonded to the system electrical ground plane.

The following paragraphs define the criteria that shall be used in the development of an electrical ground equipment system with a multi-point grounding configuration. Single-point grounding will be used only when an item of commercial equipment requires a single-point ground configuration.

3.3.2.2.1 Racks and consoles. The type of racks or consoles specified for electronic equipment installation provides the basis for achieving an effective ground reference plane, as well as providing shielding integrity for the equipment mounted within the rack. Each rack or console shall have as a minimum the following features:

(a) All equipment racks shall be electrically bonded together.

(b) All mounting hardware within the rack shall be plated with a conductive finish.

(c) Electrical bonding between each group of racks shall be accomplished with either a copper bus bar 0.030 to 0.050 inch thick and 2.5 inches wide, or an equivalent copper braid 2.0 inches wide connected between the lower rear corners of each adjacent rack. At least 1/4-20 bolts with 3/4 inch washers should be used to fasten bonding bus bars to each rack.

3.3.2.2.2 Equipment drawers and chassis

(a) The back surface of all front panels shall be free of paint or anodized finishes in areas which mate to the rack mounting rails and shall be protected by a suitable conductive finish. All commercially purchased equipment panels which mate with the rack mounting rails shall meet this criteria. If this requirement cannot be satisfied with equipment as purchased, the equipment panels shall be modified to allow the desired bonding to be achieved.

(b) All metal-to-metal surfaces on the drawer, chassis, and components shall form a good electrical bond between the mated surfaces.

(c) Chassis containing integrated circuit cards shall be provided with a chassis ground bus which shall be mounted directly to and electrically bonded to the chassis.

(d) All rack-mounted drawer assemblies with front panels whose back surfaces are free of paint or anodized finishes shall be considered adequately bonded for grounding purposes when fastened to the rack structure with the necessary mounting screws.

(e) All rack mounted drawers with slides or shelves shall have a minimum of one AWG No. 16 wire which grounds the drawer to the rack for personnel safety when intended drawer wiring will not satisfy this requirement.

For especially sensitive equipment; which may be susceptible to extraneous interference when pulled out of the rack for service, calibration, etc.; the use of a solid flexible copper alloy bonding strap is recommended in lieu of the No. 16 safety grounding wire.

3.3.2.2.3 Electrical circuit grounding

(a) Primary AC power neutrals shall be isolated from racks and chassis and treated exactly as the high side of the AC power line. RF filters shall be placed in each side of the line in a balanced configuration. Only one set of filters is normally required for each rack or console.

(b) Within units, signal and power returns shall be grounded to chassis by short direct wire or printed circuitry.

(c) Sharing or daisy-chaining of return lines or printed circuitry should be avoided. Where a single-ground bus is used on a PC board, it should be chassis grounded at each board mounting pad.

(d) The DC return or ground line shall be carried in each secondary power cable between power supplies and user units and shall be grounded to chassis immediately adjacent to the connector in each case.

(e) Filter capacitor returns shall be connected to the chassis by the shortest most direct means. Filter or decoupling capacitors shall not share common return lines with one another or with other circuits.

(f) Where one power supply supplies secondary power to several similar chassis, care must be taken to provide adequate filtering or decoupling in each unit served by the supply, in addition to avoiding common DC returns, to reduce interaction between units.

3.3.2.3 Bonding

3.3.2.3.1 Bonding definition. Electrical bonding shall be defined as the method by which all elements of the system, including equipment racks, equipment panels, electrical and RF connectors, chassis covers, and subassemblies, are electrically interconnected to establish a low-impedance reference plane. The bonding methods and techniques employed in making the electrical connections (using MIL-B-5087B as a guide) shall ensure a minimum DC resistance and RF impedance between the above items.

3.3.2.3.2 Bonding methods. Electrical bonding shall be accomplished by metal-to-metal contact over the entire faying surface areas which are held in mechanical contact. Specific provisions shall be made to preclude contamination of bonding surfaces with nonconductive oxides and finishes. The electrical bonding technique employed shall provide, as a design goal, a bonding impedance not to exceed 1.0 ohm from 0.2 MHz through 10 GHz and as a firm requirement, a DC resistance of less than 2.5 milliohms at all unit-to-rack interfaces and between unit connector shells to unit cases.

3.3.2.4 Shielding criteria. Interference control by means of magnetic or electrostatic shielding techniques shall be assessed in terms of benefits derived. Maximum utilization of the inherent shielding effectiveness provided by equipment racks, equipment enclosures, and physical location/isolation parameters, will be considered in the application of shielding design to control internally and externally generated electromagnetic environments and to afford suitable protection to susceptible system elements.

3.3.2.4.1 Equipment shielding. The enclosures of all equipment capable of generating EMI or having excessive sensitivity to external fields shall be designed to provide necessary attenuation of electromagnetic fields. Electrical bonding of connector shell hardware, enclosure panels, cover plates, base plates, and end panels will be provided in accordance with paragraph 3.3.2.3.2. The following items shall be observed in the design of unit packaging, structural design, and equipment mounting:

(a) Access openings will be kept to a minimum size compatible with the equipment performance requirements. Multiple small openings are preferred to large single openings of the same cross-sectional area. Access openings for adjustment screws or visual inspection will be provided with cover plates or screw-on caps for continuity of the shielding design when not in use.

(b) Specific provisions will be made to ensure constant and uniformly distributed contact pressure across the metal-to-metal bonded joints. A suitable electrically conductive interface filler material, such as an approved knitted-wire mesh gasketing, may be applied between the interface surfaces to fill the voids and thus ensure the necessary continuity across the entire bonded joint.

3.3.2.4.2 Cable shielding. Electromagnetic shielding shall be employed on all interface cables between units or racks which are capable of generating excessive electromagnetic fields or which may be susceptible to electromagnetic fields. The degree of shielding employed shall be sufficient to ensure an adequate compatibility margin. Wire shields shall not be used as an intentional current carrying conductor. All shields shall be covered by a layer of insulation.

(a) All RF signals not run in waveguide shall be carried in semi-rigid or double shielded coaxial cable.

(b) Within racks, all signal circuits and DC power shall be shielded. Shields shall be grounded to the rack structure at the input connector and the unit chassis. Between units within a rack, shields shall be grounded to the unit chassis of each unit.

(c) Shield grounds shall be made via the connector shell wherever practical. Daisy-chaining of shield grounds should be avoided. Shields should be grounded individually by short (preferably the shield braid itself) direct means to the connector back-shell. (RFI back-shells are available for most connector types)

(d) Between racks, unbalanced signal circuits shall use single conductor shielded cable properly grounded as described above to the rack at each interface with an overall shield over the cable bundle grounded to the connector shell at each interface.

(e) Control circuits, voltage or current step functions, teletype or line printer circuits should not be carried in cable bundles with analog or digital signals.

(f) Balanced signal circuits shall be run in twisted shielded pair cable with shields grounded at source and load end.

(g) Primary shields of individual or multiconductor wires within the cable shall be carried through the connector on one or more pins and connected to the housing or chassis immediately adjacent to the connector within the unit, using the shortest possible terminating lead length. In no case shall this terminating wire exceed 1 inch in length.

(h) Where a large number of separate shields are used within the cable, the individual shields may be tied to a halo-ring and two grounding leads from the halo-ring carried through the connector and terminated at the rack by wires from the receptacle. The daisy chain technique of terminating shields shall not be used.

(i) Shields shall not be used intentionally for signal or power returns except in the case of coaxial cables.

(j) Internal harnesses should be designed, grouped, and routed to minimize interference effects due to common circuit impedances, cross-talk, radiation, and pickup.

3.3.2.4.3 Interface cable installation. In addition to cable shielding, proper segregation, transposition, and routing requirements for cables and cable bundles will be considered in the design of all subsystem and equipment interface cabling.

(a) Physical isolation. Interface cabling of different signal categories will be physically isolated from each other by employing separate cable bundles. Typical cable bundling categories recommended are as follows: primary power, secondary power, RF input, RF output, digital, analog, and command (discrete pulse).

(b) Installation. All interface cabling will be routed in proximity to, and preferably in direct contact with, the electrical reference plane (structures).

3.3.2.4.4 Control and display signals

(a) Control circuits are defined as switch circuits which operate indicator lights, relays, or cause other direct application or interruption of power to electrical or electromechanical devices.

(b) All control line returns shall be maintained separate from all data signal returns. Individual control line returns shall be AWG No. 20 wire (minimum) and shall be grounded directly to chassis, or chassis ground bus when applicable. If lamps, indicators, or relays are grouped together such as on a front panel, they may be bussed together and connected to chassis ground with an AWG No. 18 stranded wire (minimum). Control lines between drawers or racks should be run in single-shielded or multiple-conductor shielded cables. Shields shall be grounded to chassis or rack at each connector interface.

(c) Remote control circuit lines emanating from a drawer or console should have their returns grounded only at the power supply. Multiple control functions may employ a common return, but shall not be returned through the shield or a conductor common to any signal function. Control lines within drawers and between racks should be run in twisted pair shielded or multiple conductor shielded cables. Shields shall be grounded to chassis or rack at each connector interface.

3.3.2.5 Circuit decoupling. Design provisions will be made to ensure that circuit isolation decoupling devices are properly selected and installed to achieve maximum effectiveness through their use. The provisions for the application of such devices will be required in the following areas:

3.3.2.5.1 Power decoupling. Power conductors will be decoupled directly at the entering or exiting interface of an equipment. RF decoupling devices of the feedthrough variety will be employed, wherever practical, in the circuit design of equipment.

(a) Each equipment rack shall incorporate a balanced L-C type filter, designed to provide maximum isolation from 15 kHz to 10 GHz. (Also see paragraph 3.2.2.2.)

(b) In conjunction with the basic input filter design of each rack, each primary input and return line of sensitive equipment such as receivers shall employ and RF decoupling device of the feed-through variety to obtain RF isolation as required for that unit.

(c) The feedthrough filtering devices will be of the flange or screw-neck mounted types and will be mounted through the equipment enclosure wall or an RF extension thereof, with total circumferential filter-case contact to the electrical reference plane.

3.3.2.5.2 Analog circuit decoupling. Analog circuits will be decoupled, wherever required, by means of capacitive filtering components designed to eliminate power switching and rectification spikes and digital signal coupling.

3.3.2.5.3 Transient suppression. Transient or arc suppression components shall be employed in the design of all switching, pulse, relay, and solenoid circuit applications. Transient suppression techniques will be applied to limit the amplitude and duration of single-event switching transients as well as to control the rise and fall times of recurring pulse signals.

(a) Switching contacts. Integrated R-C type suppression components will be applied directly across all relay or switching contacts designed to provide closure or switching of inductive type loads at the primary bus voltage. Typical applications of such suppression components for the equipment will include: control relays, high-level command relays, and RF switching solenoids.

(b) Relays. Diode suppression components shall be applied directly across all relay coils.

(c) Solenoid operated devices. Transient suppression techniques will be applied in the design of all solenoid control circuitry and associated interface circuitry. In remote solenoid applications, where diode suppression components cannot be applied directly across the solenoid coil or transient source, alternate provisions of transient suppression and control of generated interference will be provided.

3.3.3 Nameplates and product marking. Each subsystem shall be identified in accordance with MIL-STD-130. Where practical, a minimum character height of 0.9 inch shall be used. Marking shall include the manufacturer's name or initials, assembly part, number and revision status, assembly serial number, contract number, and others as delineated in the component equipment specifications.

3.3.4 Workmanship. Equipment covered by this specification shall be engineered, assembled, and tested in a manner that appearance, fit, and adherence to specific tolerances shall be observed. Particular attention shall be given to the neatness of parts and assemblies, cleaning, finishing, plating, painting, drilling, machine screw assemblage, welding, brazing, and freedom of parts and connections from debris, burrs, and sharp edges.

3.3.5 Interchangeability and part substitution. All equipment supplied under provisions of this contract shall be examined and provided with features of interchangeability to accommodate direct replacement of identical functioning assemblies and parts during the life-cycle of the equipment. Additionally, parts having suitable operating characteristics which may permit substitution of an original part shall be so identified for purposes of repair and spare-parts provisioning. Substitute parts shall be identified by manufacture, part, or drawing number and other identifying characteristics to permit an original part to be totally substituted by an alternate from commercially available sources.

3.3.6 Safety and hazards control. The supplier of the equipment, covered by specification, shall provide for both personnel safety and equipment safety. Each of these safety categories will encompass safe, marginal, critical, and catastrophic conditions coincident with the development, manufacture, test, installation, handling, training, and operation of the CDPF. The supplier shall also be required to disclose that his facility complies with appropriate federal, state, and local safety codes and regulations governing the identification, analysis, and control of hazards associated with the above mentioned conditions. In addition, the supplier shall conduct a program of safety and hazards control that identifies dangerous events and their means of control while the supplier's equipment is operating. Demonstration of special safety and hazard-control features of the CDPF shall be in compliance with provisions of Section 4.

3.3.7 Human performance/human engineering. MIL-STD-1472A shall be used for the design of man/machine interfaces.

3.4 Documentation. Documentation shall be as specified in EOS-3.3-8.

3.5 Logisitics. All equipment may be unpacked, installed, calibrated and operated by NASA, or personnel other than supplier personnel, at designated destinations. It is for this reason that the supplier shall ensure that all necessary parts, supplies, information, and other provisions for the equipment are accounted for and are complete.

3.5.1 Maintenance. The installed segment equipment shall be configured for ease of maintenance. Drawers or modules to be installed in racks shall be equipped with slides and cables of sufficient length to provide access to the unit without loss of power or signal input when the unit is extended from the rack for required servicing and maintenance actions. All rack-mounted or rack-supported hardware shall be capable of being maintained from the front of the rack. All drawers shall be provided with positive locking mechanisms. Where equipment maintenance, testing, or repair requires access to the bottom of the chassis, the slides shall be a tilt type with detent locking at 45 degrees and 90 degrees tilt. Modules shall be arranged so that access to any module will not require removing any other modules or parts except access panels.

3.5.2 Spares. The contractor shall provide NASA with his recommendations of quantities of spare equipment necessary to support operations for an 18-month period.

3.5.3 Facilities and facility equipment. The vendor shall provide installation planning and design engineering instructions for installation and facility equipment designs for installing and operating the equipment covered by this specification. Such data shall include, but not be limited to, reinforced concrete foundations and related structure(s), grounding and power components, and requirements and facilities interfaces. Vendors designs and data shall comply with all environmental and other constraints identified herein.

4. QUALITY ASSURANCE PROVISIONS

4.1 General. The quality assurance program controls shall be in accordance with the requirements of MIL-Q-9858 as augmented by EOS-4.1.

4.1.1 Responsibility for inspection and test. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified herein. Except as otherwise specified, the supplier may use his own or any commercial facilities acceptable to the government. The government reserves the right to perform any of the inspections set forth in the specifications where such inspections are deemed necessary to ensure that supplies and services conform to prescribed requirements.

4.2 Quality conformance inspections

4.2.1 Category I tests

4.2.1.1 Development tests. Development tests shall be conducted to verify the unit performance parameters, proper interfacing between units of the subsystems, and proper interfacing between subsystems. The development tests shall be conducted at the unit and/or subsystem levels. The type of developmental evaluation or test to be applied to verify satisfaction of the requirements defined in this specification are outlined in Table 9. The tests shall be conducted in accordance with EOS-4.2.

4.2.1.2 Qualification tests. System qualification shall be accomplished by means of tests on the individual units and/or as a normal consequence of having the units integrated in the system. The types of qualification evaluation or test to be applied to verify satisfaction of the requirements of this specification are outlined in Table 9. Qualification test verification methods and requirements shall be as defined in EOS-4.2.

4.2.1.2.1 Test classification. The types of tests shall include:

(a) Physical inspection

(b) Performance verification tests

(c) Subsystem operational tests

(d) System tests.

4.2.1.2.2 EMI tests. Design qualification tests shall be performed on the fully integrated system to verify compliance with the applicable electromagnetic interference and susceptibility control requirements.

4.2.1.2.3 Inspection sequence. The inspection sequence shall be governed by the following:

- Examination of the unit/subsystem shall be performed prior to functional testing.
- Funcational tests shall be performed prior to, during, where appropriate, and following any environmental testing. The functional testing conducted at the conclusion of an environmental test may serve as the functional testing to be performed prior to the next test.
- Environmental testing may be performed in any sequence.

Table 9. Verification Cross Reference Index

VERIFICATION CROSS REFERENCE INDEX									
LEGEND: N/A - Not Applicable; I - Inspection; A - Analysis; S - Similarity; T - Test									
SECTION 3 REQUIREMENTS	Not Applicable	Acceptance			Qualification				SECTION 4 VERIFICATION REQUIREMENTS
		I	A	T	I	A	S	T	
(TBD)									

SAMPLE

4.2.1.2.4 Failure criteria. The module shall exhibit no failure, malfunction or out-of-tolerance performance degradation as a result of the examinations and test specified herein. If a failure occurs during the performance of any test, the test shall be suspended and the discrepancy, failure reporting, analysis, and corrective action procedures as set forth in EOS-4.1 shall be followed.

4.2.2 Test conditions. Unless otherwise specified, the conditions under which all inspections are accomplished shall be specified below.

4.2.2.1 Atmospheric and environmental. All examinations and tests shall be conducted under the local prevailing pressure at the time of test, at a temperature of 75 $\pm 5^{\circ}$F and at a relative humidity of 55 percent or less.

4.2.2.2 Deviations of atmospheric conditions. When tests are performed with atmospheric conditions substantially different from the specified values, proper allowances for changes in instrument readings shall be made to compensate for the deviation from the specified conditions.

4.2.3 Temperature stabilization. Temperature stabilization shall have been achieved when temperature of the equipment varies not more than 5°F during a period of 15 minutes.

4.2.4 Measurements. Measuring instruments used to determine functional parameter values (such as voltage, frequency, current, flow, pressure, etc.) shall indicate true values with an accuracy determined by the tolerance allowed for the parameter variation itself, such that the measuring instrument shall not introduce an uncertainty greater than 10 percent of the allowable variation of the measured parameter. However, no such measurement accuracy shall be required to exceed 0.5 percent of the required value of the parameter unless otherwise specified.

Except as specifically noted in the inspection methods, tolerances for environmental test conditions shall be defined as follows:

Temperature: $\pm 5^{\circ}$F

Barometric pressure: ± 10 percent

Relative humidity: +5, -10 percent

Time: ± 5 percent

Vibration amplitude (sine): ± 10 percent

Vibration frequency: ± 2 percent.

4.3 System acceptance tests. TBD.

4.4 Analyses. Reliability requirements shall be verified by review of analytical data in accordance with EOS-4.1.

4.5 Category II tests. TBD.

5. PREPARATION FOR DELIVERY

TBD.

TITLE

PRIME ITEM DEVELOPMENT SPECIFICATION
FOR THE

LOW-COST GROUND STATION

DATE 20 SEPT 1974 NO. SP-313

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

ONE SPACE PARK • REDONDO BEACH, CALIFORNIA 90278

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SPECIFICATION SP-313

LOW-COST GROUND STATION

1. SCOPE

This specification establishes the performance, design, test and qualification requirements for the equipment comprising the Low Cost Ground Station (LCGS) of the Earth Observatory Satellite (EOS) system. The LCGS rapidly provides earth observation data to users in a format compatible with their unique needs. The station performs as an integral element of the EOS ground system.

2. APPLICABLE DOCUMENTS

2.1 Government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

SPECIFICATIONS

NASA

SP-312	Central Data Processing Facility Specification

Military

MIL-B-5087B	Bonding Electrical and Lightning Protection for Aerospace Systems

STANDARDS

Military

MIL-STD-130D	Identification Marking of U.S. Military Property
MIL-STD-188C	Military Communications System Technical Standards
MIL-STD-1472A	Human Engineering Design Criteria for Military Systems Equipment and Facilities

OTHER PUBLICATIONS

2.2 Non-government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. If no revision or date is shown, the latest released issue of the applicable document shall apply. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

OTHER PUBLICATIONS

EOS-3.3-8	Configuration Management Plan
EOS-4.1	System Effectiveness Program Plan
EOS-4.2	Integrated Test Plan

3. REQUIREMENTS

3.1 General

3.1.1 Function. The Low-Cost Ground Station (LCGS) provides earth observation data available to users in a timely manner and in a format which allows processing to their unique needs. The concept requires the implementation of an RF equipment subsystem for acquiring sensor data over an X-band link and an image processing subsystem to produce output products. NASA/LCGS interfaces are also provided to link the LCGS and the Operational Control Center (OCC) and the Central Data Processing Facility (CDPF). These interfaces and the relationship of the LCGS to the rest of the system are shown in Figure 1.

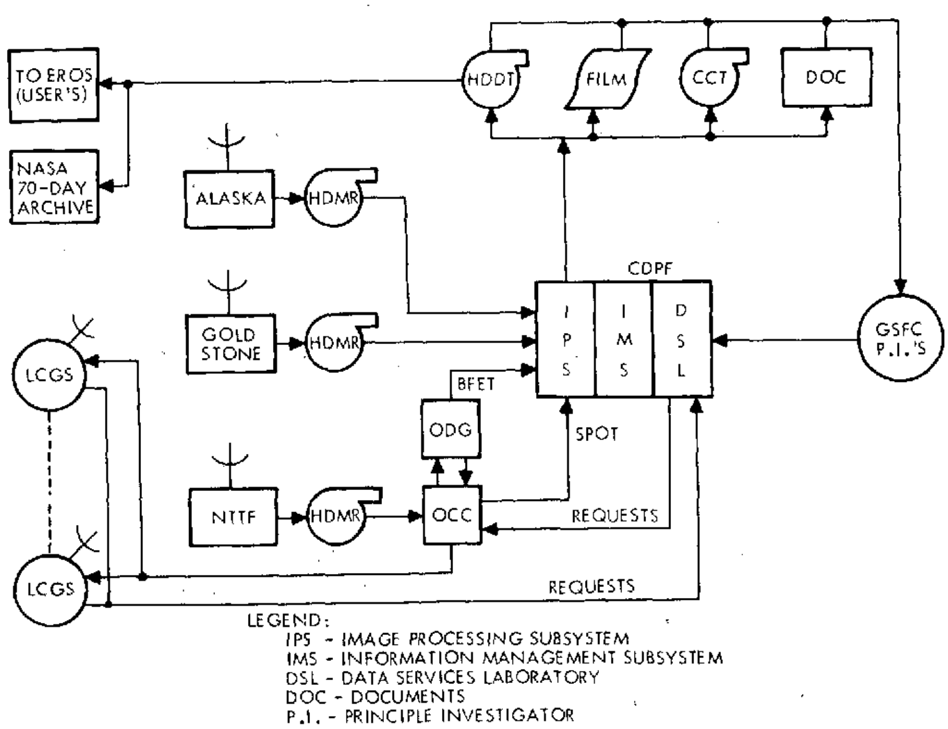

Figure 1. Baseline Ground Data Handling System Interfaces

The baseline design of the image processing subsystem consists of two processing equipment options.

(a) Direct display option

(b) Record and process option

Each option requires the same acquisition and operational interface characteristic between the user and NASA. The difference between the two is the approach to image processing. Direct display produces only film output products, recording them in real-time, while the record and process design uses a minicomputer-based system to process recorded image data. The latter produces a range of output products tailored to the needs of the user and has output data quality commensurate with the CDPF. Figure 2 summarizes the functional aspects of each design. Performance characteristics of each option are specified herein. Either or both options may be procured for any LCGS.

3.1.2 Operation. Figure 1 is a simplified block diagram of the baseline ground data handling system interfaces and is provided to illustrate the relationship of the LCGS installation to the other elements of the system. Figure 2 is a functional block diagram of the LCGS which identifies the subsystems that comprise the LCGS and the two image processing options. Figure 3 is an expanded block diagram of the acquisition equipment subsystem. Figure 4 provides a description of the record and process image processing option. Figure 5 is an expanded block diagram of the antenna positioning equipment and Figure 6 describes X-band data wipeoff demodulator.

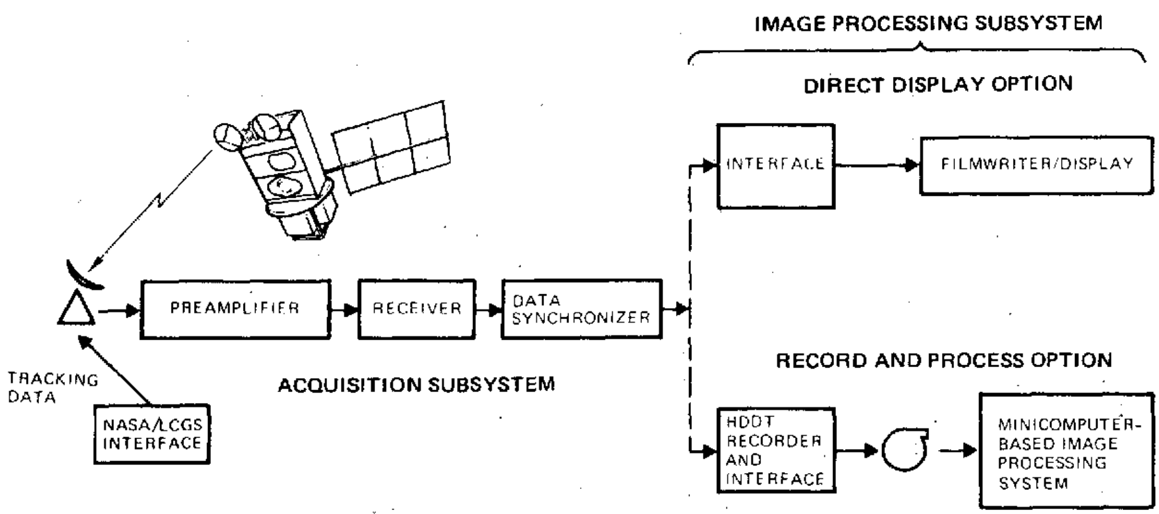

Figure 2. LCGS Functional Block Diagram

3.1.2.1 NASA/LCGS interface. A modem shall be provided using telephony (or its equivalent) for communication with the NASA control center; this link will be used to drive a paper tape machine to be used for the tracking input for the antenna system, and to provide ancillary data needed for LCGS operation. The equipment consists of a telephone subsystem to interface with standard commercial telephone equipment and a standard computer modem with two-way teletype and paper tape punch and read capability.

Only thematic mapper sensor data is transmitted to the LCGS. The baseline orbit yields a 17-day repeat cycle for this data coverage. The LCGS can acquire up to 10 scenes in one pass. Pass requests for the LCGS are sent to the CDPF one cycle prior to the desired pass date. The CDPF automatically processes these standing orders and serves as the interface between the LCGS and the control center. Priorities are established at this point for conflicting pass requests through arbitration by the EOS project office. The processed requests are delivered to the control center for scheduling the pass assignments. The control center generates tracking data for the LCGS antenna system and image correction process; these data are transmitted to the LCGS facility using computer data link technology on the same day as the LCGS pass coverage.

The concept is that the principal NASA/LCGS interface is implemented within the CDPF; the control center only interacts with the LCGS to satisfy operational pass requirements. The CDPF provides LCGS support for siting and equipment calibration; it also provides any requested technical information within the purview of NASA.

The acquisition equipment block diagram is shown in Figure 3. This system is contained in both designs and its function is to acquire the X-band downlink containing the edited thematic mapper sensor data. The antenna system uses a taped program for pointing and tracking and consists of an antenna, pedestal, feed, and control electronics. The tape program information would be prepared by the control center and punched at the LCGS using a telephone link and computer as indicated in Figure 1.

An uncooled parametric amplifier shall be used for the X-band preamplifier which satisfies the signal-to-noise ratio requirements for the communication link. The receiver shall be X-band biphase demodulator. The bit synchronizer and decommutator complete the subsystem components.

3.1.2.2 Image processing subsystem. The quality of the LCGS output products differs for the two baseline options. Radiometric calibration of the LCGS sensor data will be performed onboard ensuring the same potential radiometric accuracy for both designs. The direct display option only produces a film output product. Therefore, the radiometric accuracy of its output product will be intrinsically less than the digital products of the record and process option. Furthermore, the direct display will not perform any geometric corrections on the data. The film output product, then, will have a geometric accuracy dictated by the inherent accuracy of the transmitted sensor data and will be equivalent to the accuracy of the uncorrected HDDT output product of the CDPF. The record and process

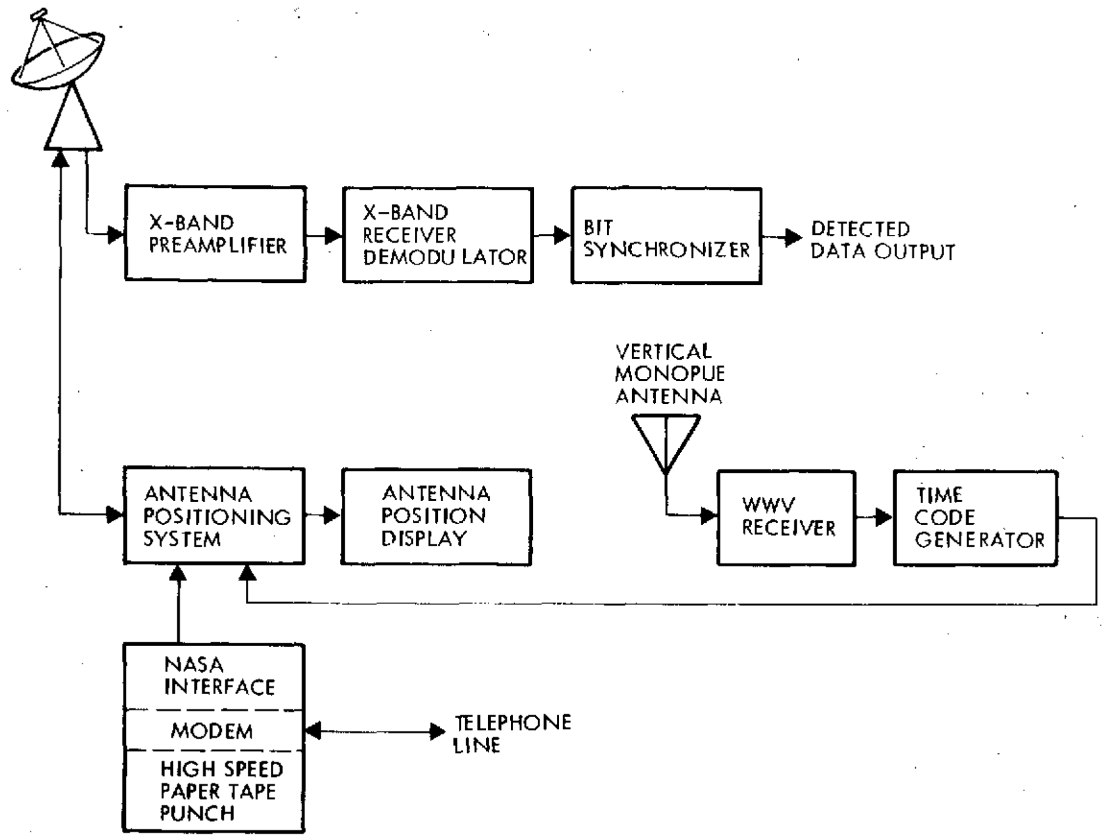

Figure 3. LCGS Acquisition Equipment and NASA Interface Subsystem

design is configured so that it has the capability of producing output products with geometric accuracy equal to the quality of those produced in the CDPF.

3.1.2.2.1 Direct display option. The output of the RF equipment subsystem shall be a frame-synchronized data stream. The direct display subsystem uses this data stream to produce an output film product.

The data stream is buffered, line-by-line, and then laser beam recorded on film using a continuous 9.5-inch film transport. The data is band interleaved, i.e., line one of band one, then line one of band two, etc. The same line is scanned across the film, one band at a time, then the next scan presents the next sensor line. A maximum resolution of 8192 pixels per scan is required.

The direct display option shall process the image data in real-time to produce film output products. It shall consist of a filmwriter and an interface to the RF equipment subsystem. This equipment will have the capability of recording 300 seconds of output image data on film. Geometric and radiometric corrections on the sensor data need not be performed by this equipment. Film size is the 241-mm (9.5 inch) format used by the CDPF. Nominal values of output data quality performance requirements are 5 percent relative radiometric accuracy, 50 percent MTF at 8192 pixels and 20 percent at 17,070 pixels for the film size and position accuracy of 0.2 percent of a resolution element. Standard laboratory environmental conditions during operation will prevail, but the capability for storage during periods of inactivity in an enclosed shelter while subjected to seasonal variations in weather typical over CONUS must be available.

3.1.2.2.2 Record and process option. The record and process option shall record the output data from the RF equipment subsystem, then digitally process the recorded data to produce corrected output products. A minimal set of output products shall be corrected film and corrected computer compatible tapes of all scenes recorded. This corrected output data quality shall be commensurate with the output product quality specifications of the CDPF. Figure 4 presents a diagram of a typical record and process option. The subsystem should be capable of processing 25 scenes during the period of one cycle, 17 days. The tape recorder shall be the same as used to implement the HDDT tape recorder function in the CDPF The LCGS tape recorder represents the means for obtaining additional TM scenes and HRPI data from the CDPF for subsequent processing. Sufficient flexibility shall exist in the subsystem to process this additional data.

3.1.3 Equipment list. TBS

3.2 Characteristics

3.2.1 Performance. The following paragraphs detail the performance criteria of the various systems and subsystems comprising the LCGS.

3.2.1.1 RF equipment and NASA interface subsystem (Figure 4). Performance characteristics for the RF and NASA interface components of the X-band telemetry system are specified below.

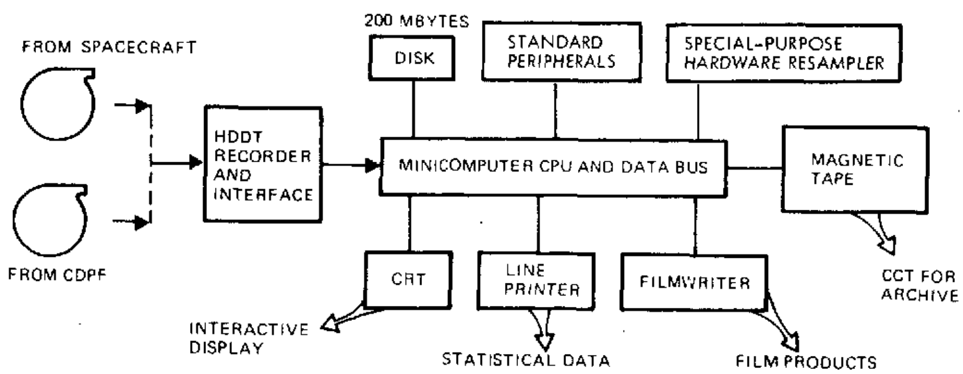

Figure 4. LCGS Record and Process Option

3.2.1.1.1 X-band antenna system. An 8-foot diameter parabolic dish antenna positioned by a numerically controlled servo system shall be used to receive the EOS X-band signal.

3.2.1.1.1.1 Parabolic dish antenna. The parabolic dish antenna shall have the following performance characteristics:

Diameter: 8 feet

Feed structure: TBD

Frequency range: 8.025 to 8.4 GHz

Minimum gain: 43.5 dB

Maximum 3 dB bandwidth: 1.1 deg

Maximum VSWR: TBD

Polarization: RHCP

Positioning accuracy: ±0.2 deg RMS

Slew rate: TBD

Input impedance: TBD

3.2.1.1.1.2 Punched tape programmer. A punched paper tape programmer shall be provided to permit programmed two-axis positioning of the servo-controlled X-band parabolic dish antenna. The programmer shall accept a punched paper tape which has been generated by a computer. The punched paper tape programmer shall meet the following performance criteria:

Positioning accuracy: ±0.1 deg

Inputs: TBD

Outputs: TBD

Tape coding format: standard ASCII 8-level code

3.2.1.1.1.3 Antenna position display. The antenna position display shall display, to the nearest tenth of a degree, the azimuth and elevation angles at which the parabolic dish antenna is positioned at any given time. The display shall be in digital form. Additional characteristics the display shall have are:

Input: 1.1 and 36:1 three wire synchro

Ranges: 0 to 360 deg and 180 ±180 deg

Accuracy: 0.2 deg ±1 significant digit (referred to synchro input)

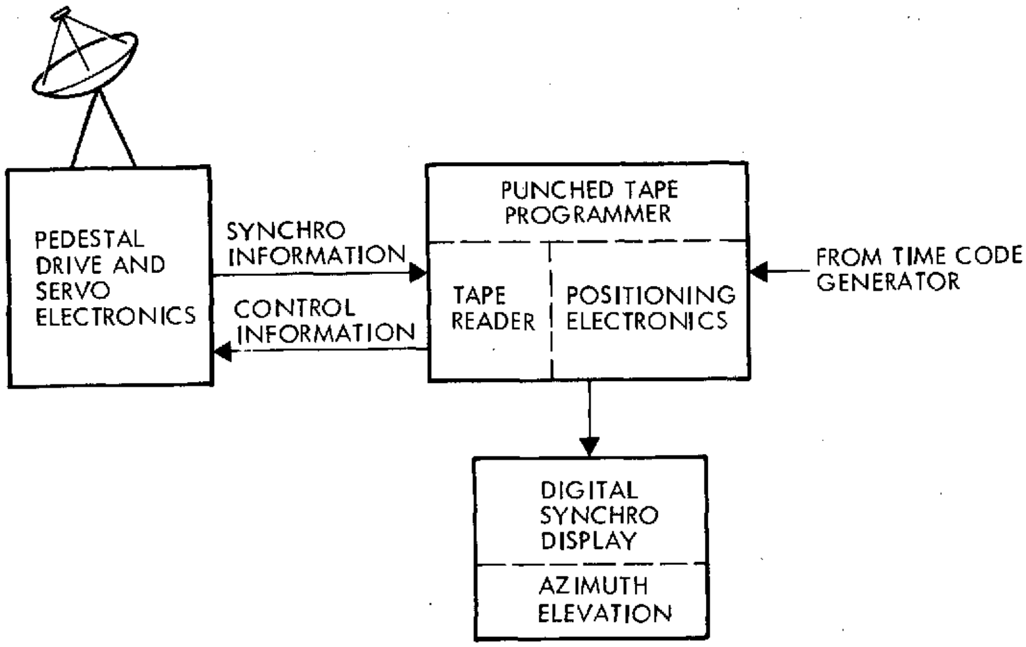

Figure 5. Antenna Positioning Equipment Block Diagram

3.2.1.1.2 X-band preamplifier. An X-band uncooled parametric amplifier shall be employed to establish the noise figure of the LCGS. The performance requirements of this unit are as follows:

Center frequency: 8.050 GHz

Bandwidth (-3 dB): 50 MHz

Noise figure: 1.5 dB, maximum

Gain: 15 dB, minimum

Input impedance: TBD

Output impedance: 50Ω

VSWR across passband: TBD

Dynamic range: TBD

Gain stability: TBD

Phase and gain linearity: TBD

3.2.1.1.3 X-band receiver/demodulator (Figure 6). An X-band receiver/PSK demodulator shall be used in the LCGS to track and demodulate the 20 Mbit/sec biphase signal utilizing the data wipeoff technique. This unit shall have the following minimum performance characteristics:

Input frequency: 8.050 GHz

Data format: biphase PCM/PM

Data rate: 20 Mbits/sec

Modulation index: TBD

Bit error rate: 1 in 10^6

Image rejection: TBD

Spurious responses: -70 dB

Input impedance: TBD

IF frequency: 1.5 GHz

IF bandwidth: 50 MHz

Local oscillator stability: TBD

Noise figure: TBD

Acquisition threshold SNR: ≥12 dB

Doppler range: TBD

AFC acquisition range: TBD

AFC search rate: TBD

Output impedance: TBD

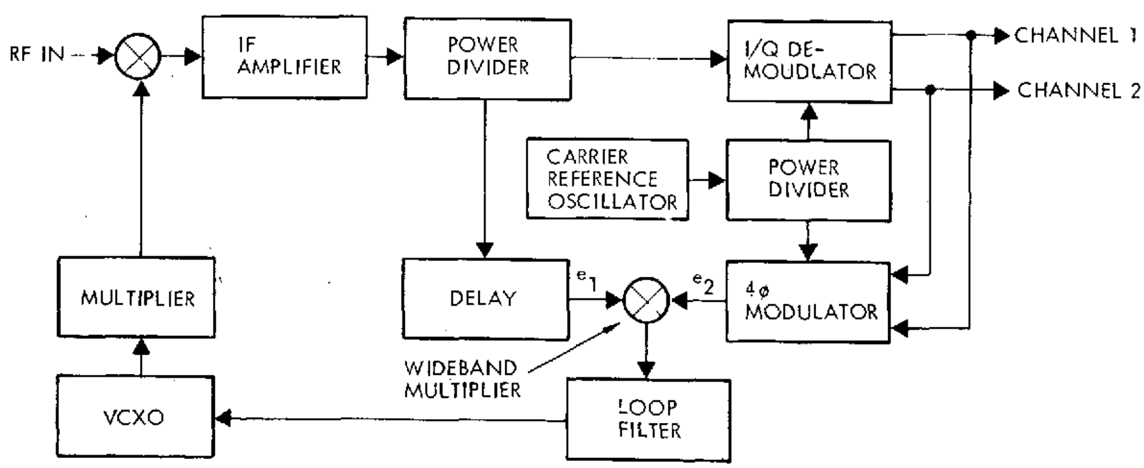

Figure 6. X-Band Data Wipeoff Demodulator

3.2.1.1.4 Bit synchronizer. A bit synchronizer shall be used to detect the demodulated 20 Mbit/sec biphase data. It shall have the following minimum performance characteristics:

Input level: TBD

Input impedance: TBD

Input code: biphase

Acquisition time: The bit synchronizer shall acquire and detect data at an SNR value that results in a BER of 10^{-3} in a maximum of 10 seconds

Capture and tracking range: TBD

Loop bandwidth: TBD

Loop stability: TBD

Synchronization threshold: TBD

Error performance: The degradation in bit error rate (BER) shall not exceed 1.5 dB from theoretical for bit error rates from 10^{-3} to 10^{-8}.

Cycle slipping: The average cycle slipping interval shall be 10 minutes or more at a data bandwidth SNR of +9 dB and 100 minutes or more at +12 dB.

Break lock interval: 10 sec at data bandwidth SNR of 0 dB

Data output format: NRZ

Bit rate clock outputs:

Output logic levels:

Data and clock output rise and fall times: TBD

Bit rate: 20 Mbits/sec

3.2.1.1.5 NASA interface. An interface shall be provided using telephony (or its equivalent) for communication with the NASA control center. This link will be used for the tracking input for the antenna system and to provide ancillary data needed for LCGS operation. The equipment will consist of a telephone subsystem to interface with standard commercial telephone lines and a computer modem with two-way teletype and paper tape punch and read capability. This equipment shall have the following specifications.

3.2.1.1.5.1 Modem. The electrical characteristics of the computer modem shall conform to MIL-STD-188C. The unit shall also meet the following specifications:

Data input format of transmitter: TBD

Data input amplitude to transmitter: standard TTL

Number of input lines to transmitter: TBD

Input impedance to transmitter: TBD

Output signal modulation of transmitter: TBD

Type of coupling: acoustic

Data rate: TBD

Data input format of receiver: TBD

Number of output lines from receiver: TBD

Output impedance of receiver: TBD

Output level from receiver: standard TTL

Input signal modulation of transmitter: TBD

3.2.1.1.5.2 Paper tape reader/punch. The paper tape reader/punch shall have the following characteristics:

Storage level: 8-level paper tape

Reader speed: TBD

Punch speed: TBD

Input impedance of punch: TBD

Output impedance of reader: TBD

Input signal level of punch: standard TTL

Output signal level of reader: standard TTL

3.2.1.1.6 Standard time unit. A standard time unit consisting of antenna, receiver, and associated time code generator shall be incorporate into the LCGS to perform the following functions:

(a) Receive WWV time code signals

(b) Generate a universal time code

(c) Provide system time synchronization with WWV.

3.2.1.1.6.1 Antenna. The antenna shall consist of a vertical monopole antenna having the following performance characteristics:

Height (from top of mounting structure): 20 feet

VSWR: 3:1 maximum

Input impedance: 50Ω, nominal

3.2.1.1.6.2 WWV receiver. A receiver capable of receiving all five frequencies transmitted by NBS radio station WWV shall be used having the following performance characteristics:

Sensitivity for 10 dB SNR: TBD

RF input impedance: 50Ω

BFO frequency: variable, 0 to 1000 Hz

Local oscillator stability: TBD

Tuning range: 2.5, 5, 10, 15, 20 MHz ±50 kHz

Audio output level: TBD

3.2.1.1.6.3 Time code generator. The time code generator shall generate a universal time code for operations and data time correlation and provide system time synchronization with WWV. It shall contain a crystal-controlled oscillator time base and a visual front panel decimal display of hours, minutes, and seconds. The time code generator shall also have the following characteristics.

(a) Time base accuracy of ±0.001 percent.

(b) BCD encoded parallel output of hours, minutes and seconds at standard TTL voltage levels for driving remote displays and ancillary devices.

(c) Output serial code in standard IRIG B, BCD encoded, hours, minutes, and seconds.

3.2.1.2 Image processing subsystem. Performance characteristics for the image processing system are detailed below. Two primary options are available: direct display option and record and process option. Each image processing system option shall process data derived from one of the four options presented in Table 1.

3.2.1.2.1 Direct display image processing subsystem. The direct display subsystem will process the RF equipment output in real-time to form thematic mapper images. It shall consist of a laser beam recorder and a unit to interface with the RF subsystem. The serial bit stream at 20 Mbit/sec is to be decommutated by the RF subsystem. Figure 7 shows the four formats for displaying up to seven bands to image data. No geometric or radiometric corrections are to be performed by the direct display subsystem.

Table 1. TM Sensor Data LCGS Options

Option	Swath Width (km)	Bands	Ground Resolution (M)	Pixel Line Length per Band (no. of bands)	
1	185 (full)	Any 1	30	8192	(1)
2	90 (1/2 full)	Any 2	30	4096	(2)
3	45 (1/2 full)	Any 4	30	2048	(3)
4	185 (full)	All	90	2731 683	(6) (1)

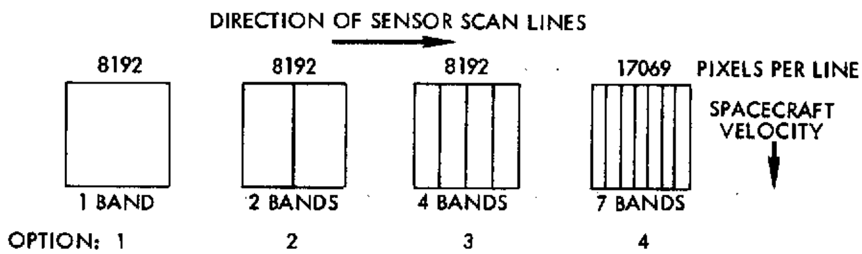

Figure 7. Image Format for TM LCGS Options

3.2.1.2.1.1 Interface to RF subsystem. It is assumed that the filmwriter input will be digital and thus the conversion to the analog signal necessary to intensity modulate the laser is accomplished within the filmwriter. Thus, the interface unit is required to provide the signals necessary to synchronize the filmwriter and possibly a line buffer. No digital formatting or data storage is envisioned. Functions and characteristics of the interface unit are:

Data word buffer: 8 bit parallel (as required)

Line sync: Pulse characteristics TBD

Transport sync: TBD

3.2.1.2.1.2 Filmwriter. The laser beam recorder shall accept 8 bits parallel data words and synchronizing signals from the interface unit and provide the end product of the direct display subsystem. Requirements and characteristics of this unit are as follows:

Recording capability for 10 scenes (185 km x 185 km per scene): 300 sec

Output data quality

Relative radiometric accuracy: 5%

Resolution element position accuracy: 0.2%

Modulation transfer function

at 8192 pixels: 50%
at 17,069 pixels: TBD (≈20%)

Spot wobble:
Transport jitter:
Scanning linearity:
Film distortion:
} Commensurate with sensor quality (0.1 pixel)

Dynamic range (grey level): 6 bit

Laser

Spectral region: TBD

Aperture function: TBD

Film

Size: 241 mm

ASA rating: TBD

Processing (laboratory vs direct develop): TBD

Transport type: Continuous

3.2.1.2.2 Record and process option. The record and process option shall consist of the following hardware and associated software:

(a) Minicomputer CPU with standard peripherals (teleprinter and fixed head disk memory, hs paper tape read/punch)

(b) High density digital tape recorder and interface

(c) High capacity removable disk memory

(d) Interactive CRT display

(e) Line printer

(f) Filmwriter

(g) Computer compatible magnetic tape recorder

(h) Special purpose hardware resampler

3.2.1.2.2.1 Minicomputer CPU and standard peripherals. A minicomputer with associated hardware and teleprinter, paper tape reader/punch, and fixed head disk memory shall be provided.

3.2.1.2.2.1.1 Minicomputer CPU hardware. The minicomputer CPU shall have the following performance characteristics:

Maximum memory size: 32k words

CPU cycle time: $\leq 0.5\ \mu sec$

Add time: $\leq 0.5\ \mu sec$

Automatic interrupt: yes

Floating point arithmetic: yes

Memory protection: yes

Number of registers: 8

Memory access time: $\leq 0.5\ \mu sec$

Number of CPU I/O channels: 9

Data word length: ≥ 16 bits

Addressing techniques: indirect, direct, relative, extended, and indexed

3.2.1.2.2.1.2 Minicomputer software. The following software packages shall be provided with the minicomputer CPU:

Disk operating system: real-time DOS

Assembler: Marco-assembler

Compilers: FORTRAN IV

Diagnostics: On-line diagnostic package

Utilities: Editor, file packages, library, linkage editor; overlay structure

3.2.1.2.2.1.3 Standard peripherals. The computer peripherals shall have the following performance characteristics:

(a) Teleprinter

Carriage width: at least 9 inches

Line length: 84 characters

Printable character set: 64 upper-case ASC II subset

Print speed 30 characters/sec

Input characteristics: 97 or 128 characters

(b) Fixed-head disk memory

Capacity/disk: 256k words

Data transfer speed: 16 μsec/word

Average access time (1/2 revolution): 17 msec

Number of tracks: 128

Words/track: 2048

Recording method: NRZI

Recording density: 1900 bits/in

(c) Paper tape reader/punch

Peak read speed (characters/sec): 300 characters/sec

Peak punch speed (characters/sec): 50 characters/sec

Checks: parity

Input characteristics: ASC II or binary formatted or unformatted

Output characteristics: ASC II or binary formatted or unformatted.

3.2.1.2.2.2 High density digital tape recorder and controller/interface. The high density digital tape recorder and interface to be incorporated into the LCGS record and process option shall meet the following performance criteria.

3.2.1.2.2.2.1 High density digital tape recorder. One HDDT tape unit shall be provided equivalent in performance to the Ampex FR 2028 unit specified in 3.2.1.1.1.1d, of Specification SP-312, Central Data Processing Facility.

3.2.1.2.2.2.2 HDDT controller/interface. The HDDT controller/interface shall perform the following functions.

(a) Accept data from the bit synchronizer at a rate of 20 Mbits/sec

(b) Output data to the HDDT at a rate of 20 Mbits/sec in a format which is acceptable to the HDDT.

3.2.1.2.2.3 High capacity removable disk memory. The high capacity removable disk memory shall have the following performance specifications:

Capacity/pack: 232 Mbytes

Data transfer speed: 624 Kbytes/sec

Average track positioning time: 32 milliseconds

Maximum track positioning time: at most 70 milliseconds

Data surfaces/drive: 20

Track/surface: 40

Sectors/track: TBD

Words/sector: TBD

Bits/words: TBD

Recording method: TBD

Recording density: TBD

3.2.1.2.2.4 CRT display. A CRT display is included in the LCGS record and process option for initial image evaluation. It shall have the following performance characteristics.

(a) Must contain an internal refresh memory sufficient for storage of TBD 512 x 512 images of up to 64 gray level intensities

(b) Must incorporate a trackball or joystick and hardware generated cursors

3.2.1.2.2.5 Line printer. A line printer shall be provided for printing out statistical data. It shall have the following performance specifications:

Number of characters: 132

Code set: ASC II

Number of lines per inch: depends on font

Character style: 5 x 7 or 7 x 10

Print rate: 3200 lines/minute (5 x 7 matrix)

Paper type: electrostatic

Number of columns: 132

3.2.1.2.2.6 Filmwriter. A high speed digital filmwriter shall be provided for the generation of corrected thematic mapper images. Input to the filmwriter shall consist of 8-bit parallel binary words and appropriate synchronizing data; and shall consist of a 9.5-inch exposed sheet of film. The filmwriter shall meet the following performance characteristics:

(a) Maximum size of photographic transparencies: 10 x 10 in

(b) Maximum photographic density (specular): 256

(c) Illuminating system square spot size: 25, 50, 100 microns

(d) Sampling grid or roster size: TBD

(e) Density range: 0.1 x 2.50 D

(f) Density resolution: 64 gray levels

(g) Density repeatability: 64 gray levels

(h) Positional accuracy: X-axis: TBD
Y-axis: TBD

(i) Data rates: at most 45 minutes/image

3.2.1.2.2.7 Computer compatible magnetic tape recorder. Magnetic tape products having corrected image data recorded in standard format are required to user needs for missed scenes and off-site image generation. The magnetic tape drive which shall be used to generate these computer compatible tape products shall have the following performance characteristics:

(a) Number of tracks: 9

(b) Recording method: NRZI or PE

(c) Recording density: 1600 and 6250 bpi

(d) Data transfer rate: 200 Kbytes/sec for 1600 bpi
100 Kbytes/sec for 800 bpi

(e) Tape speed: 125 ips

(f) Rewind speed: 500 ips

3.2.1.2.2.7 Special purpose hardware resampler. This optional component is essentially a microprocessor and is used for implementing high-order image processing algorithms and increasing data throughput. The hardware is composed of the across scan resampler and the along scan resampler.

3.2.1.2.2.7.1 Along-scan resampler. The along-scan resampler hardware shall be provided to perform the first step of the sensor image correction process. Its performance specifications shall be as follows: TBD.

3.2.1.2.2.7.2 Across-scan resampler. Across-scan resampler hardware shall be provided to perform the final step of the image correction process. It shall meet the following performance criteria: TBD.

3.2.2 Physical characteristics

3.2.2.1 Mechanical. TBD

3.2.2.2 Electrical

3.2.2.2.1 Power. The LCGS shall be operated from electrical power derived from local commercial sources. The quality and quantity of this input power shall be as shown in Table 2.

Table 2. Input Power

Power Input	
Continuous:	TBD watts
Standby:	TBD watts
Voltage:	208/120 Vrms
Phase:	3-phase (4 wire)
Steady-state voltage:	±5%
Steady-state frequency:	±3%
Voltage modulation:	1%
Frequency modulation:	1%
Harmonic content:	5% (maximum)
Crest factor:	1.414 ±10%
Phase unbalance:	5% (maximum)

High voltage transients due to lightning or other phenomena is 2000 volts, duration 500 microseconds (maximum).

The transformer secondary connection from the commercial line shall be 3-phase, 4-wire, wye, with solidly grounded neutral.

3.2.3 Reliability. Compliance with reliability requirements shall be taken by prediction techniques in conformance with EOS-4.1. The allocated reliability for the LCGS baseline configuration operating under conditions specified herein for a TBD period is TBD.

3.2.4 Maintainability. Maintenance of the LCGS shall be minimal in terms of special tools, time of maintenance and expense per installed hour of equipment. The supplier shall prepare and deliver preventive maintenance and equipment calibration instructions that will permit the LCGS to operate without maintenance for (TBD) hours elapsed time.

3.2.5 Environmental conditions. The LCGS equipment shall be designed and constructed to withstand any combination of the following service conditions without mechanical or electrical damage or performance degradation below that specified in the detailed equipment specifications.

The environmental conditions shown in Table 3 shall apply for each of the following categories of equipment:

Type I On-site and protected

Type II On-site and unprotected, exposed to ambient environment

Type III Transportation, packaged

The antenna, pedestal and pedestal-mounted equipment will be exposed to the TBD environment for an excess of TBD years. The design and manufacture of these equipments must consider this environment. An analysis identifying the design features employed and servicing actions required to survive in this environment shall be performed.

3.2.6 Transportability. Transportability requirements shall be considered in the design of the LCGS equipment such that it can be transported by all standard modes with a minimum of protection. Special packaging may be used, as required, to ensure that common carrier transportation does not impose design restrictions.

3.3 Design and construction. The contractor shall utilize to the maximum extent practicable, proven, existing equipment to meet the requirements and characteristics of Section 3.2 of this specification. Where existing equipment cannot meet performance, packaging or interface requirements, preference will be given to modification of existing equipment in order to permit collection of established performance history and accumulated part operating experience for all items comprising the LCGS. Use, application and operating characteristics of the LCGS shall be

Table 3. Environmental Conditions

Environment	Type I		Type II		Type III
	Operating	Non Operating	Operating	Non Operating	Non Operating
Temperature, °C	+25 ±5	+10 to +55	-10 to +55, +15 solar radiation	Same	-10 to 55, +15 solar radiation
Rainfall	N/A	N/A	100 mm/hr	Same	100 mm/ hr
Relative Humidity, %	50	90	30-100	Same	30-100
Wind	N/A	N/A	<95 km/hr Operate at specified accuracy 110 km/ hr drive to stow	<160 km/hr survive	
Salt Atmosphere	TBD	TBD	TBD	TBD	TBD
Salt Spray	N/A	N/A	N/A	N/A	TBD
Blow Sand and Dust	N/A	N/A	TBD	TBD	TBD
Fungus	Non-nutrient materials or treated to inhibit growth in tropic region	Same	Same	Same	Same
Shock and Vibration	TBD	TBD	TBD	TBD	TBD

documented, when so specified by contract. Additional material and technical information shall also be provided in support of any patent limitation, trade secret protection or compliance with any code or agency regulation of the U.S.A. governing operation, use frequency allocation or radiated power.

3.3.1 Materials, processes, and parts. Materials, processes, and parts of the LCGS shall be of excellent quality and of highest grades obtainable commensurate with precision electronics equipment manufactured for scientific field use. Toxic, critically limited and strategic materials shall not be used. Special processes not readily obtained by modern manufacturing and finishing techniques in commercial use shall also be avoided. Wherever possible, materials, processes and parts will be of recent manufacture and proven capable of providing their required function or life by recent demonstration or test. The utilization of any material process or part in the equipment shall not relieve the supplier of the responsibility for complying with all acceptance requirements in Section 4 or provisions covering the quality and conformance of materials, processes or parts furnished as spare articles. Materials or processes used as protective coatings or finishes for fungi and oxidation control shall be approved prior to application. Such materials to be applied by vendors shall be identified for evaluation by NASA.

3.3.2 Electromagnetic radiation. Electromagnetic interference requirements shall be as specified in this specification and as defined in the individual equipment specifications. The contractor shall ensure that undesirable electromagnetic radiations from his equipment are so limited as to not induce malfunctions or degradation to the performance of his equipment.

Provisions such as line filters, arc arrestors, fitted enclosures and shielding shall be provided for all powered equipment, wave guides and cables to ensure that undesirable frequencies and spurious noise spikes are sufficiently attenuated through shielding and distance to be below the threshold sensitivity of the receiving equipment:

3.3.2.1 General EMC/EMI requirements. In addition to the normal environmental RF noise sources such as factory equipment, machinery, and discrete RF signal sources, the various interference generators within the overall system itself must be considered as contributors to the EMI environment.

The broadband RF environment will consist of switching circuit, digital clock, teleprinter, and other equipment-generated interference of a broadband nature.

It is the intent of this specification to describe design practices to guide all equipment suppliers which will minimize the levels of this RF noise environment conducted and radiated to other equipments.

Electrical and electronic equipment shall operate without degradation, not only independently, but also in conjunction with other such equipment which may be placed nearby. This requires that the operation of all such equipment shall not be adversely affected by interference voltages and fields reaching it from external sources, and also requires that such equipment shall not, in itself, be a source of interference which might adversely affect the operation of other equipments.

3.3.2.1.1 Facilities

(a) The facility shall provide the basic ground plane for all equipment and subsystems installed.

(b) All equipment racks, consoles and peripheral equipment shall be electrically bonded to the ground plane by direct metal-to-metal contact.

3.3.2.1.2 Racks

(a) All equipment racks or consoles are preferred to be RFI-shielded type provided with RF gasketing or spring finger contact strips at all doors and front panel mating surfaces.

(b) All rack-mounted equipment front panels, blank panels or control panels shall have clean (conductive) rack mating surfaces.

(c) All racks utilizing AC power shall have power line filters (including the neutral) installed immediately behind the input power connector. The filters shall provide a minimum of 80 dB insertion-loss from 15 kHz through 10 GHz.

(d) The primary power safety wire shall be grounded to the rack immediately adjacent to the power input connector.

(e) AC power carried within a rack (usually twisted pair wire) shall be physically separated from signal or DC power wiring.

(f) Connectors shall not employ any nonconductive protective coatings. Connector mounting areas on racks shall be free from paint or any other nonconducting material. Filler materials between connectors and racks shall not impair the conductivity between mating surfaces. Silver-loaded epoxy or similar materials should be used as a filler if a dissimilar metal problem is anticipated.

3.3.2.2 Grounding configuration. The LCGS shall employ a common electrical reference ground plane system. In order to develop an effective system ground plane, it is imperative that the system be built into well designed equipment racks that are electrically bonded together, and that all rack-mounted assemblies are electrically bonded to the rack structures by continuous metal-to-metal contact. The rack structures shall then be securely bonded to the system electrical ground plane.

The following paragraphs defined the criteria that shall be used in the development of an electrical ground equipment system with a multi-point ground configuration. Single point grounding will be used only when an item of commercial equipment requires a single point ground configuration.

3.3.2.2.1 Racks and consoles. The type of racks or consoles specified for electronic equipment installation provides the basis for achieving an effective ground reference plane, as well as providing shielding integrity for the equipment mounted within the rack. Each rack or console shall have as a minimum the following features:

(a) All equipment racks shall be electrically bonded together.

(b) All mounting hardware within the rack shall be plated with a conductive finish.

(c) Electrical bonding between each group of racks shall be accomplished with either a copper bus bar 0.030 to 0.050 inch thick and 2.5 inches wide, or an equivalent copper braid 2.0 inches wide connected between the lower rear corners of each adjacent rack. At least 1/4-20 bolts with 3/4 inch washers should be used to fasten bonding bus bars to each rack.

3.3.2.2.2 Equipment drawers and chassis

(a) The back surface of all front panels shall be free of paint or anodized finishes in areas which mate to the rack mounting rails and shall be protected by a suitable conductive finish. All commercially purchased equipment panels which mate with the rack mounting rails shall meet this criteria. If this requirement cannot be satisfied with equipment as purchased, the equipment panels shall be modified to allow the desired bonding to be achieved.

(b) All metal-to-metal surfaces on the drawer, chassis, and components shall form a good electrical bond between the mated surfaces.

(c) Chassis containing integrated circuit cards shall be provided with a chassis ground bus which shall be mounted directly to and electrically bonded to the chassis.

(d) All rack-mounted drawer assemblies with front panels whose back surfaces are free of paint or anodized finishes shall be considered adequately bonded for grounding purposes when fastened to the rack structure with the necessary mounting screws.

(e) All rack-mounted drawers with slides or shelves shall have a minimum of one AWG No. 16 wire which grounds the drawer to the rack for personnel safety when intended drawer wiring will not satisfy this requirement.

For especially sensitive equipment which may be susceptible to extraneous interference when pulled out of the rack for service, calibration, etc., the use of a solid flexible copper alloy bonding strap is recommened in lieu of the No. 16 safety grounding wire.

3.3.2.2.3 Electrical circuit grounding

(a) Primary AC power neutrals shall be isolated from racks and chassis and treated exactly as the high side of the AC power line. RF filters shall be placed in each side of the line in a balanced configuration. Only one set of filters is normally required for each rack or console.

(b) Within units, signal and power returns shall be grounded to chassis by short direct wire or printed circuitry.

(c) Sharing or daisy-chaining of return lines or printed circuitry should be avoided. Where a single ground bus is used on a PC board, it should be chassis grounded at each board mounting pad.

(d) The DC return or ground line shall be carried in each secondary power cable between power supplies and user units and shall be grounded to chassis immediately adjacent to the connector in each case.

(e) Filter capacitor returns shall be connected to chassis by the shortest most direct means. Filter or decoupling capacitors shall not share common return lines with one another or with other circuits.

(f) Where one power supply supplies secondary power to several similar chassis, care must be taken to provide adequate filtering or decoupling in each unit served by the supply, in addition to avoiding common DC returns, to reduce interaction between units.

3.3.2.3 Bonding

3.3.2.3.1 Bonding definition. Electrical bonding shall be defined as the method by which all elements of the system, including all equipment racks, equipment panels, electrical and RF connectors, chassis covers and subassemblies, are electrically interconnected to establish a low impedance reference plane. The bonding methods and techniques employed in making the electrical connections (using MIL-B-5087B as a guide) shall ensure a minimum DC resistance and RF impedance between the above items.

3.3.2.3.2 Bonding methods. Electrical bonding shall be accomplished by metal-to-metal contact over the entire faying surface areas which are held in mechanical contact. Specific provisions shall be made to preclude contamination of bonding surfaces with nonconductive oxides and finishes. The electrical bonding technique employed shall provide, as a design goal, a bonding impedance not to exceed 1.0 ohm from 0.2 MHz through 10 GHz and as a firm requirement, a DC resistance of less than 2.5 milliohms at all unit-to-rack interfaces and between unit connector shells to unit cases.

3.3.2.4 Shielding criteria. Interference control by means of magnetic or electrostatic shielding techniques shall be assessed in terms of benefits derived. Maximum utilization of the inherent shielding effectiveness provided by equipment racks, equipment enclosures, and physical location/isolation parameters, will be considered in the application of shielding design to control internally and externally generated electromagnetic environments and to afford suitable protection to susceptible system elements.

3.3.2.4.1 Equipment shielding. The enclosures of all equipment capable of generating EMI or having excessive sensitivity to external fields shall be designed to provide necessary attenuation of electromagnetic fields. Electrical bonding of connector shell hardware, enclosure panels, cover plates, base plates and end panels will be provided in accordance with paragraph 3.3.2.3.2. The following items shall be observed in the design of unit packaging, structural design and equipment mounting:

(a) Access openings will be kept to a minimum size compatible with the equipment performance requirements. Multiple small openings are preferred to large single openings of the same cross-sectional area. Access openings for adjustment screws or visual inspection will be provided with cover plates or screw-on caps for continuity of the shielding design when not in use.

(b) Specific provisions will be made to ensure constant and uniformly distributed contact pressure across the metal-to-metal bonded joints. A suitable electrically conductive interface filler material, such as an approved knitted-wire mesh gasketing, may be applied between the interface surfaces to fill the voids and thus ensure the necessary continuity across the entire bonded joint.

3.3.2.4.2 Cable shielding. Electromagnetic shielding shall be employed on all interface cables between units or racks which are capable of generating excessive electromagnetic fields or which may be susceptible to electromagnetic fields. The degree of shielding employed shall be sufficient to ensure an adequate compatibility margin. Wire shields shall not be utilized as an intentional current carrying conductor. All shields shall be covered by a layer of insulation.

(a) All RF signals not run in waveguide shall be carried in semi-rigid or double-shielded coaxial cable.

(b) Within racks, all signal circuits and DC power shall be shielded. Shields shall be grounded to the rack structure at the input connector and the unit chassis. Between units within a rack, shields shall be grounded to the unit chassis of each unit.

(c) Shield grounds shall be made via the connector shell wherever practical. Daisy-chaining of shield grounds should be avoided. Shields should be grounded individually by short (preferably the shield braid itself) direct means to the connector back-shell. (RFI back-shells are available for most connector types.)

(d) Between racks, unbalanced signal circuits shall use single conductor shielded cable properly grounded as described above to the rack at each interface with an overall shield over the cable bundle grounded to the connector shell at each interface.

(e) Control circuits, voltage or current step functions, teletype or line printer circuits should be carried in cable bundles with analog or digital signals.

(f) Balanced signal circuits shall be run in twisted shielded pair cable with shields grounded at source and load end.

(g) Primary shields of individual or multiconductor wires within the cable shall be carried through the connector on one or more pins and connected to the housing or chassis immediately adjacent to the connector within the unit, utilizing the shortest possible terminating lead length. In no case shall this terminating wire exceed 1 inch in lenth.

(h) Where a large number of separate shields are used within the cable, the individual shields may be tied to a "halo-ring" and two grounding leads from the "halo-ring" carried through the connector and terminated at the rack by wires from the receptacle. The "daisy chain" technique of terminating shields shall not be used.

(i) Shields shall not be used intentionally for signal or power returns except in the case of coaxial cables.

(j) Internal harnesses should be designed, grouped, and routed to minimize interference effects due to common circuit impedances, cross-talk, radiation, and pickup.

3.3.2.4.3 Interface cable installation. In addition to cable shielding, proper segregation, transposition, and routing requirements for cables and cable bundles will be considered in the design of all subsystem and equipment interface cabling.

(a) Physical isolation. Interface cabling of different signal categories will be physically isolated from each other by employing separate cable bundles. Typical cable bundling categories recommended are as follows: primary power, secondary power, RF input, RF output, digital, analog, and command (discrete pulse).

(b) Installation. All interface cabling will be routed in proximity to, and preferably in direct contact with, the electrical reference plane (structures).

3.3.2.4.4 Control and display signals

(a) Control circuits are defined as switch circuits which operate indicator lights, relays, or cause other direct application or interruption of power to electrical or electromechanical devices.

(b) All control line returns shall be maintained separate from all data signal returns. Individual control line returns shall be AWG No. 20 wire (minimum) and shall be grounded directly to chassis, or chassis ground bus when applicable. If lamps, indicators, or relays are grouped together such as on a front panel, they may be bussed together and connected to chassis ground with an AWG No. 18 stranded wire (minimum).

Control lines between drawers or racks should be run in single shielded or multiple conductor shielded cables. Shields shall be grounded to chassis or rack at each connector interface.

(c) Remote control circuit lines emanating from a drawer or console should have their returns grounded only at the power supply. Multiple control functions may employ a common return, but shall not be returned through the shield or a conductor common to any signal function. Control lines within drawers and between racks should be run in twisted pair shielded or multiple conductor shielded cables. Shields shall be grounded to chassis or rack at each connector interface.

3.3.2.5 Circuit decoupling. Design provisions will be made to en ensure that circuit isolation decoupling devices are properly selected and installed to achieve maximum effectiveness through their use. The provision for the application of such devices will be required in the following areas.

3.3.2.5.1 Power decoupling. Power conductors will be decoupled directly at the entering or exiting interface of an equipment. RF decoupling devices of the feedthrough variety will be employed, wherever practical, in the circuit design of equipment.

(a) Each equipment rack shall incorporate a balanced L-C type filter, designed to provide maximum isolation from 15 kHz to 10 GHz (also see paragraph 3.2.2).

(b) In conjunction with the basic input filter design of each rack, each primary input and return line of sensitive equipment such as receivers shall employ an RF decoupling device of the feedthrough variety to obtain RF isolation as required for that unit.

(c) The feedthrough filtering devices will be of the flange or screw-neck mounted types and will be mounted through the equipment enclosure wall or an RF extension thereof, with total circumferential filter-case contact to the electrical reference plane.

3.3.2.5.2 Analog circuit decoupling. Analog circuits will be decoupled, wherever required, by means of capacitive filtering components designed to eliminate power switching and rectification spikes and digital signal coupling.

3.3.2.5.3 Transient suppression. Transient or arc suppression components shall be employed in the design of all switching, pulse, relay, and solenoid circuit applications. Transient suppression techniques will be applied to limit the amplitude and duration of single-event switching transients as well as to control the rise and fall times of recurring pulse signals.

(a) Switching contacts. Integrated R-C type suppression components will be applied directly across all relay or switching contacts designed to provide closure or switching of inductive type loads at the primary bus voltage. Typical applications of such suppression components for the equipment will include: control relays, high-level command relays, and RF switching solenoids.

(b) Relays. Diode suppression components shall be applied directly across all relay coils.

(c) Solenoid operated devices. Transient suppression techniques will be applied in the design of all solenoid control circuitry and associated interface circuitry. In remote solenoid applications, where diode suppression components cannot be applied directly across the solenoid coil or transient source, alternate provisions of transient suppression and control of generated interference will be provided.

3.3.3 Nameplates and product marking. Each subsystem shall be identified in accordance with MIL-STD-130. Where practical, a minimum character height of 0.9 inch shall be used. Marking shall include the manufacturer's name or initials, assembly part number and revision status, assembly serial number, contract number, and others as delineated in the component equipment specifications.

3.3.4 Workmanship. Equipment covered by this specification shall be engineered, assembled and tested in a manner that appearance, fit and adherence to specific tolerances shall be observed. Particular attention shall be given to the neatness of parts and assemblies, cleaning, finishing, plating, painting, drilling, machine screw assemblage, welding, brazing and freedom of parts and connections from debris, burrs and sharp edges.

3.3.5 Interchangeability and part substitution. All equipment supplied under provisions of this contract shall be examined and provided with features of interchangeability to accommodate direct replacement of identical functioning assemblies and parts during the life cycle of the equipment. Additionally, parts having suitable operating characteristics which may permit substitution of an original part shall be so identified for purposes of repair and spare-parts provisioning. Substitute parts shall be identified by manufacture, part or drawing number and other identifying characteristics to permit an original part to be totally substituted by an alternate from commercially available sources.

3.3.6 Safety and hazards control. The supplier of the equipment covered by this specification shall provide for both personnel safety and equipment safety. Each of these safety categories will encompass safe, marginal, critical and catastrophic conditions coincident with the development, manufacture, test, installation, handling, training and operation of the LCGS. The supplier shall also be required to disclose that his facility complies with appropriate Federal, State and local safety codes and regulations governing the identification, analysis and control of hazards associated with the above mentioned conditions. In addition, the supplier shall conduct a program of safety and hazards control that identifies dangerous events and their means of control while the supplier's equipment is operating. Demonstration of special safety and hazard-control features of the LCGS shall be in compliance with provisions of Section 4.

3.3.7 Human performance/human engineering. MIL-STD-1472A shall be used for the design of man/machine interfaces.

3.4 Documentation. Documentation shall be as specified in EOS-3.3-8.

3.5 Logistics. All equipment may be unpacked, installed, calibrated and operated by NASA, or personnel other than supplier personnel, at designated destinations. It is for this reason that the supplier shall ensure that all necessary parts, supplies, information and other provisions for the equipment are accounted for and are complete.

3.5.1 Maintenance. The installed equipment shall be configured for ease of maintenance. Drawers or modules to be installed in racks shall be equipped with slides and cables of sufficient length to provide access to the unit without loss of power or signal input when the unit is extended from the rack for required servicing and maintenance actions. All rack-mounted or rack-supported hardware shall be capable of being maintained from the front of the rack. All drawers shall be provided with positive locking mechanisms. Where equipment maintenance, testing or repair requires access to the bottom of the chassis, the slides shall be tilt type with detent locking at 45 and 90 degrees tilt. Modules shall be arranged so that access to any module will not require removing any other modules or parts except access panels.

3.5.2 Spares. The contractor shall provide NASA with his recommendations of quantities of spare equipment necessary to support operations for an 18-month period.

3.5.3 Facilities and facility equipment. The vendor shall provide installation planning and design engineering instructions for preparation of sites for installation and facility equipment designs for installing and operating the equipment covered by this specification. Such data shall include, but not be limited to, reinforced concrete foundations and related structure(s), grounding and power components and requirements and facilities interfaces. Vendors designs and data shall comply with all environmental and other constraints identified herein.

3.5.3.1 Site data. Site soils data, ground conducting data and appropriate base facilities drawings will be provided to the supplier in accordance with contract schedules.

3.5.3.2 Reinforced concrete construction. Reinforced concrete construction, materials and use of equipment required for construction will be identified by the contractor at the design review.

3.5.3.3 Materials of construction. Vendor shall select suitable facilities materials to be incorporated in his designs and data from facilities materials lists which will be furnished to the supplier in accordance with contract schedules.

3.5.3.4 Construction tolerances. Reinforced foundations, related structures, power supply, and atmospheric lightning terminations and other facilities interfaces will be provided in support of suppliers' equipment in accordance with these construction tolerances except as indicated below.

3.5.3.4.1 Formed or finished concrete surface tolerances. Interior and exterior exposed surfaces will be dimensionally true to within 1/4-inch.

Exposed or finished surfaces will be flat, level and plumb as applicable to within: 1/4-inch in 10-feet, 3/8-inch in 20-feet, and 3/4-inch in 40-feet or more.

Angular dimensions shall be ±2 degrees but will not exceed ±3/8 inch from the true calculated position.

Deviations from indicated center-to-center dimensions with an integrated group of anchors or fasteners in reinforced concrete shall be within ±1/16 inch. Where anchors or fasteners interface with other parts, tolerance of the part with the most stringent tolerances will apply. Deviations from indicated center-to-center dimensions within an integrated group of holes, bolts and studs in structural steel assemblies shall be within ±1/32 inch.

Location dimensions for equipment or structures shall be ±1/8 inch for dimensions 8 feet or less; ±1/4 inch for dimensions more than 8 feet.

Length of structural steel members with milled ends shall be within ±1/32 inch.

Length of structural steel members without milled ends less than 30 feet in length shall be within ±1/8 inch. Any single structural steel dimension more than 8.0 feet or the cumulative tolerance on any group of related dimensions giving the overall distance between any two points, more than 8.0 feet shall be within ±1/4 inch.

Location of penetrations through structure and location of equipment shall be within ±1 inch, but location of piping, ducting and conduit or cable shall be within ±2 inches.

4. QUALITY ASSURANCE PROVISIONS

4.1 General. The quality assurance program controls shall be in accordance with the requirements of MIL-Q-9858 as augmented by EOS-4.1.

4.1.1 Responsibility for inspection and test. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified herein. Except as otherwise specified, the supplier may use his own or any commercial facilities acceptable to the government. The government reserves the right to perform any of the inspections set forth in the specifications where such inspections are deemed necessary to ensure that supplies and services conform to prescribed requirements.

4.2 Quality conformance inspections

4.2.1 Category I tests

4.2.1.1 Development tests. Development tests shall be conducted to verify the unit performance parameters, proper interfacing between units of the subsystems, and proper interfacing between subsystems. The development tests shall be conducted at the unit and/or subsystem levels. The type of developmental evaluation or test to be applied to verify satisfaction of the requirements defined in this specification are outlined in Table 4. The tests shall be conducted in accordance with EOS-4.2.

4.2.1.2 Qualification tests. System qualification shall be accomplished by means of tests on the individual units and/or as a normal consequence of having the units integrated in the system. The types of qualification evaluation or test to be applied to verify satisfaction of the requirements of this specification are outlined in Table 4. Qualification test verification methods and requirements shall be as defined in EOS-4.2.

4.2.1.2.1 Test classification. The type of tests shall include:

(a) Physical inspection

(b) Performance verification tests

(c) Subsystem operational tests

(d) System tests

4.2.1.2.2 EMI tests. Design qualification tests shall be performed on the fully integrated system to verify compliance with the applicable electromagnetic interference and susceptibility control requirements.

4.2.1.2.3 Inspection sequence. The inspection sequence shall be governed by the following:

- Examination of the unit/subsystem shall be performed prior to functional testing.
- Functional tests shall be performed prior to, during, where appropriate, and following any environmental testing. The functional testing conducted at the conclusion of an environmental test may serve as the functional testing to be performed prior to the next test.
- Environmental testing may be performed in any sequence.

4.2.1.2.4 Failure criteria. The module shall exhibit no failure, malfunction or out-of-tolerance performance degradation as a result of the examinations and test specified herein. If a failure occurs during the performance of any test, the test shall be suspended and the discrepancy, failure reporting, analysis, and corrective action procedures as set forth in EOS-4.1 shall be followed.

Table 4. Verification Cross Reference Index

VERIFICATION CROSS REFERENCE INDEX									
LEGEND: N/A - Not Applicable; I - Inspection; A - Analysis; S - Similarity; T - Test									
SECTION 3 REQUIREMENTS	Not Applicable	Acceptance			Qualification				SECTION 4 VERIFICATION REQUIREMENTS
		I	A	T	I	A	S	T	
(TBD)									

SAMPLE

4.2.2 Test conditions. Unless otherwise specified, the conditions under which all inspections are accomplished shall be specified below.

4.2.2.1 Atmospheric and environmental. All examinations and tests shall be conducted under the local prevailing pressure at the time of test, at a temperature of 75 ±5°F and at a relative humidity of 55 percent or less.

4.2.2.2 Deviations of atmospheric conditions. When tests are performed with atmospheric conditions substantially different from the specified values, proper allowances for changes in instrument readings shall be made to compensate for the deviation from the specified conditions.

4.2.3 Temperature stabilization. Temperature stabilization shall have been achieved when temperature of the equipment varies not more than 5°F during a period of 15 minutes.

4.2.4 Measurements. Measuring instruments used to determine functional parameter values (such as voltage, frequency, current, flow, pressure, etc.) shall indicate true values with an accuracy determined by the tolerance allowed for the parameter variation itself, such that the measuring instrument shall not introduce an uncertainty greater than 10 percent of the allowable variation of the measured parameter. However, no such measurement accuracy shall be required to exceed 0.5 percent of the required value of the parameter unless otherwise specified.

Except as specifically noted in the inspection methods, tolerances for environmental test conditions shall be defined as follows:

Temperature: ±5°F

Barometric pressure: ±10 percent

Relative humidity: +5, -10 percent

Time: ±5 percent

Vibration amplitude (sine): ±10 percent

Vibration frequency: ±2 percent

4.3 System acceptance tests. TBD

4.4 Analyses. Reliability requirements shall be verified by review of analytical data in accordance with EOS-4.1.

4.5 Category II tests. TBD

5. PREPARATION FOR DELIVERY

TBD

TITLE

PRIME ITEM DEVELOPMENT SPECIFICATION
FOR THE

SPACECRAFT STRUCTURE ASSEMBLY

DATE 20 SEPT 1974 NO. SP-1111

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

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SPECIFICATION SP-1111

SPACECRAFT STRUCTURE/THERMAL ASSEMBLY

1. SCOPE

This specification establishes the requirements for the performance, design, development, test, and quality assurance provisions for the spacecraft structure assembly, hereafter referred to as the assembly. The assembly includes the observatory electrical integration, provisions for thermal control, and provides support and protection for all elements of the Earth Observatory Satellite.

2. APPLICABLE DOCUMENTS

The following documents of the exact issue specified form a part of this specification to the extent specified herein. In the event of conflict between documents referenced and other detail contents of this specification, the detail requirements herein shall be considered superseding.

SPECIFICATIONS

<u>NASA</u>

JSC Spec SL-E-0002	Electromagnetic Compatibility Control Plan
SP-1	EOS System Specification
SP-115	EOS Environmental Specification
SP-1112	Communications and Data Handling Module Specification
SP-1113	Power Module Specification
SP-1114	Attitude Determination Module Specification
SP-1115	Actuation Module Specification
SP-1116	Solar Array/Drive Specification
SP-1121	Instrument A Specification
SP-1122	Instrumentation B Specification
SP-1123	Data Collection Module
SP-1124	Wideband Communication Module Specification
SP-1125	TDRS Data Handling Module

Military

MIL-B-5087	Bonding, Electrical, and Lightning Protection for Aerospace Systems
MIL-E-8983A	Electronic Equipment, Aerospace Extended Space Environment, General Requirements

STANDARDS

Military

MIL-STD-130	Nameplates
MIL-STD-143	Standard and Specifications, Order of Precedence and Selection
MIL-STD-454	Standard General Requirement for Electronic Equipment
MIL-STD-462	EMI/EMS Test Methods
MIL-STD-470	Maintainability Program Requirements (for Systems and Requirements)
MIL-STD-749	Preparation and Submissions of Data for Approval of Nonstandard Electronic Parts
MIL-STD-882	System Safety Program for Systems and Associated Subsystems and Equipment, Requirements for
MIL-STD-1472A	Human Engineering Design Criteria for Military Systems, Equipment, and Facilities

EOS Program Documents

EOS-3.3-1	Program Authorized Parts List
EOS-3.3-2	Program Authorized Materials List
EOS-3.3-3	Program Authorized Processes List
EOS-3.3-4	Electromagnetic Compatibility Control Plan
EOS-3.3-5	EMI/EMS Limits and Test Methods
EOS-3.3-6	Marking of Parts and Assemblies

EOS-3.3-7	Safety Design Criteria
EOS-3.3-8	Configuration Management Plan
EOS-4-1	System Effectiveness Program Plan
EOS-4-2	Integrated Test Plan

DRAWINGS

2.2.1	EOS Baseline Structural Configuration
2.2.2	Configuration for SP-1112, SP-1113, SP-1114, Module Structures (Figures 6-49, 6-50 of Report 3)
ICD 10.10	Spacecraft/Payload Electrical ICD

3. REQUIREMENTS

3.1 General. The structure assembly, together with the modules defined by SP-1112 through SP-1116 and SP-1121 through SP-1125, forms the EOS Observatory and is intended to satisfy the requirements of SP-1.

3.2 Characteristics

3.2.1 Structural assembly. The spacecraft structure assembly consists of the spacecraft structure, transition ring, structure harness, thermal control, solar array release and deployment mechanisms, and solar array release ordnance. While not part of the structure assembly, the requirements for the module structure design are included herein. The module structure will be provided to the spacecraft module contractors by the structure assembly contractor.

3.2.1.1 Spacecraft structural description. The spacecraft structure consists of a primary structure (equilateral triangular main frame) plus five subsystem modules, defined by SP-1112 through SP-1116.

3.2.1.1.1 Primary structure. The primary spacecraft structure consists of four major subassemblies:

(1) Equilateral triangular main frame outlined by three longerons projecting aft from the plane of the transition ring

(2) Aft stiffened cylindrical shell structure which mounts the forward section of the separation joint

(3) An I-section transition ring and sandwich bulkhead in the plane of the ring, plus load fittings, for Shuttle compatibility

(4) Six struts joining the transition ring to the forward end of the aft cylindrical shell structure.

Overall dimensions are defined in Figure 1.

The forward end of the longerons intercepts the bulkhead in the plane of the transition ring. The bulkhead and ring serve as load redistribution members between the payload section and the spacecraft section of the Observatory.

The structural design shall have provisions for the actuation module (Ref. SP-1115) to be space-serviceable by the addition of special attach mechanisms. Serviceability is defined as the capability of being exchanged within the Shuttle orbiter payload bay by means of a special purpose manipulator system. The structure includes a beam to pick up two of the four actuation module attach points for the serviceable case only (Figure 1).

3.2.1.1.2 Spacecraft modules. The EOS baseline spacecraft has five modules. Three are very similar structurally (Figure 2), with internal bulkheads and platforms to accommodate the specific equipment complement of each. (Ref. SP-1112, SP-1113 and SP-1114). These similar modules are 48 x 48 x 18 inches. Bolts are used for attaching the modules to the primary structure, but provisions shall be included for in-orbit serviceability by attachment mechanisms replacing the bolts.

The remaining module (Ref. SP-1116) is of similar construction but differs dimensionally.

3.2.1.1.3 Solar array deployment/retention mechanisms. Retention of the solar array panels (and the wideband communications and omni antennas) shall be accomplished with redundant pyrotechnic devices (Shuttle approved ordnance). Initial deployment shall be made with spring-driven hinges and pivots. The boom hinge shall initiate unfolding of the panels upon reaching a latched position. The configuration shall require three boom hinges for more compact storage due to its confined fairing. In this case, a cable shall maintain the proper angular relationship of the hinged joints during deployment.

For retracting the array to permit Observatory refurbishment or retrieval, the SAMS manipulator, part of the Shuttle Orbiter standard equipment complement, will be used. Each hinge latch shall have an over-center lever attached to facilitate disablement by SAMS. The array will be sequentially retracted by SAMS to the stowed position, where over-center clamps secure the stowed array to the Observatory structure against retrieval loads.

3.2.2 Electrical integration. Electrical integration of the EOS spacecraft design shall be based on the use of data bus and on-board computer capability for intermodule communications. A simple backup function with a minimum number of hardlines shall ensure that the Observatory can be retrieved or resupplied. Primary power for the modules shall be distributed on redundant pairs. The data bus shall consist of two pairs (four lines) to provide full duplex operation. A separate pair of lines provides power for the heaters in each module. The use of separate lines for the module heaters allows thermal maintenance during Shuttle servicing without powering the total spacecraft.

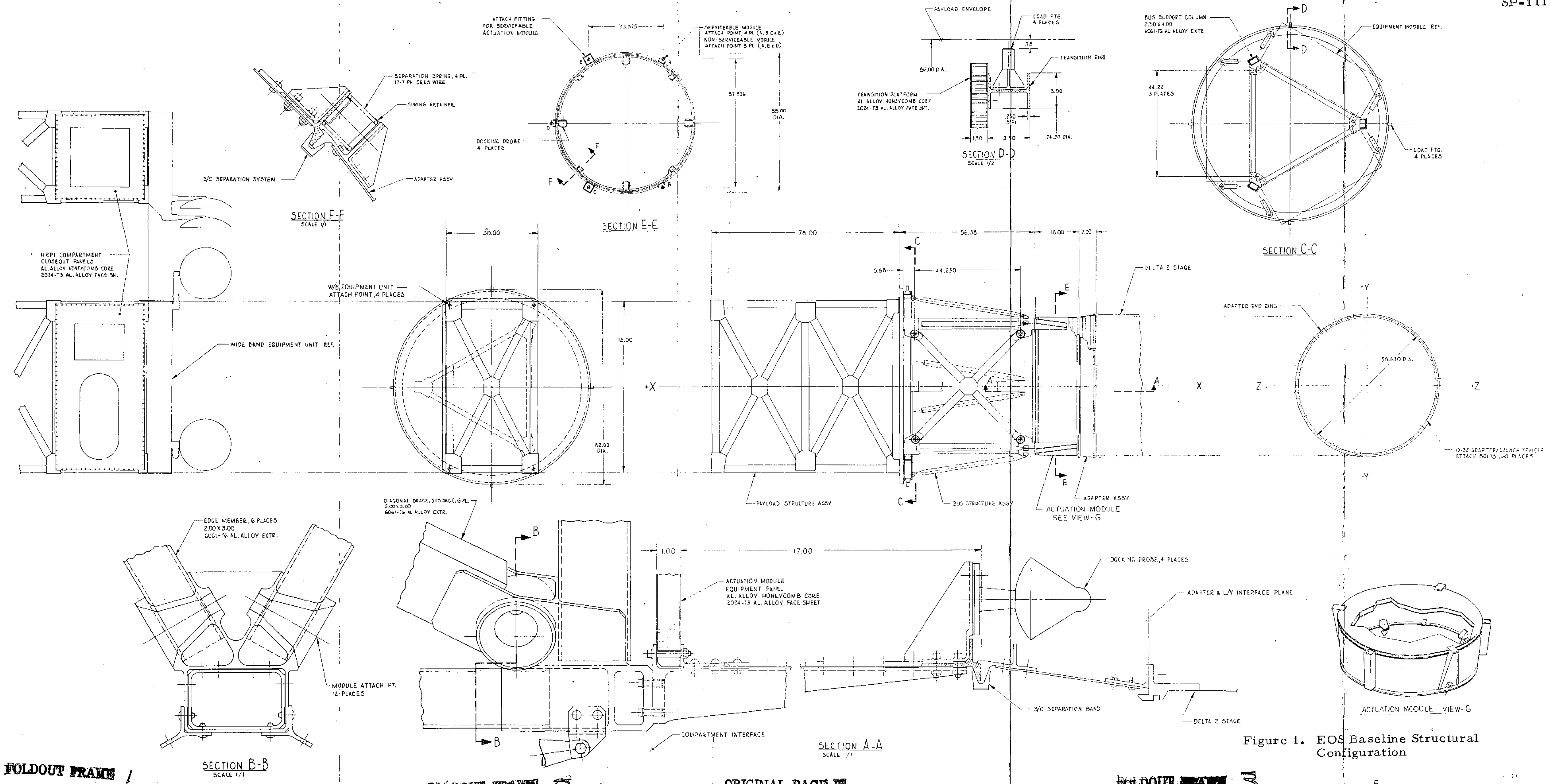

Figure 1. EOS Baseline Structural Configuration

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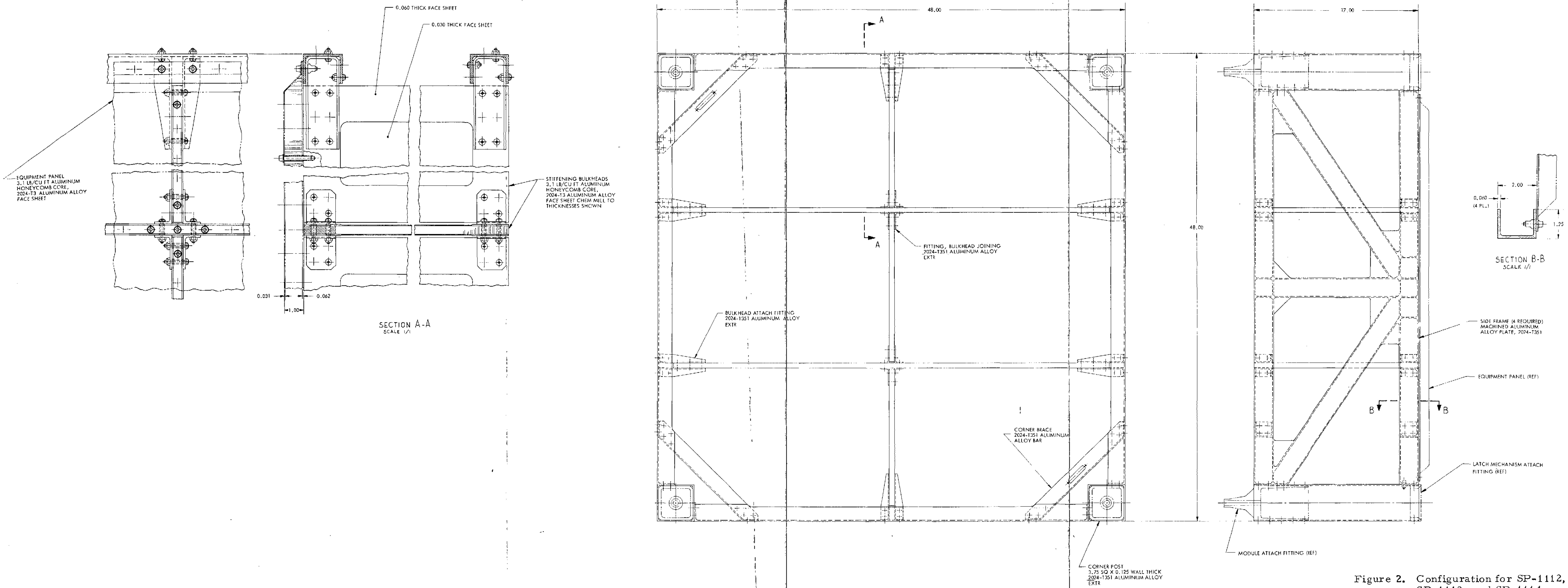

Figure 2. Configuration for SP-1112, SP-1113, and SP-1114 Module Structures

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3.2.2.1 Power distribution. The primary power distribution design shall provide separate buses for the spacecraft and payload to allow individual control and reduce the impact of mission-to-mission changes in the harness. Figure 3 shows an example of the redundant primary power bus protection concept. The primary power buses for the modules and payload shall be controlled by current sensing and magnetically-driven circuit breakers. The circuit breakers for the spacecraft power bus shall be interconnected so that if one circuit breaker opens the other will be commanded to close (whether it is closed or not), in order to ensure that power to the spacecraft is uninterrupted.

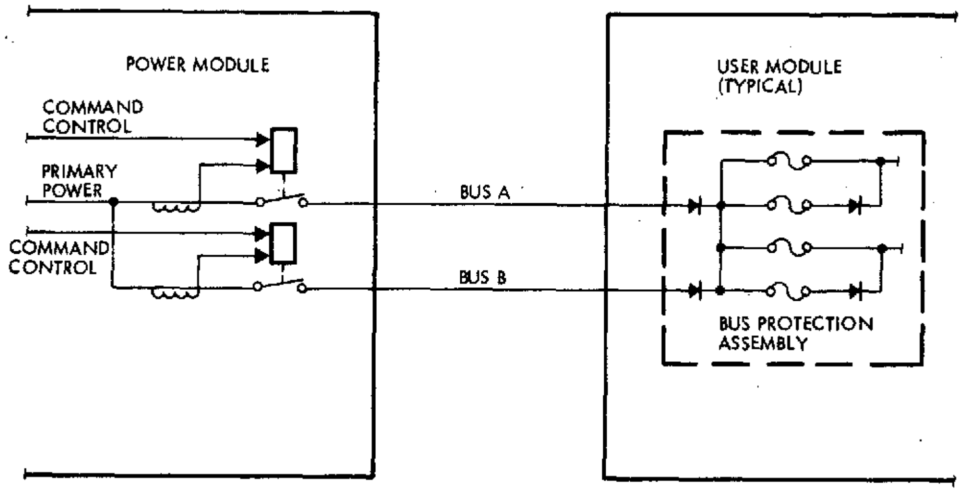

Figure 3. Redundant Primary Power Bus Protection Concept

The module heater power shall be distributed on a bus pair (two lines) to all spacecraft and payload modules. The heater bus shall be fault-isolated in the bus protection assembly of each module.

Structural heaters on the payload and spacecraft structures shall be powered and controlled through the solar array and drive module.

Provisions shall be included for separately energizing the primary power bus, the module heater bus, and the structure heaters from the umbilical connector. Each bus shall be diode-isolated to eliminate the possibility of sneak current paths between the Observatory and ground or Shuttle power sources.

3.2.2.2 Signal distribution. Signal distribution within the Observatory shall be accomplished on the full-duplex data bus interconnecting all modules. A few hardline signals may be required to ensure retrievability of the Observatory, and perform one-time-only functions. The data bus shall provide module-to-module communication, ground-to-module command, and module-to-ground telemetry.

A single wire safe mode bus between all modules shall be used to put the Observatory solar array in a sun-pointing mode in the event of anomalous behavior which could cause battery power depletion. Additional lines for safe mode control shall interconnect the attitude determination and actuation modules for valve control in the safe mode. Solar array drive control for sun-pointing shall be self-contained within the solar array and drive module.

Additional hardlines interconnecting modules are shown in Table 1; umbilical connections are listed in Table 2.

Figure 4 provides the detailed interconnection information for each of the electrical interfaces discussed above and the electrical grounding configuration for a typical module.

3.2.2.3 Observatory harness configuration. The electrical interfaces between the spacecraft harness and the payload harness shall be as identified in ICD 10.10. Figure 5 shows the harness concept.

The umbilical functions required for launch and resupply, as well as electrical interconnections for the spacecraft structure heaters, shall be included in the spacecraft harness section. Interconnection between the spacecraft and payload harness sections shall be made in a J-box located at the transition ring.

Table 1. Hardline Signal Interconnections

From	To	Number of Lines	Purpose
HRPI	Wideband communications	1	High rate (128 Mbit/sec) data
Telemetry	Wideband communications	1	High rate (128 Mbit/sec) data
Telemetry	Wideband	8	LCGS unbuffered data
Wideband communications	HRPI	1	MODS control
Wideband communications	Telemetry	1	MODS control
Solar array and drive	Hard-mounted J-box	14 (7 safe/arm) (7 fire)	Ordnance control and firing
Communication and data handling	Structure	Coax	Aft omni (deployed) RF output

Table 2. Umbilical Interconnections

From	To	Purpose
Communications data handling	Umbilical	Telemetry output
Communications data handling	Umbilical	Computer output
Umbilical	Communications data handling	Shuttle command input
Umbilical	Power	External power
Umbilical	Power	Load disconnect
Umbilical	Hard-mounted J-box	Module heaters
Umbilical	Solar array and drive	Array disconnect
Umbilical	Solar array and drive	Structure heaters

A J-box, located near the power module in the spacecraft structure shall be used for central distribution of the power and signal lines to the modules.

Each harness section shall be divided into power and signal subassemblies to minimize interaction. The ordnance harness for the pyrotechnic circuitry shall be separated from the other harnesses.

3.2.2.4 EMC controls. The electromagnetic compatibility (EMC) design criteria and controls which shall be imposed at the Observatory and/or module levels, as applicable, are summarized in the following paragraphs.

3.2.2.4.1 Electrical systems ground. The primary DC power and party line data bus distribution shall have single-point grounds. Single-point grounding shall be employed for low-level analog sensor circuits and high-current control circuits.

Figure 4 represents the grounding scheme for a typical module.

3.2.2.4.2 Electrical bonding. The electrical bonding configuration of the Observatory shall be designed to provide maximum electrical conductivity across all mechanical joints between metallic members except where DC or thermal isolation is a design requirement. In general, the standardized bonding provisions of MIL-B-5087, "Bonding, Electrical, and Lightning Protection for Aerospace Systems," shall be implemented.

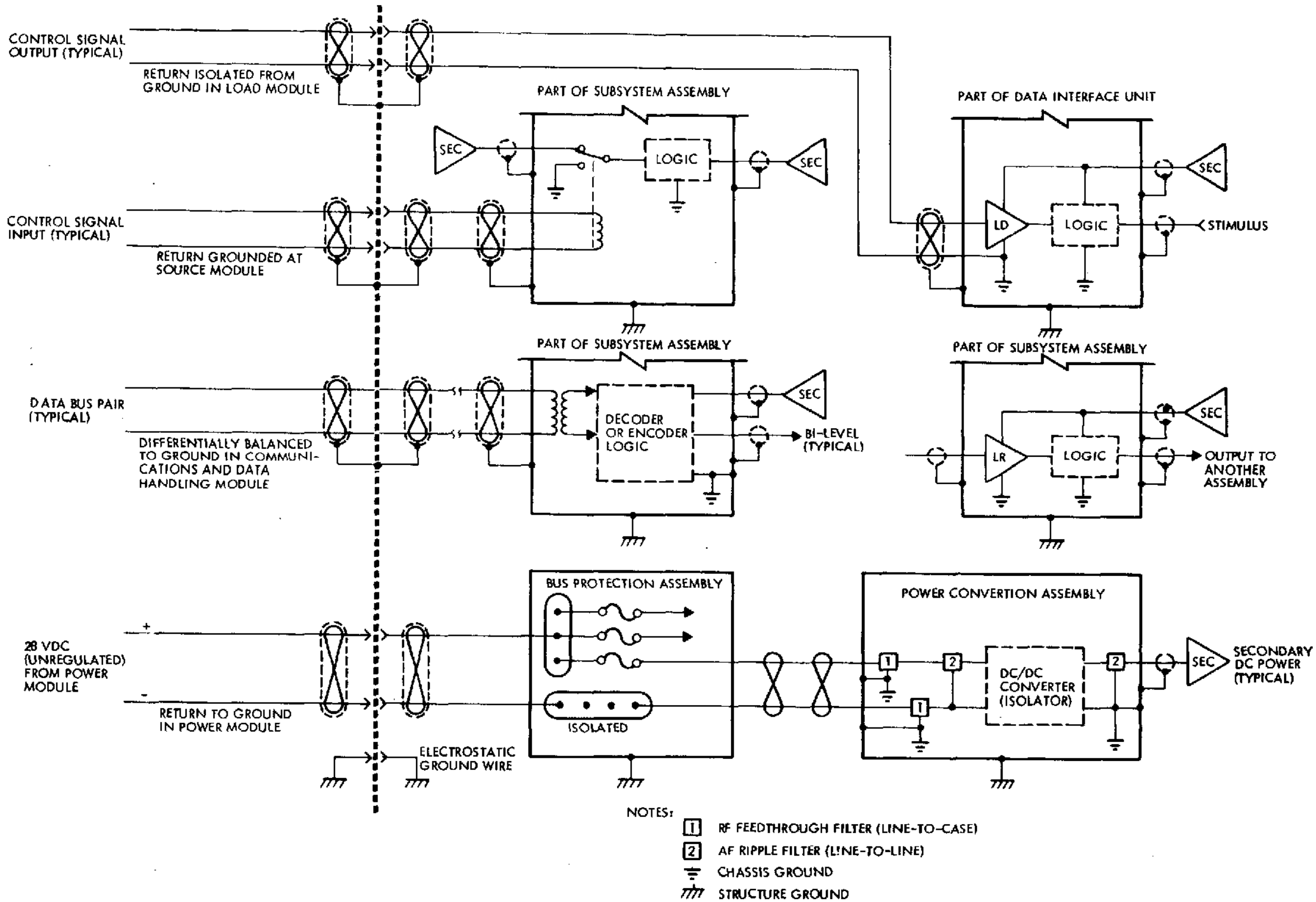

Figure 4. Detailed Module Electrical Interfaces

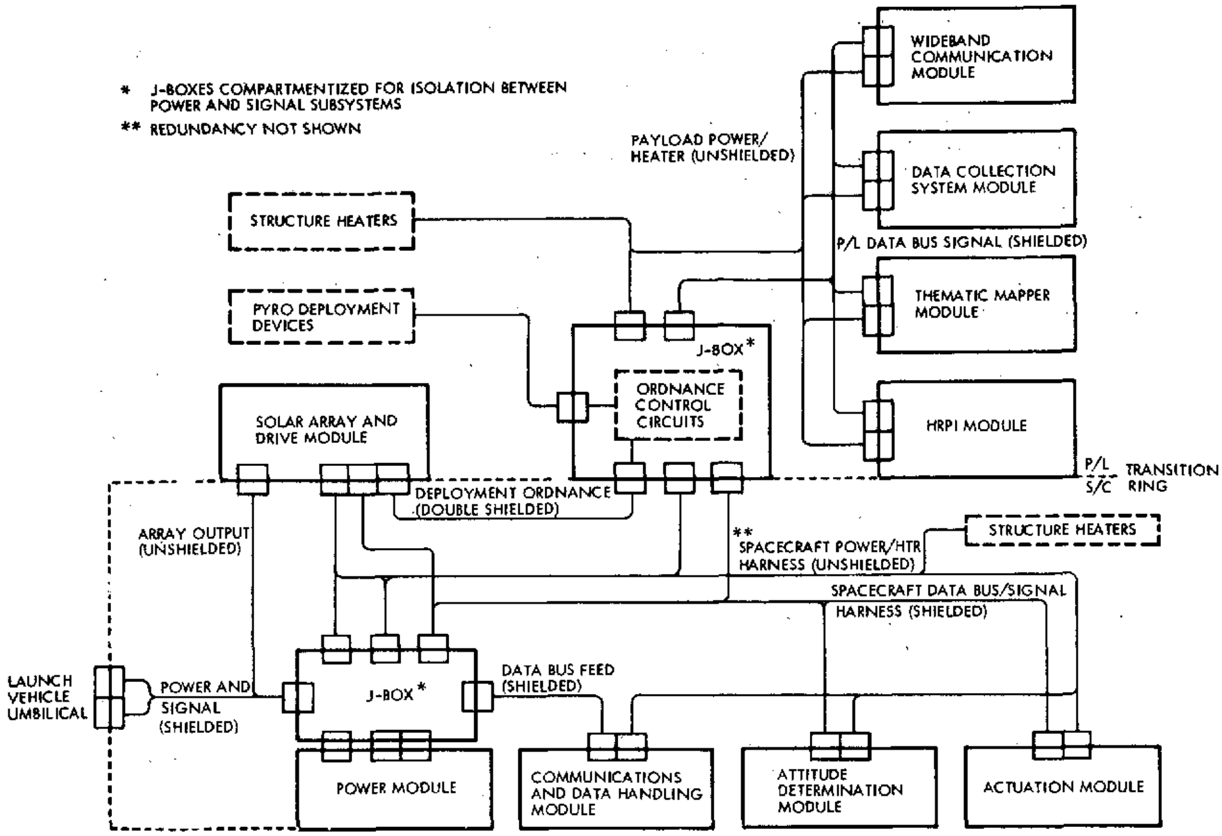

Figure 5. Wire Harness Concept

Also, interfacing metals shall be suitably treated with conductive protective coatings to prohibit the deteriorization of electrical bond joints through electrolytic or galvanic action. The design goal of the electrical bonding configuration shall be to provide a low impedance, electrically continuous, homogeneous ground reference plane at the module level.

3.2.2.4.3 Interconnect wiring/harness. The amplitudes of noise voltages and currents injected into the input terminals of a module by functional operations in another module through wire-to-wire coupling shall be minimized by using shielded wires for all signal and control circuits between components within a module and between modules except for the primary DC power distribution which shall employ unshielded, twisted pairs. Wherever practical, power wiring shall be physically separated from the shielded signal wire harness.

3.2.2.4.4 Unit case shielding. The cases of all units or equipment housing active electrical or electronic circuitry except communications receivers and transmitters shall be designed, using standardized RF gasketing materials and techniques, to provide for a minimum of 40 dB attenuation of RF radiated fields. Communications equipment housing shall be designed for a minimum of 60 dB shielding effectiveness.

3.2.2.4.5 Filtering. EMI filters shall be required at all interfaces from the primary DC power module to user power conversion units. These shall be two-stage filters: 1) an input stage consisting of a high performance feedthrough, bulkhead-mounted filter in each power leg to reduce high frequency regulator switching noise voltages to acceptable levels; and, 2) a line-to line, "L" or double-L audio frequency ripple filter to reduce input line currents at the fundamental and lower order harmonic frequencies of the regulator. The line-to-chassis capacitance of the filter section shall be minimized so as not to unduly degrade the beneficial effects of the wire twisting in the power harness.

3.2.3 Thermal control. The overall EOS thermal control system shall consist of both structure and module thermal designs, each constrained by module/structure interface requirements.

3.2.3.1 Design requirements. General design requirements that shall govern the thermal control system design are shown in Table 3.

Table 3. General EOS Observatory System Thermal Design Requirements and Functions

- Thermally decouple modules/sensors from payload/spacecraft structures to:
  - Allow independent module thermal design
  - Permit module interchangeability
  - Prevent significant module impact on structural thermal distortion
  - Allow on-oribt module replacement
- Provide design flexibility and growth margin to accommodate:
  - A wide range of experiments
  - Various near-earth missions
- Provide for module replacement in orbit to permit module replacement via Shuttle
- Provide thermal control that maintains structural thermal distortion within pointing allocation. (Thermal distortion pointing allocation has an uncertainty component of 30 arc-sec between in-orbit calbration points, and a rate of change allocation not to exceed 0.01 deg/hr.)
- Thermal control elements (coatings, insulation, baffles, dynamiccally-controlled heaters, etc.) satisfy performance requirements for a mission life of 3 years.

3.2.3.2 Structure thermal control design. Structure as defined here refers to the spacecraft structure. The thermal control shall consist of multilayer insulation (MLI) and independently-controlled heaters. Outside layer of MLI shall be aluminized Kapton. Heater-induced thermal transients shall be minimized by reducing the temperature control dead-band or eliminated by having constant power heater circuits. The temperature dead-band shall be reduced by using either computer-control or electronic switching. Module attachment points shall be controlled to $70 \pm 10^{o}F$ for nominal external environments and nominal operating duty cycles.

3.2.3.3 Module/instrument to structure interface. Each module and instrument shall be attached to the spacecraft structure with four fittings, one at each corner of the inboard side, as described in paragraph 3.1.1.2. A design objective shall be restriction of this interface heat flow to a reasonable level of less than 1 watt per attachment point. This goal shall be achieved by controlling structure/module temperatures with heaters and by mechanical design of attachment fittings to provide a reasonably high resistance. Design goal shall be a resistance of > 10 Hr-^{o}F/BTU but spacecraft thermal control system shall be based on a value of 5 Hr-^{o}F/BTU.

3.2.4 Structural interfaces

3.2.4.1 Volume constraints. The Observatory including instruments and modules shall not exceed a diameter of 86 inches in a stowed condition. The length of the observatory, exclusive of the spacecraft adapter, shall not exceed 170 inches.

3.2.4.2 Shroud clearance. The Observatory shall not interfere with the standard Thor-Delta 2910 launch vehicle payload shroud and shall be sufficiently rigid to have adequate clearance with the shroud in its maximum deflected shape under all critical load conditions.

3.2.4.3 Separation ring line load. The spacecraft structure design shall provide for maximum distribution or spread of loads into the Observatory separation ring.

3.2.4.4 Instruments. The spacecraft structure shall be designed to ensure that instruments are not used as spacecraft load-carrying structural members.

3.2.4.5 Orbital drag. The Observatory deployed configuration shall provide a minimum cross-sectional area and orbital drag.

3.2.4.6 Primary structure/attitude control and determination. The mechanical alignment and arrangement between the structure and the attitude control and determination system shall be in accordance with the following drawings:

Control moment gyro installation	TBD
Sun sensor installation	TBD
Primary attitude sensor assembly installation	TBD

3.2.4.7 Primary structure/electrical. The mechanical alignment and arrangement between the structure and electrical systems shall be in accordance with the following drawings:

Electronic equipment installation	TBD
Solar array installation	TBD
Electronic distribution equipment installation	TBD

3.2.4.8 Spacecraft structure/communication and data handling. The mechanical alignment and arrangement between the structure and communication and data handling system shall be in accordance with the following drawings:

Antenna installation	TBD
Electronic equipment installation	TBD

3.2.4.9 Structure/thermal. The thermal interface of the observatory is defined by the following drawings:

Thermal insulation installation	TBD

3.2.4.10 Structure/shroud. The interface between the structure and the shroud shall be in accordance with Drawing TBD.

3.2.4.11 Structure/MGSE. The interface between the structure and the MGSE shall be in accordance with Drawing TBD, (Title, TBD).

3.2.4.12 Weight. Total structure assembly weight shall not exceed TBD.

3.2.4.13 Reliability assessment. The assembly shall have a proability of successfully surviving launch, boost, injection and TBD days in orbit of at least TBD under operating and nonoperating use and environmental conditions as specified herein. The assessed reliability of the assembly shall not be less than TBD.

3.2.4.14 Environmental conditions. The subsystem shall withstand the environmental conditions encountered during assembly, test, storage, checkout, transport and handling, launch, and orbit as simulated in accordance with specification SP-115.

3.2.4.15 Limit load factors. Limit load factors for various flight events are shown in Table 4. Limit load is defined as the maximum anticipated flight load expected for the event shown.

3.2.4.16 Design loads. A yield factor of 1.5 on limit and on ultimate factor of 1.88 on limit shall be used in the design.

3.3 Design and construction

3.3.1 Parts, materials, and processes

3.3.1.1 Selection of parts, materials, and processes. Selection of parts, materials, and processes (PMP) shall be to the requirements identified in MIL-STD-143 Group I. Such selections shall be identified as standard PMP. When PMP selection cannot be made within this requirement, the item is nonstandard and justification must be made in accordance with MIL-STD-749. Control of this action shall be effected by the EOS parts, materials, and processes control board (PMPCB), with the PMPCB approving all selections, both standard and nonstandard, in accordance with EOS-4.1.

Table 4. Maximum Expected Flight Loads*+ (g's)

Flight Event	Shuttle			Thor-Delta 2910		
	X	Y	Z	X	Y	Z
Liftoff	-2.3	±0.3	-0.8	-2.9 +1.0	±2.0	±2.0
High or maximum dynamic pressure	-2.0	±0.5	±0.6			
Booster or Stage I burnout	-3.3	±0.2	-0.4	-12.3 -4.0	±0.65	±0.65
Orbiter or Stage II burnout	-3.3	±0.2	-0.75			
Shuttle space operations	-0.2 0.1	±0.1	±0.1			
Entry and descent	+1.6 -0.25	±1.5	+3.0 -1.0			
Landing and braking	±1.5	±1.5	+2.5			
Crash**	+9 -1.5	±1.5	4.5 -2.0			

+Each triad of X, Y, Z loads is applied simultaneously.

*X, Y, Z refer to Shuttle axes.

**Crash loads are ultimate and used only for satellite support fitting design.

3.3.1.2 Program authorized parts list. All selections of electronic parts shall be identified and authorized by EOS-3.3-1. This list will reflect all PMPCP electronic part selections.

3.3.1.3 Program authorized materials list. All selections of materials shall be identified and authorized by EOS-3.3-2. This list will reflect all PMPCB material selections.

3.3.1.4 Program authorized processes list. All selections of processes shall be identified and authorized by EOS-3.3-3. This list will reflect all PMPCB process decisions.

3.3.1.5 Dissimilar metals. To avoid electrolytic corrosion, dissimilar metals shall not be used in direct contact unless protection against corrosion has been provided in accordance with MIL-E-8983 and MIL-STD-454, Requirement 16.

3.3.1.6 Magnetic materials. Magnetic materials shall be used only if necessary for equipment operation. Those magnetic materials which are used shall have minimum permanent, induced, and transient magnetic fields.

3.3.1.7 Fungus-inert materials. Materials which are fungus-inert, in accordance with MIL-E-8983, and MIL-STD-454, Requirement 4, shall be used.

3.3.1.8 Flammable toxic and unstable materials. Flammable, toxic, and unstable materials shall not be used.

3.3.1.9 Finish. The surface of each component of the subsystem shall be adequately finished to prevent deterioration from exposure to the specified environments that might jeopardize fulfillment of the specified performance. The finish shall also meet the applicable bonding requirements. Thermal properties of the finishes used on the component shall be compatible with the requirements given in detail in applicable equipment specifications. The requirements for finishes identified in MIL-E-8983 shall be implemented.

3.3.1.10 Outgassing. Low outgassing polymeric materials shall be used where sensitive thermal control and other surfaces are in direct line-of-sight and where temperature differences can exist between such surfaces.

3.3.2 Electromagnetic radiation

3.3.2.1 EMC/EMI requirements. The assembly shall be designed for compliance with the radiated emission and susceptibility requirements of NASA/JSC Specification SL-E-0002 as modified or amended by EOS-3,3-4, and EOS-3.3-5. The conducted emission and susceptibility requirements of the aforementioned documents are applicable only at the module to spacecraft structure interface. EMI/EMS levels at interfaces within the module shall be controlled to the extent necessary to ensure self-compatibility of the module subsystem with a safety margin of at least six (6) dB.

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3.3.2.2 Electrical bonding

3.3.2.2.1 Structural bonds. All metallic members of the basic CDH module radiator panel and support structure shall be electrically bonded to each adjacent member to form an electrically continuous, equipotential ground reference plane. The maximum allowable DC impedance across any one structural joint shall be 2.5 milliohms with an overall design goal of 10 milliohms, or less, between any two diametrically opposite points on the module. The general methods of MIL-B-5087, Class R, may be used for implementation of these requirements.

3.3.2.2.2 Equipment mounting pads. Surfaces on the module radiator panel and structure which are intendned for unit, equipment, or component mounting shall be free of paint, anodize, or other nonconductive finishes. The maximum DC impedance between the baseplate of any electrically active unit, equipment or component, and the module radiator panel shall be 2.5 milliohms.

3.3.2.2.3 Electrical connectors. All interface electrical connectors, both plug and recptical, which form a part of the unit and cable RF shielding system within the module shall have electrically conductive body shells, free of nonconductive finishes. They shall also have provisions for terminating (grounding) the shields on the interconnecting electrical harness. The maximum DC impedance between the shield termination point and the baseplate of the parent unit shall be 2.5 milliohms.

3.3.2.2.4 Electrostatic bonds. Electrically passive components or appurtenant structures which are attached to the basic module structure through thermal isolators shall be provided with electrostatic bonds having a DC impedance of 1.0 ohms, or less. The general methods of MIL-B-5087, Class S, may be used for implementation of this requirement.

3.3.3. Nameplates and product marking. Each unit shall be identified in accordance with MIL-STD-130 as implemented by EOS-3.3-6. Where practical, a minimum character height of 0.9 inch shall be used. Marking shall include the manufacturer's name or initials, assembly part number and revision status, assembly serial number, contract number, and others as delineated in the component equipment specifications.

3.3.4 Workmanship. The assembly and its component equipment shall be constructed, finished, and assembled in accordance with MIL-STD-454. Requirement 9 and the specifications and drawings specified herein.

3.3.5 Interchangeability and replaceability. The assembly shall be designed to permit removal and replacement of components with a minimum of disturbance of associate or adjacent equipment. Design for service and access shall conform to the principles and requirements of MIL-STD-470.

3.3.6 Safety. The assembly shall be designed to meet or exceed the requirements of EOS-3.3-7, as implemented by EOS-4-1. The design criteria shall include but shall not be limited to those set forth in MIL-STD-882.

3.3.7 Human performance/human engineering. MIL-STD-1472A shall be used for the design of man/machine interfaces.

3.4 Documentation. Documentation shall be as specified in EOS-3.3-8.

4. QUALITY ASSURANCE PROVISIONS

4.1 Responsiblity for inspection and test. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified herein. Except as otherwise specified, the supplier may utilize his own facilities or any commercial laboratory acceptable to the government. The government reserves the right to perform any of the inspections set forth in the specification where such inspections are deemed necessary to ensure that supplies and services conform to prescribed requirements.

4.1.1 Quality assurance and reliability plans. The assembly shall be produced in accordance with applicable portions of EOS-4.1.

4.2 Classification of inspection and test. The examination and testing of the assembly shall be classified as follows:

(a). Qualification inspection and test (See 4.3)

(b) Acceptance inspection and test (See 4.4).

4.3 Qualification inspection and test. Qualification inspection and test shall consist of the following examinations and tests:

(a) Examination of product in accordance with 4.6.1

(b) Functional test in accordance with 4.6.2

(c) Qualification environmental test in accordance with 4.6.3.

4.3.1 Inspection sample. The inspection sample shall be the (TBD) serialized item produced in accordance with this specification.

4.3.2 Inspection sequence. The inspection sequence shall be governed by the following:

- Examination of the assembly shall be performed prior to function testing.

- Functional tests shall be performed prior to, during, where appropriate, and following environmental testing. The functional testing conducted at the conclusion of an environmental test may serve as the functional testing to be performed prior to the next environmental test.

- Environmental testing may be performed in any sequence.

4.3.3 Failure criteria. The assembly shall exhibit no failure, malfunction or out-of-tolerance performance degradation as a result of the examinations and tests specified herein. Any such failure, malfunction or out-of-tolerance performance degradation shall be cause for rejection.

4.4 Inspection conditions. Unless otherwise specified in 4.6, the conditions under which all inspections are accomplished shall be specifed below:

4.4.1 Atmospheric and environmental. All examinations and tests shall be conducted under the local prevailing pressure at the time of test, at a temperature of 75 ± 5°F and at a relative humidity of 55 percent or less.

4.4.1.1 Deviations of atmospheric conditions. When tests are performed with atmospheric conditions substantially different from the specified values, proper allowances for changes in instrument readings shall be made to compensate for the deviation from the specified conditions.

4.4.2 Equipment warmup time. The equipment warm-up time shall be less than 1 minute.

4.4.3 Temperature stabilization. Temperature stabilization shall have been achieved when temperature of the equipment mounting structure varies not more than 5°F during a period of 15 minutes.

4.4.4 Measurements. Measuring instruments used to determine functional parameter values (such as voltage, frequency, current, flow, pressure, etc.) shall indicate true values with an accuracy determined by the tolerance allowed for the parmeter variation itself, such that the measuring instrument shall not introduce an uncertainty greater than 10 percent of the allowable variation of the measured parameter. However, no such measurement accuracy shall be required to exceed 0.5 percent of the required value of the parameter unless otherwise specified.

Except as specifically noted in the inspection methods tolerances for environmental test conditions shall be defined as follows:

Temperature: ±5°F

Barometric pressure; ±10 percent

Relative humidity: +5 percent, -10 percent

Time: ± 5 percent

Vibration amplitude (sine): ±10 percent

Vibration frequency: ±2 percent

4.5 Inspection methods

4.5.1 Physical inspection of the product. Each assembly shall be visually examined for workmanship, identification, finish and conformance to Drawing Number (TBD).

4.5.2 Classification of inspections and tests. The examination and testing of the subsystem shall be classified as follows:

(a) Qualification

(b) Acceptance

4.5.3 Qualification tests. Assembly qualification is achieved by testing the assembly as an integral part of EOS observatory system level tests.

4.5.3.1 Modal survey test. TBD.

4.5.3.2 Acoustic test. TBD.

4.5.3.3 Static loads test. TBD.

4.5.3.4 Pyrotechnic shock test. After exposure to the qualification acoustic test the solar array pyrotechnic release system shall be actuated. This test serves as a functional verification of the ordnance release system and the solar array deployment system. The pyrotechnic test verifies the capability of the solar cells and local structure to withstand the pyrotechnic environment.

4.6 EMI tests. Design qualification tests shall be performed on the fully assembled assembly to verify compliance with the applicable electromagnetic inte ference and susceptibility control requirements. The general test me ıods of MIL-STD-462, as modified or amended by EOS-3.3-5, shall be used during this test program except that, wherever practical, spectrum analyzers or other forms of rapid display and analysis equipment may be used in lieu of equipment requiring manual tuning and data collection.

5. PREPARATION FOR DELIVERY

5.1 General. The assembly as specified herein shall be protected from degradation by environments anticipated during shipment, handling, and storage. Standard commercial packaging practices are acceptable provided they fulfill these requirements.

5.2 Preservation and packaging

5.2.1 Cleaning. The assembly shall be clean and free of contaminants which might impair its use prior to packaging.

5.2.2 Attaching parts. When attaching parts such as nuts, bolts, washers, etc., accompanying the assembly they shall be preserved, bagged, appropriately identified, and attached to, or adjacent to, the fitting for which they are intended.

5.2.3 Electrical connectors. All electrical connectors shall be capped with protected dust caps. Caps used shall be a friction fitting or threaded type which do not require tape or a mechanical device to secure.

5.2.4 Critical surfaces. External machined surfaces and mounting surfaces of the assembly shall be protected with protective pads. Materials used for pads shall not cause item deterioration.

5.2.5 Wrapping. The assembly shall be wrapped using antistatic polyethylene film.

5.2.6 Cushioning. When required for protection, the assembly shall be cushioned using a suitable resilient foam cushioning material such as polyethylene, polyurethane, or polystyrene.

5.3 Packing. The assembly shall be packed in a suitable shipping container designed for one item only. The shipping container used shall provide protection to the item against corrosion, deterioration, and damage during shipment from the source of supply to the receiving activity. Containers shall comply with applicable tariffs and regulations for particular modes of transportation when so shipped.

5.3.1 Storage conditions. The assembly shall not be adversely affected by storage within its container at temperatures between 60 and 90°F and relative humidities of 60 percent or less.

5.3.2 Shipping conditions. The assembly shall be capable of withstanding the following environments:

Temperature: +160°F in an unsheltered area (125°F + 35°F due to solar radiation) and -40°F in accordance with restricted air transport criteria

Humidity: Up to 100 percent in an unsheltered area

Rough Handling: Capable of withstanding and physically protecting the unit from rough handling during shipping by common carrier.

5.4 Marking for shipment. Each assembly and shipping container shall be marked with the following:

(a) Item nomenclature

(b) Part number

(c) Contract or purchase order number

(d) Manufacturer's name

(c) Manufacturer's part number and serial number (on items container only)

(f) Quantity

(g) Date of manufacture (on item container only)

(h) Fragile - Handle with Care (when applicable)

(i) Space Vehicle Material - Do Not Open in Receiving Inspection (shen applicable - shipping container only)

(j) Actual weight.

5.4.1 Documentation. All required reliability and test documentation such as test reports, certifications, shipping invoices, etc., shall be either packed in the CDH module container or attached to the exterior surface of the shipping container. Attachment shall be such a manner as to preclude loss of this data during handling and shipment by common carrier.

TITLE

PRIME ITEM DEVELOPMENT SPECIFICATION
FOR THE

COMMUNICATIONS AND DATA HANDLING MODULE

DATE 20 SEPT 1974 NO. SP-1112

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

ONE SPACE PARK • REDONDO BEACH, CALIFORNIA 90278

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SPECIFICATION SP-1112

COMMUNICATION AND DATA HANDLING MODULE

1. SCOPE

1.1 Scope. This document establishes the requirements for the performance, design, test, and qualification of a communications and data handling (CDH) module for the Earth Observatory Spacecraft. The CDH module provides the spacecraft with a ground-to-space and space-to-ground communications and onboard processing capability.

2. APPLICABLE DOCUMENTS

2.1 Government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

SPECIFICATIONS

NASA

SP-11	Observatory Segment Specification
SP-115	EOS Environmental Specification
SP-1111	Structure Assembly Specification
SP-1113	Electric Power Specification

Military

MIL-B-5087	Bonding, Electrical, and Lightning Protection for Aerospace Systems
MIL-E-8983A	Electronic Equipment, Aerospace Extended Space Environment, General Requirements
MIL-Q-9858A	Quality Program Requirements

STANDARDS

Military

MIL-STD-143B	Standard and Specifications, Order of Precedence for Selection
MIL-STD-454C	Standard General Requirement for Electronic Equipment

MIL-STD-749B	Preparation and Submissions of Data for Approval of Nonstandard Electronic Parts
MIL-STD-1472A	Human Engineering Design Criteria for Military Systems, Equipment, and Facilities
MIL-STD-882 15 July 1969	System Safety Program for Systems and Associated Subsystems and Equipment, Requirements for
MIL-STD-462	EMI/EMS Test Methods
MIL-STD-130D	Identification Marking of U.S. Military Property
MIL-STD-470	
MIL-STD-470	Maintainability Program Requirements (for Systems and Requirements)

OTHER PUBLICATIONS

NASA

SL-E-0002	Electromagnetic Compatibility Control Plan
NASA STDN101.1	STDN User's Guide
X-560-63-2	Aerospace Data Systems Standards
- - - -	Electromagnetic Compatibility Requirements for Space Systems
EOS-3.3-1	Program Authorized Parts List
EOS-3.3-2	Program Authorized Materials List
EOS-3.3-3	Program Authorized Processes List
EOS-3.3-4	Electromagnetic Compatibility Control Plan
EOS-3.3-5	EMI/EMS Limits and Test Methods
EOS-3.3-6	Marking of Parts and Assemblies

2.2 Non-Government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. If no revision or date is shown, the latest released issue of the applicable document shall apply. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

DRAWINGS

10.5	CDH Module Wire List
10.4	Satellite Telemetry Allocation, ICD
10.3	Satellite Command Allocation, ICD
10.2	Satellite Primary Power Allocation, ICD
10.1	CDH Module Envelope, ICD
TBS	CDH Module Assembly Drawing

OTHER PUBLICATIONS

EOS-3.3-7	Safety Design Criteria
EOS-3.3-8	Configuration Management Plan
EOS-4.1	System Effectiveness Program Plan
EOS-4.2	Integrated Test Plan
EOS-4.3	Component/Module Development Test Plan

3. REQUIREMENTS

3.1 General

3.1.1 Function. The communications and data handling (CDH) module contains the equipment required to receive information from and transmit information to, NASA ground station. It provides the spacecraft with the capability to receive, demodulate, and process uplink command information; collect, process, and transmit housekeeping telemetry and medium-rate user data; coherently transpond range information; and centrally perform on-board computations. Finally, it contains the equipment required to implement and control the spacecraft data bus system. All communication shall be compatible with GSFC document X-560-63-2.

3.1.2 Operation. The CDH module communications system will be composed of the following S-band channels: command, telemetry, and ranging (Figure 1). The command channel RF equipment will receive, and demodulate to baseband, the command information transmitted on a phase-modulated carrier in the 2050 to 2150 MHz range. The baseband output from the communications equipment will be provided as an input to the data handling equipment. The telemetry channel RF equipment will receive from the data handling equipment, a composite signal consisting of 32 kbit/sec telemetry data biphase modulated onto a 1.024 MHz subcarrier, and a signal consisting of either 512 kbit/sec direct digital data or

500 kHz range data. The telemetry channel RF equipment will transmit this data as a phase-modulated carrier, in the 2200 to 2300 MHz range, on either right or left hand antenna circular polarization.

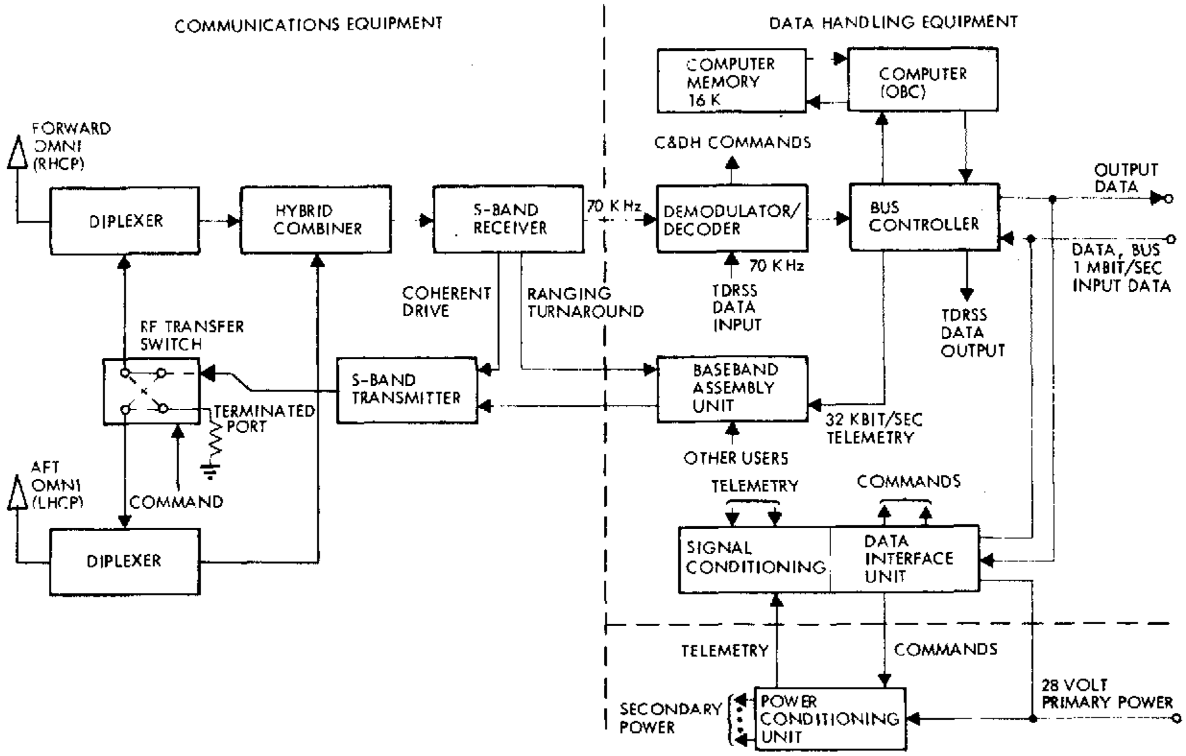


Figure 1. Communications and Data Handling Module Block Diagram

The data handling equipment shall provide detection and decoding of baseband command information; on-board command storage; control of the on-board data bus; dissemination of real-time or stored commands via the data bus; on-board computations; and assembly of 64 kbit/sec housekeeping data, 512 kbit/sec medium-rate user data, and range data for transmission to the communications equipment.

Secondary DC power for the CDH module will be provided from a central power conditioning unit and module command and telemetry data will be provided from a data interface unit that interfaces with the spacecraft data bus.

3.1.3 Government-furnished module equipment. Certain items of module equipment which are identical or nearly identical in function will be fabricated by the integration contractor and provided as GFE to the remaining module contractors. This equipment shall include:

Item	Reference
Data interface unit	3.2.1.3.4 of SP-1113
Power control unit	3.2.1.9 of SP-1113

All module structures and integration of the modules will be provided by the integrating contractor.

3.1.4 Equipment list. The minimum and nominal redundancy configurations for the CDH module shall consist of the following equipments (see Section 6.1 for definition):

Component Name	Quantity per Module: Minimum Redundancy	Quantity per Module: Nominal Redundancy
Communications Equipment Group		
S-band RHCP omnidirectional antenna	1	1
S-band LHCP omnidirectional antenna	1	1
S-band diplexer	2	2
Hybrid combiner	1	1
S-band receiver	1	2
S-band transmitter	1	2
S-band transfer switch	1	1
Data Handling Equipment Group		
Demodulator/decoder	1	2
Bus controller unit	1	2
On-board computer	1	2
Computer memory module	1	2
Baseband assembly unit	1	2
Signal conditioning and data interface unit	1	2
Power conditioning unit	1	2
Harness	1	1
Thermal control (heaters)	1	1
Module structure	1	1

3.2 Characteristics

3.2.1 Performance

3.2.1.1 Electrical interface requirements. The module electrical interface lines are illustrated in Figure 2.

3.2.1.1.1 RF interface. The RF links shall provide two-way communications between the spacecraft and the NASA Satellite Tracking and Data Network (STDN). All communications shall be compatible with GSFC document X-560-63-2. Signals transmitted from the STDN shall consist of command and ranging information. Signals transmitted from the spacecraft shall consist of user and telemetry data and transponded range information. Range rate information shall be provided by coherently receiving and tracking the uplink carrier and providing it as a coherent reference which, when suitably multiplied in frequency, will comprise the downlink carrier.

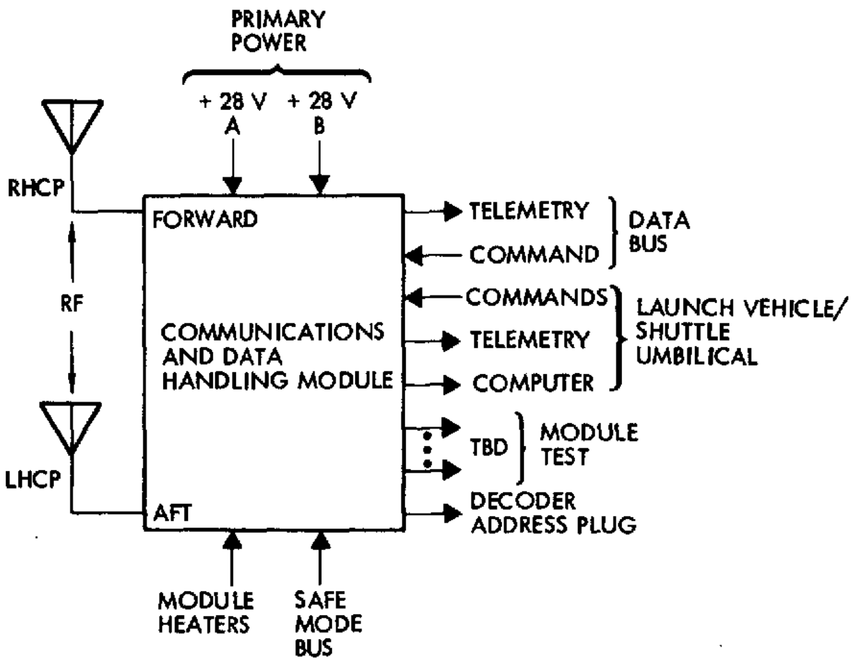


Figure 2. Communications and Data Handling Module — Electrical Interfaces

3.2.1.1.1.1 S-band uplink. The S-band uplink characteristics shall be as follows:

Receive carrier range:	2050 to 2150 MHz
Receive frequency:	TBD MHz
RF signal polarization:	RHCP and LHCP

Receive signal level:	-130.0 dBm and above
Transponder turnaround ratio:	221/240
Ranging signal tone frequency:	500 Hz, maximum
Command data rate:	2,000 bit/sec
Command modulation:	PCM/PSK/FM/PM
Command subcarrier frequency:	70 kHz
Command subcarrier modulation:	Split-phase-M, PCM
Command format:	as defined in Figure 3
Command preamble:	> 200 bits
Probability of false command execution:	$< 1 \times 10^{-10}$
Probability of good command rejection:	$< 1 \times 10^{-3}$ over range from -105 to -40 dBm

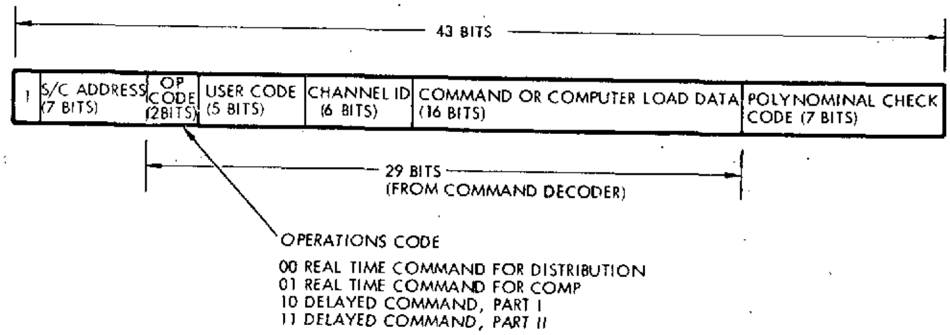


Figure 3. Uplink Command Format. (The first transmission is processed by a preamble of at least 200 bits.)

3.2.1.1.1.2 S-band downlink. The S-band downlink characteristics shall be as follows:

Transmit carrier range:	2200 to 2300 MHz
Transmit frequency:	TBD MHz
Transmit carrier stability (open loop):	One part in 10^5

Narrowband data rate:	Selectable: 64 kbit/sec, 32 kbit/sec, 16 kbit/sec, 8 kbit/sec, 4 kbit/sec, 2 kbit/sec, 1 kbit/sec
Narrowband modulation:	Split-phase, PCM/PSK
Narrowband subcarrier frequency:	1.024 MHz
Medium-band data rate:	512 kbit/sec maximum
Medium-band data modulation:	Split phase, PCM/PM
Transmitter RF output power:	2 watts
Telemetry format:	Compatible with the requirements of STDN 101.1.

3.2.1.1.2 Data bus interface. A 4-wire, full duplex, party line data bus shall be used for intermodule data transfer. One pair of wires shall be used for data from the CDH modules (supervisory line), the other pair shall be used for data to the CDH module (reply line). Characteristics of the data bus signals shall be as follows:

Bit rate:	1.024 Mbit/sec
Data code:	Manchester - M
Word synchronization:	3 bits illegal Manchester followed by a logical 1
Word format:	As illustrated in Figure 5
Bus time slots:	As illustrated in Figure 6
Clock:	Transmitted on supervisory line with continuous data stream (all "0"s between commands or data requests)
Drive level:	Up to 32 remote units may be tied to one bus
Coupling:	A-C (all signals are coupled with transformers or capacitors).

3.2.1.1.3 Primary power. The CDH module shall receive primary power from the power module on two redundant lines as follows:

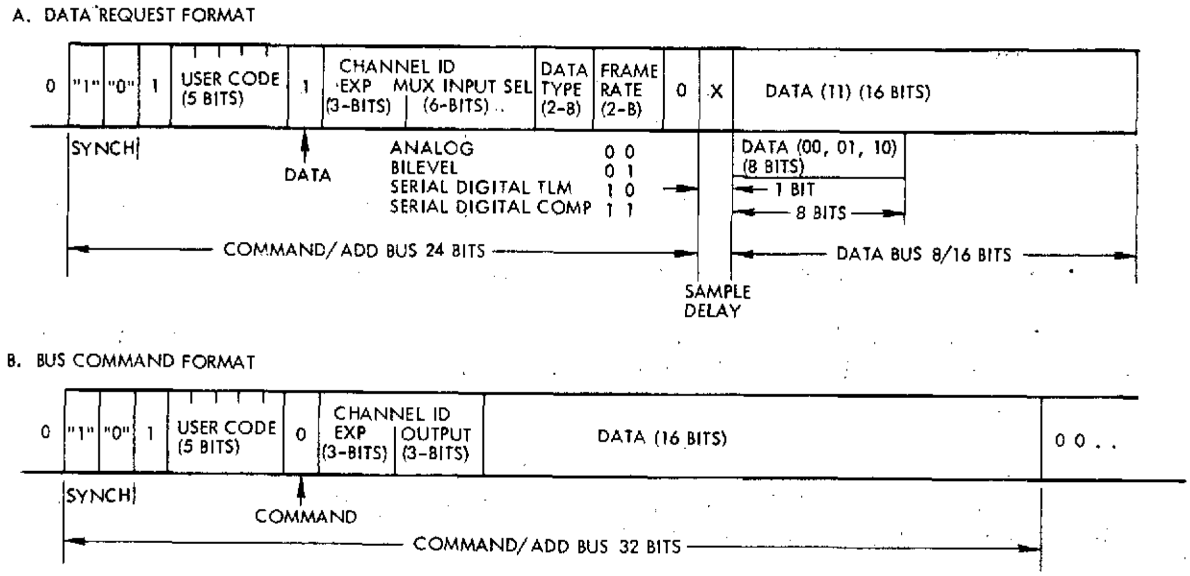
Voltage: 28 ±7 volts

Current: 3 amperes maximum

Transients

Load switching: ±1 volt (100 ms or less)

Fault correction: Down to +20 VDC or up to +39 VDC for 100 ms or less



NOTES:
USER CODE (5 BITS) IDENTIFIES THE DIU OR MODULE
CHANNEL ID IDENTIFIES THE DATA REQUEST CHANNEL WITHIN THE DIU

Figure 5. Data Bus Word Format

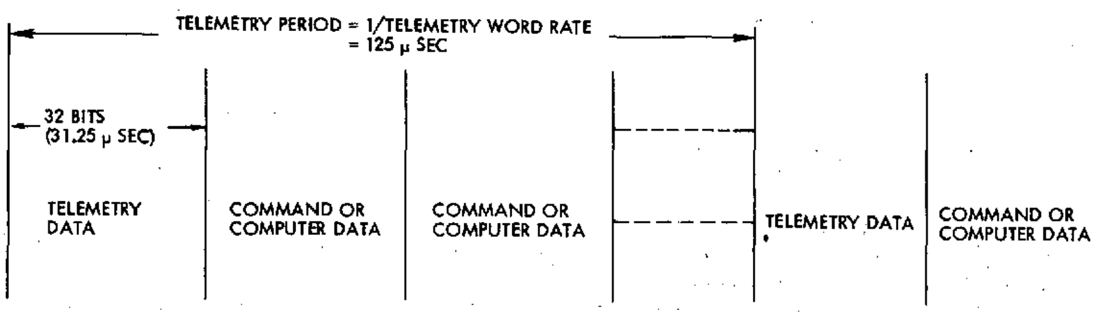


Figure 6. Data Bus Time Slots. (64 kbit/sec Data.)

3.2.1.1.4 Module heaters. A connection to heater power shall be provided. The line shall operate at 28 volt nominal with TBD amperes maximum current.

3.2.1.1.5 Safe mode bus. A connection to the safe mode bus shall be provided. Normal mode shall be indicated by a +5 volt level (10 ma maximum module sink current). Safe mode shall be indicated by a 0 volt level.

3.2.1.1.6 Launch vehicle/Shuttle umbilical. Buffered lines shall be provided for the umbilical interface for use as follows:

Command input	- Serial NRZ data stream, 2 kbit/sec (bypasses demodulator portion of the demodulator decoder)
Telemetry output	- Serial NRZ data stream, 1 kbit/sec to 64 kbit/sec (output from the bus controller)
Computer output	- 16- or 18-bit parallel words

3.2.1.1.7 Module test connector. A test connector shall be provided for module test of the following signals:

TBD

3.2.1.2 Communications group. The communications equipment group (Figure 1) contains the following equipment:

(a) S-band RHCP omnidirectional antenna

(b) S-band LHCP omnidirectional antenna

(c) S-band diplexer

(d) Hybrid combiner

(e) S-band receiver

(f) S-band transmitter

(g) S-band transfer switch

3.2.1.2.1 S-band omnidirectional antennas. Two omnidirectional antennas having opposite sense circular polarization shall be employed to provide 25 percent spherical coverage for the S-band receive and transmit frequency ranges. Each of the omnidirectional antennas shall operate in the uplink frequency band of 2050 to 2150 MHz and downlink frequency band of 2200 to 2300 MHz. Both antennas shall be of the conical log spiral type with electrical design parameters and polarization as defined below.

3.2.1.2.1.1 RHCP omni-antenna

(a) Polarization. An omnidirectional antenna shall be mounted on the forward or earth-pointing side of the CDH module. Its electrical polarization shall be right-hand circular polarization (RHCP). The rotating electrical field vector shall be clockwise for an observer looking in the direction of propagation.

(b) Pattern requirement. With reference to RHCP, the antenna shall have a minimum gain of -1.0 dBi over 220 degrees centered on axis.

(c) Passband VSWR. The input impedance of the RHCP antenna shall present a voltage standing wave ratio (VSWR) of 2.0 to 1.0 or less when referenced to 50 ohms resistive over the specified frequency band.

(d) Axial ratio. The axial ratio of the RHCP antenna shall not exceed 3.0 dB over the halfpower beamwidth.

3.2.1.2.1.2 LHCP omni-antenna

(a) Polarization. A left-hand circular polarized (LHCP) omnidirectional antenna shall be provided as part of the CDH module equipment. This antenna, while not integral to the CDH module itself, shall be connected to the module by means of suitable RF connectors and coaxial cabling for boom mounting and deployment on the aft (anti-earth) side of the spacecraft. The rotating electrical field vector shall be counterclockwise for an observer looking in the direction of propagation.

(b) Pattern requirement. With reference to LHCP, the antenna shall have a minimum gain of -1.0 dBi. Additionally, the 3 dB beamwidth shall be a minimum of 210 degrees centered on axis.

(c) Passband VSWR. The input impedance of the LHCP antenna shall present a wave ratio VSWR of 2.0 to 1.0 or less when referenced to 50 ohms resistive over the specified frequency band.

(d) Axial ratio. The axial ratio of the LHCP antenna shall not exceed 4 dB in the hemisphere about its positive RF axis.

3.2.1.2.2 S-band diplexer. Two S-band diplexers shall be provided to allow the omni-antennas to supply received signals to the S-band command receiver. Each diplexer shall be a three terminal network consisting of a transmit filter, receive filter, and a coupling network.

3.2.1.2.2.1 Passband. The filters located within the transmit channel of each diplexer shall provide a 10 MHz minimum passband centered about the transmitter frequency for rejection of spurious frequencies. The filters shall be tunable over the transmitter frequency range.

The filters in the receive channel shall provide isolation between the transmitter and receiver for their simultaneous operation as well as image rejection. The filters shall be tunable over the range from 2050 to 2150 MHz with a matched bandwidth of 10 MHz about the S-band transmitter frequency.

3.2.1.2.2.2 Insertion loss. The insertion loss between the diplexer antenna input port and the receiver output port shall not exceed 1.0 dB. The insertion loss between the diplexer antenna output port and the transmitter input port shall not exceed 0.5 dB.

3.2.1.2.2.3 <u>Passband VSWR</u>. The passband VSWR, with respect to a resistive impedance of 50 ohms, shall not be greater than 1.2 to 1.0 at each terminal when all other terminals are terminated in a matched load.

3.2.1.2.2.4 <u>Isolation.</u> The isolation between the transmitter and receiver terminals shall not be less than 100 dB over the frequency range of the receive frequency and not less than 100 dB over the frequency range of 2200 to 2300 MHz. The isolation between receiver inputs, in either direction, shall not be less than 55 dB over the frequency range of 2050 to 2150 MHz.

3.2.1.2.3 <u>Hybrid combiner</u>. A hybrid combining network shall be provided to allow either or both omnidirectional antennas and their associated diplexers to supply received signals to the S-band command receiver. The hybrid network shall be a four port network with one port terminated into a 50 ohm external load. The hybrid network shall operate over the 2050 to 2150 MHz range.

3.2.1.2.3.1 <u>Insertion loss.</u> The insertion loss between either input port and the unterminated output port shall not exceed 3.3 dB, including the hybrid coupling loss.

3.2.1.2.3.2 <u>VSWR</u>. The voltage standing wave ratio, with respect to a resistive impedance of 50 ohms, shall not be greater than 1.2 to 1.0 at each terminal when all other terminals are terminated in a matched load.

3.2.1.2.3.3 <u>Isolation.</u> The isolation between each input port shall not be less than 25 dB when all other terminals are terminated in a matched load.

3.2.1.2.4 <u>S-band command receiver</u>. An S-band phase-lock carrier tracking receiver shall be provided to demodulate to baseband the received command signal and ranging signal.

3.2.1.2.4.1 <u>RF input frequency</u>. The carrier frequency of the S-band receiver shall be in the range of 2050 to 2150 MHz. The RF input spectral occupancy with full phase modulation shall be ±1.5 MHz about the carrier frequency.

3.2.1.2.4.2 <u>RF outputs.</u> The receiver shall supply to the transmitter an RF signal coherent with the received carrier. The frequency of the RF output signal shall be 10/221 of the received carrier frequency. The RF signal level into a 50 ohm load shall be 0 dBm or greater. Spurious outputs shall be 30 dB or more below the 0 dBm output signal level.

3.2.1.2.4.3 <u>Wideband demodulated output</u>. The receiver shall provide a wideband output consisting of the demodulated ranging and command subcarrier signals and shall have the following characteristics:

(a) <u>Frequency response</u>. With a -40 dBm RF input signal, the amplitude of the output shall be down less than 3 dB at 2.2 kHz and

2.0 MHz when referenced to 100 kHz. The output response between 5 kHz and 100 kHz shall not exceed +2 dB when referenced to 100 kHz.

(b) Amplitude. For a 0.2 radian phase modulation index at 100 kHz and an RF input signal of -40 dBm, the output amplitude shall be 0.2 ±0.032 volt, peak-to-peak at ambient conditions. The output amplitude change caused by exposure to the operating thermal range shall not exceed +5 percent, -12 percent of the ambient level obtained at -40 dBm RF input level and 100 kHz modulation frequency. The output amplitude shall not vary more than 2 dB for RF input signal levels between -15 dBm and -85 dBm. The load on the wideband output shall not be less than 750 ohms with a parallel capacitance not to exceed 390 pf.

(c) Output squelch. The outputs shall be squelched when the receiver is not locked, and shall be capable of functioning when cross-strapped to an identical powered receiver.

3.2.1.2.4.4 Received signal strength indication. The receiver shall provide as an output, a signal whose amplitude varies in direct proportion to the received RF carrier strength.

3.2.1.2.4.5 Coherent/noncoherent control. The receiver shall provide a bilevel output to the transmitter to indicate when phase lock has been achieved.

3.2.1.2.4.6 Noise figure. The average noise figure of the S-band receiver shall not exceed 10.0 dB including the contribution from the preselection filter.

3.2.1.2.4.7 Tracking threshold. In the presence of a swept RF carrier, the receiver shall be capable of acquiring the carrier within ±180 kHz of the assigned frequency.

3.2.1.2.4.8 Sweep period. The sweep period shall be 10 seconds.

3.2.1.2.4.9 Dynamic input range. The receiver performance shall satisfy all the specified requirements when the RF input signal is within the range from threshold to -30 dBm.

3.2.1.2.4.10 Loop noise bandwidth. The receiver shall have a loop noise bandwidth of 800 Hz at RF threshold conditions. Threshold conditions are defined as a ±6 dB signal-to-noise ratio (SNR) in the phase-lock loop noise bandwidth.

3.2.1.2.4.11 Loop damping ratio. The receiver carrier tracking phase-lock loop shall have a damping ratio of 0.7 ±0.3 at threshold.

3.2.1.2.4.12 Loop natural resonant frequency. The receiver carrier tracking phase-lock loop shall have a natural resonant frequency of 700 radians/second at threshold.

3.2.1.2.4.13 TDRSS IF output. An IF output shall be provided from the S-band receiver for the purpose of driving a TDRSS spread

3.2.1.2.4.13 Passband VSWR. The passband VSWR, with respect to a resistive impedance of 50 ohms, shall not be greater than 1.2 to 1.0 at each terminal when all other terminals are terminated in a matched load.

3.2.1.2.4.14 TDRSS IF input. An IF input shall be provided in the S-band receiver for the purpose of accepting a despread carrier signal from the TDRSS input and shall be demodulated when one is present onboard. This signal shall be selected by means of a command. The IF input shall be provided as an input to the receiver carrier tracking loop. It shall have a level of at least -110 dBm.

3.2.1.2.4 Hybrid combiner. A hybrid combining network shall be provided to allow either of both omnidirectional antennas and their associated diplexers to supply received signals to the S-band command receiver. The isolation between receiver inputs, in either direction, shall not be less than 55 dB over the frequency range of 2050 to 2150 MHz.

3.2.1.2.5 S-band telemetry transmitter. An S-band transmitter shall be provided for transmission of telemetry, medium-rate user data, and other information from either of both operational modes and its associated equipment, coherent RF frequency received from the S-band command receiver. The hybrid network shall be a four port network with one port terminated into a 50 ohm external load. The hybrid network shall operate over the 2050 to 2150 MHz range.

3.2.1.2.5.1 RF input signal characteristics. The transmitter shall be capable of being driven by an output reference signal from the S-band receiver which has the following characteristics:

3.2.1.3.1 Insertion loss. The insertion loss between either input port and the unterminated output port shall not exceed 3.3 dB, including the hybrid coupling loss.

Frequency range: 92.176 to 97.29 MHz

Level: 30 dBm

Spurious: all -30 dB below 0 dBm

3.2.1.3.2 VSWR. The voltage standing wave ratio, with respect to a resistive impedance of 50 ohms, shall not be greater than 1.2 to 1.0 at each input when all other terminals are terminated in a matched load.

3.2.1.2.5.2 Coherent/noncoherent mode switching. When provided an input from the S-band receiver, the S-band transmitter shall

3.2.1.3.3 Isolation. The isolation between each input port shall not be less than 25 dB when all other terminals are terminated in matched load.

be coherent with the reference from the receiver with the characteristics specified in paragraph 3.2.1.2.5.1, or from an internal reference oscillator.

3.2.1.2.4 S-band command receiver. An S-band phase-lock carrier tracking receiver shall be provided to demodulate to baseband the received command signal.

3.2.1.2.5.3 RF output power. The minimum S-band transmitter output power shall be 33 dBm into a nominal 50-ohm load over the spacecraft qualification temperature range.

3.2.1.2.4.1 RF input frequency. The carrier frequency of the S-band receiver shall be in the range of 2050 to 2150 MHz.

3.2.1.2.5.4 RF output frequency. The S-band transmitter RF output spectrum shall be within the range from 2200 to 2300 MHz. The RF input with full phase modulation shall be ±1.5 MHz about the carrier frequency.

3.2.1.2.5.5 RF output stability. The receiver shall supply to the transmitter a RF signal coherent with the received carrier. The frequency stability of the transmitter RF output shall be at least ±0.001 percent over the spacecraft qualification temperature range. The frequency of the RF output signal shall be 240/221 of the received carrier frequency. The RF signal level into a 50 ohm load shall be 0 dBm or greater. Spurious outputs shall be 30 dB or more below the 0 dBm output signal level.

3.2.1.2.5.6 Out-of-band outputs. Out-of-band outputs from the transmitter shall be attenuated by a minimum of 100 dB with respect to carrier in the band from 2.025 to 2.120 GHz.

3.2.1.2.4.3 Wideband demodulated output. The receiver shall provide a wideband output consisting of the demodulated ranging and command subcarrier signals and shall have the following characteristics:

3.2.1.2.5.7 Modulation characteristics.

(a) Frequency response. With a -40 dBm RF input signal, the transmitter modulator shall amplitude of the baseband shall be down less than 3 dB at 2.2 MHz and accept a composite baseband structure with a spectral occupancy from 10 Hz to 1.1 MHz.

(b) Modulation sensitivity. The modulation sensitivity over the 2200 to 2300 MHz frequency range shall be 43 degrees per volt, ±20 percent.

(c) Modulation stability. The modulation sensitivity specified in paragraph (b) shall not vary more than ±10 percent under all conditions of the spacecraft qualification temperature range.

3.2.1.2.6 RF transfer switch. A four port coaxial transfer switch shall be provided to couple the S-band transmitter to either omnidirectional antenna via the diplexers. The transfer switch shall be of the latching type with a double-throw double-pole four port coaxial configuration. It shall operate over the 2200 to 2300 MHz frequency range.

3.2.1.2.6.1 Insertion loss. The insertion loss between the input port and output port of either side of the switch shall not exceed 0.2 dB.

3.2.1.2.6.2 VSWR. The voltage standing wave ratio, with respect to a resistive impedance of 50 ohms, shall not be greater than 1.2 to 1.0 at each terminal when all other terminals are terminated in a matched load.

3.2.1.2.6.3 Isolation. The isolation between any two input or output terminals shall not be less than 60 dB when all other terminals are terminated in a matched load.

3.2.1.2.6.4 Redundancy. The switching solenoid shall be provided in redundant form such that for a failure in one solenoid, the switch can be operated by means of the redundant solenoid.

3.2.1.3 Data handling equipment group. The data handling equipment group shall contain the following components as shown in the block diagram, Figure 1:

(a) Demodulator/decoder

(b) Bus controller

(c) Computer and memory

(d) Data interface unit

(e) Baseband assembly unit

3.2.1.3.1 Demodulator/decoder unit. The demodulator/decoder accepts the demodulated 70 kHz command subcarrier from the S-band receiver and outputs a serial digital data stream to the bus controller. The decoder also outputs discrete commands for use within the CDH module.

3.2.1.3.1.1 Input signal characteristics. A 70 kHz subcarrier shall be accepted from the S-band receiver. The subcarrier shall be frequency modulated by a 2 kbit/sec Manchester-M coded data stream. Characteristics of this subcarrier are as follows:

Center frequency: 70 kHz ±100 Hz

Deviation (peak): 5 kHz ±10 percent

Signal amplitude: 0.45 volts p-p minimum

Signal distortion: TBD

Maximum load: 1 K ohms shunted by 100 pf

SNR in a 20 kHz noise bandwidth: +6 dB minimum

Data stream: Manchester-M 2 kbit/sec ±0.27

3.2.1.3.1.2 Data outputs. The command decoder shall output a 29-bit serial command word at 2 kbit/sec on four (4) individually buffered lines. The format of the word shall be as shown in Figure 3. The output of the 29-bit word shall occur after the following logic checks have been made:

(a) The uplink command is 43 bits long.

(b) A correct spacecraft address has been received.

(c) A correct 7-bit polynomial code has been viewed.

3.2.1.3.1.3 Command counter. A 7-bit counter shall be incremented when a command is verified. The counter contents shall be telemetered each minor frame indicating the count of the last accepted command.

3.2.1.3.1.4 Special command outputs. At least four (4) pulse command outputs shall be provided for use within the CDH module. These commands shall be decoded with a real-time operations code, the CDH user address and one of four special channel addresses.

3.2.1.3.1.5 Output interface levels. The output interface levels shall be telemetry compatible.

3.2.1.3.1.6 Address change. Provision shall be made for changing the vehicle address of the decoder external to the module (e.g., by an address connector).

3.2.1.3.1.7 Bit error rate. The bit error rate (BER) of the command decoder shall be less than 10^{-5} for an input carrier to noise (white gaussian) ratio of +8 dB in an equivalent noise bandwidth 20 kHz.

3.2.1.3.1.8 Automatic lock-up. With the input signal meeting the requirements of 3.2.1.3.1.1 the decoder shall lock up within a maximum of 200 bits (200 milliseconds).

3.2.1.3.1.9 TDRSS data interface. The demodulator/decoder unit shall be capable of accepting a 2 kbit/sec Manchester-M encoded data stream provided from an on-board TDRSS compatible, S-band receiver. This input shall have the same characteristics as presented in the above paragraphs and shall be distinct from the 2 kbit/sec data stream described above.

3.2.1.3.2 Bus controller unit. The bus controller shall perform the following functions:

(a) Route the incoming uplink command to the computer or directly to the data bus.

(b) Provide necessary interface signals to transmit and receive data from the computer.

(c) Provide variable rate telemetry timing on command: 64, 32, 16, 8, 4, 2, and 1 kbit/sec.

(d) Provide backup telemetry format when the computer is not used.

(e) Provide interface drivers, receivers, and timing with the data bus.

(f) Provide the master clock for the spacecraft when an OCXO is not required.

3.2.1.3.2.1 Command routing. The incoming uplink command from the demodulator/decoder shall be routed to the computer or directly to the data bus depending on the 2 bit "op" code. Commands to the computer shall be sorted on one of three direct memory access (DMA) channels depending on the "op" code function (delayed command, etc.) as shown in Figure 3.

3.2.1.3.2.2 Computer interface. Necessary signals and timing shall be provided to transmit and receive data from the computer. The bus controller shall be designed to operate with a 16 or 18 bit parallel word computer with 16 DMA channels without design change. The timing and signal levels shall be as follows:

TBD

3.2.1.3.2.3 Telemetry word rate. The telemetry word rate shall be determined by uplink command. The bit rates shall be as follows: 64, 32, 16, 8, 4, 2, and 1 kbit/sec. The bus controller shall provide all data bus timing, i.e., all requests to the computer for a data bus word (command or data) shall originate in the bus controller.

The bus controller shall interrogate the computer for the next 17-bit instruction according to the timing diagram shown in Figure 6. The computer shall respond to the telemetry format request with the next word for the telemetry frame. The computer shall respond to the interim period requests with commands, stored or real time, or a request for data independent of the telemetry system.

3.2.1.3.2.4 Backup telemetry format. The bus controller unit shall have the capability of requesting data from the data interface units so that one standard telemetry format is generated. The format shall contain as a minimum, the data illustrated in Figure 4. The variable word rates described in 3.2.1.3.2.3 shall apply to the backup format.

3.2.1.3.2.5 TDRSS data interface. An output data stream shall be provided from the bus controller unit to a TDRSS compatible transmitter when one is present onboard the spacecraft. This data stream shall be the time-division multiplex of the telemetry data defined in 3.2.1.3.2.3 and a medium-rate data stream having a maximum-data rate of 512 kbit/sec. Sufficient synchronization shall be added to allow the synchronization and decommutation of the two multiplexed data streams.

3.2.1.3.3 Computer and memory. A general purpose digital computer shall be included in the CDH module. The computer shall communicate with all spacecraft modules through time shared use of the data bus. The following computer characteristics are essential.

3.2.1.3.3.1 Memory. A capacity of 16 K words of nonvolatile memory shall be provided. Memory/processor/input-output interface design shall be capable of accommodating memory expansion to 64 K words in 8 K word segments. Cycle-by-cycle power switching of the memory shall be employed to allow expansion to 64 K with 100 mw maximum increase in standby power for each 8 K bank. Implementation of the memory shall be such that the function of any bank, including that used for fixed locations, can be achieved by any other bank.

3.2.1.3.3.2 Input/output. All input and output data channels shall be designed to operate in both a DMA mode and program control mode. One application of the DMA shall be the loading and dumping of any set of memory locations independent of processor operation and of memory content. The dump format shall include memory address and content.

3.2.1.3.3.3 Central processor. The processor design shall include the following hard-wired functions:

(a) Fixed point arithmetic

(b) One index register

(c) 16-bit word size minimum including sign

(d) 16 maskable interrupts (minimum)

(e) Add instruction (< 5 microseconds)

(f) Multiply instruction

(g) Divide instruction

(h) Four basic logical instructions minimum (AND, OR, EX OR, and COMPLEMENT)

(i) Condition/unconditional transfers

(j) Protection against illegal write-to-memory

(k) Double precision arithmetic.

Even though memory protect is a hardware function, capability shall exist for modifying via interrupt the selection of memory segments to be protected. During program execution, write cycles will be prohibited in the instruction portion of memory. To be consistent with this restriction, a "transfer and set return instruction" shall not require a write cycle in the protected area.

The processor shall have sufficient speed and instruction execution capability to perform the functions listed below. The times indicated shall include fetch and store cycles of both halves of the operands.

(a) Double precision add in 40 microseconds or less

(b) Double precision multiply in 200 microseconds or less.

3.2.1.3.4 Data interface unit. The data interface unit shall contain a remote multiplexer for data acquisition and a remote decoder for command distribution. Power for the data interface unit shall be supplied by an independent power converter which receives power directly from the primary power bus.

3.2.1.3.4.1 Remote multiplexer

(a) Multiplexer configuration. Each multiplexer shall have a minimum of 64 inputs that can be used for analog, bilevel, and serial digital signals. The signal handling capability shall allow a user to use any input for analogs, any input for bilevel (in groups of 8), and any of 32 inputs for serial digital signals.

(b) Input signal levels. All inputs of the multiplexer shall have an input impedance of 5 megohms minimum in the normal mode and 10 K ohms minimum during sampling. The multiplexer shall be capable of surviving a short circuit to ±10 VDC maximum on any one input for an indefinite time. Characteristics of the analog input signals and the digital bilevel and serial input signals shall be as follows:

Analog inputs (digitized to 8 bits)

Range: 0 to +5 VDC

Z source: 2 K ohms maximum

Accuracy: ±30 MV

Bilevel digital inputs

Logical "1": +3.0 to +5.5 VDC

Logical "0": 0 to +0.8 VDC

Fault tolerance: ±10 VDC

Z source: 500 ohms minimum; 10 K ohms maximum

Serial digital inputs (8 bits/word telemetry, 16 bits/word — computer data)

Clock rate:* 1.024 Mbit/sec

Gate width:* Envelopes 8 or 16 clock pulses

Input data

Logical "1": +3.0 to 5.5 volts

Logical "0": 0 to 3.0 volts

Z source: 500 ohms minimum

3.2.1.3.4.2 Remote decoder

(a) Characteristics. Each remote decoder shall have a minimum of 32 pulse command outputs and 7 serial magnitude command outputs. Pulse commands may serve as relay driver inputs. Characteristics of these command outputs are as follows:

Pulse commands

Pulse duration: 7.8 ms minimum

Logical "1": +3.0 to +5.5 volts

Logical "0": 0 to +0.8 volts

R source at "0": 500 ohms minimum

Magnitude commands

Clock rate:* 1.024 Mbps

Gate width:* Envelopes 16 clock pulses

Command word:* 16 bits serial

*These signal outputs have the same voltage and impedance characteristics as those shown for pulse commands.

(b) Remote unit expansion. For using modules needing more commands, the addition of expanders shall augment the capability in increments of 7 and 32 per expander to up to 56 serial digital commands and 256 pulse commands expanders.

For using modules with requirements for more than 64 data channels the addition of expanders shall augment the capability of each DIU in increments of 64 channels up to 512 channels.

3.2.1.3.5 Baseband assembly unit. A baseband assembly unit shall be provided to assemble the composite baseband signal used to phase modulate the downlink S-band carrier. The composite baseband signal shall consist of a biphase modulated subcarrier plus a direct digital data stream in one case and a biphase modulated subcarrier plus a turned-around ranging signal in the other.

3.2.1.3.5.1 Housekeeping telemetry input. A NRZ split-phase, 64, 32, 16, 8, 4, 2 or 1 kbit/sec digital signal shall be provided as an input to the baseband assembly unit from the bus controller unit. This signal shall be used to biphase modulate a 1.024 MHz subcarrier reference within the baseband assembly unit.

3.2.1.3.5.2 Telemetry subcarrier. A 1.024 MHz subcarrier reference oscillator shall be provided within the baseband assembly unit. The signal from the reference oscillator shall be biphase modulated by the digital input data stream defined in 3.2.1.3.5.1. The stability of the oscillator shall be ±21 Hz.

3.2.1.3.5.3 Subcarrier biphase modulator. The 1.024 MHz subcarrier reference defined in 3.2.1.3.5.2 shall be modulated by the data stream defined in 3.2.1.7.1.1 by means of a biphase modulator. The biphase modulator design shall be such that the residual subcarrier level shall be 20 dB below that of an unmodulated subcarrier output.

3.2.1.3.5.4 Medium-rate data input. A NRZ split-phase digital data stream shall be provided as an input to the baseband assembly unit on an optional basis. The data rate for the medium-rate data shall be a maximum of 512 kbit/sec. The medium-rate data shall be selected in an exclusive mode with the ranging data defined in 3.2.1.3.5.5 and shall be summed with the biphase modulated telemetry subcarrier defined in 3.2.1.3.5.2.

3.2.1.3.5.5 Ranging input. A ranging signal derived from the S-band command receiver shall be provided as an input to the baseband assembly unit. This signal shall have a maximum frequency component of 500 kHz and shall be filtered by means of a low-pass filter prior to combining it with the telemetry subcarrier defined in 3.2.1.3.5.2. The ranging data shall be selected in an exclusive mode with the medium-rate data defined in 3.2.1.3.5.4.

(a) Ranging turnaround filter. A low-pass filter shall be provided to attenuate noise and undesired signals from the S-band receiver. The

3 dB cutoff frequency shall be 1.5 MHz and the delay variation of the filter as a function of frequency shall be less than 200 nsecs from 50 kHz to 1.5 MHz.

(b) Summation/amplifier. A summation amplifier shall be provided within the baseband assembly unit to sum the biphase modulated 1.024 MHz subcarrier with the 512 kbit/sec medium-rate data or the 500 kHz ranging data. The voltage levels of the three signals shall be individually controllable. The composite output signal provided to the S-band transmitter shall have an attenuator adjustment to set its output voltage level to TBD.

3.2.1.4 Power conditioning unit. A power conditioning unit shall be provided to supply power to all CDH module units except the Data Interface Unit. Power conditioning for the module shall be accomplished with the following components:

(a) Bus Protection Assembly

(b) Secondary Power Converter

The block diagram of the power conditioning equipment is shown in Figure 7.

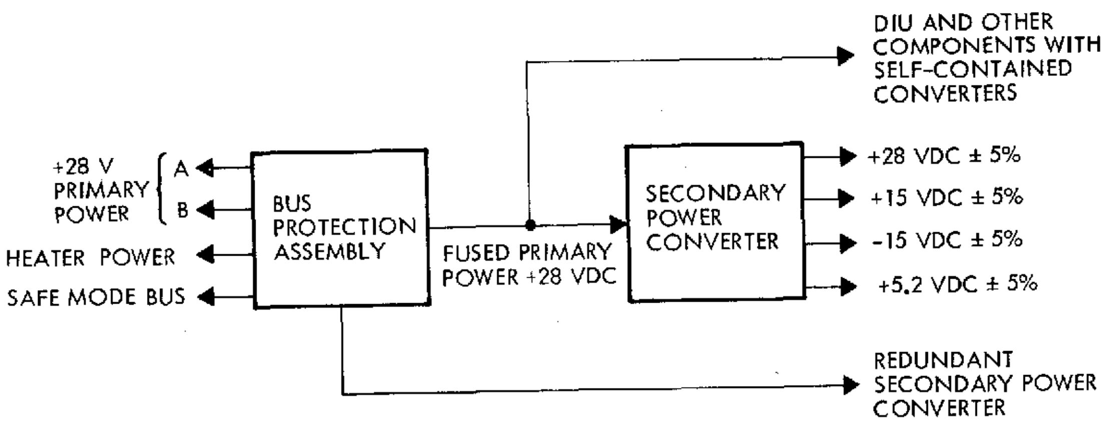


Figure 7. Power Conditioning Equipment

3.2.1.4.1 Bus protection assembly. The bus protection assembly shall provide the following functions:

(a) Fusing for the +28 volt module primary power

(b) Fusing for the +28 volt heater power

(c) Safe mode bus logic.

3.2.1.4.1.1 Module primary power fusing. Redundant fusing shall be provided for each secondary power converter as follows:

Converter No. 1: 3.7 ±0.5 ampere

Converter No. 2: 3.7 ±0.5 ampere

3.2.1.4.1.2 Heater power fusing. Redundant fusing shall be provided for each heater line as follows:

Heater No. 1: TBD ampere

Heater No. 2: TBD ampere

3.2.1.4.1.3 Safe mode bus logic. Logic shall be provided to turn off all module components except heaters when the safe mode line voltage is below 0.4 volt.

3.2.1.5 Harness. The module harness shall provide all electrical interfaces between subsystem assemblies within the module and to the module/structure interface and test connectors. The harness shall be of modular design for maximum system flexibility. Installation or removal of the harness should be possible without removing electrical assemblies. It shall be possible to remove electrical assemblies without removing the harness. Cable strain relief or back-shell potting shall be employed at all harness terminations.

Wire sizes shall be selected to hold round trip voltage drops between source and load to one percent or less of the supply voltage. The minimum wire size for power and control circuitry shall be AWG No. 20. The minimum wire size for data or test circuitry shall be AWG No. 22. Under worst case conditions, wire temperature shall not exceed the temperature rating of the wire insulation.

3.2.1.6 Thermal. The module thermal control system design constraints are presented in the following paragraphs.

3.2.1.6.1 Module thermal requirements. The module thermal design shall satisfy the following on-orbit requirements:

- The module shall be capable of operation when the heat sink temperature is $\pm 20^{\circ}F$ greater than the most severe predicted operating temperatures, where heat sink is defined as the structure or panel to which the electronic black boxes and other module equipment is mounted. These limits will be termed heat sink qualification temperatures. Less severe temperature limits can be used for components that might be damaged by the qualification temperatures if a waiver is obtained from the contractor.

- The module shall be designed so that the nominal set point temperature of the heat sink is $70^{\circ}F$ with electrical heaters turned off.

- Electrical heaters shall be incorporated to maintain the orbit-average temperature at the module attachment locations above 60°F with the heater response approximating a sine pulse over one orbital period (rather than a step-input pulse).

- All module heat dissipation shall be radiated to space from the outboard facing panel.

- The surfaces of the module, except for the panel radiator areas, shall be thermally insulated with multilayer insulation, such that the effective emissivity, $\epsilon \leq 0.01$.

3.2.1.6.2 Module/structure assembly thermal interfaces. The design of the module thermal control system shall consider the following interface constraints:

- The structure assembly/module attach point temperature will be 70 ±10°F.

- Each module attachment fitting on the structure assembly will have a thermal resistance > 5 hr-°F/BTU.

- The effective emittance, ϵ, of the structure assembly/module insulation barrier will be ≤0.02.

3.2.1.6.3 Heater power constraints. Module thermal control system heater power shall not exceed 0 watts under normal operating conditions, and 9 watts under the most severe cold operating conditions that consider predictable variations in duty cycle and heating environment as well as parameter uncertainties in thermal properties, heating environment, insulation heat loss, etc.

3.2.1.7 Module structure. The module structure shall support all equipment listed in 3.1.4 and shall be capable of supporting additional equipment listed in 3.2.1.11 for modular expansion or complete redundancy.

No amplification of the vibration or acoustic environments shall be caused by the module structure which may result in degradation of the spacecraft performance.

The module structure, when mounted on the spacecraft structure, shall withstand the launch, ascent, and on-orbit loads as defined in SP-1111.

The structure shall not cause a change in alignment of the spacecraft axes by more than TBD arc seconds.

The factors of safety shall be no less than 1.00 for limit loads and 1.25 for ultimate loads except where loads may be dangerous to personnel, the ultimate loads shall be 1.50.

3.2.1.8 Useful life. The design of the CDH module shall be such that wear out of any item or depletion of expendables will not occur prior to a useful life of TBD years. Useful life is defined as the operating time of the equipment counted from the time of launch vehicle liftoff.

3.2.1.9 Storage life. The CDH module shall have a minimum storage life of three years. Storage-life critical components may be refurbished.

3.2.1.10 Instrumentation. As a minimum, the following instrumentation (telemetry) shall be provided within the module:

Signal Definition	Signal Type
Time of day	2 digital words
Last accepted command	1 digital word
Receiver carrier loop stress	1 analog word
Receiver signal present	TBD binary words
Coherent/noncoherent mode	
Transmit RF switch position	
Decoder in/lock signal	
TBD	

3.2.1.11 Expansion capability. A capability shall be provided within the limits of module structure size and power available for expanding the baseline configuration to include the addition of redundant units or the addition of new components to perform additional functions.

3.2.2 Physical characteristics

3.2.2.1 Mechanical

3.2.2.1.1 Envelope. The module envelope shall be as shown in ICD 10.1.

3.2.2.1.2 Module volume. The module will have a volume of approximately 33 cubic feet and a maximum load carrying capability of 600 pounds of equipment. Components may be mounted to the outboard facing panel, to non-outboard surfaces and the module frame members.

3.2.2.1.3 Module weight. The total weight of the CDH module for the minimum and nominal redundancy configurations shall not exceed the listed weights.

Minimum Redundancy	Nominal Redundancy
155 lb	177 lb

3.2.2.1.4 Module center of gravity. The center of gravity of the module shall be located within TBD.

3.2.2.1.5 Attach-points. The attach-points between the module and the spacecraft shall be as shown in ICD 10.1.

3.2.2.1.6 Interface connectors. The module/structure interface connector shall be provided by the system contractor and shall be mounted on the inside face of the module. It is required that the connector position be maintained as specified to ensure interchangeability of modules.

3.2.2.1.7 Equipment expansion volume. The components shall be arranged in the module such that a minimum of 6 feet squared of the outboard facing panel is left vacant in the center of the module for the addition of up to 200 pounds of mission peculiar CDH hardware.

3.2.2.2 Electrical

3.2.2.2.1 Power. The total power required for the CDH module shall not exceed TBD watts. Allocation of this power is as follows:

Nominal redundancy configuration requirements: TBD watts
Redundancy and expansion capability: TBD watts

Power consumption of the CDH units shall be within the power allocation in ICD 10.2.

3.2.2.2.2 Commands. The CDH module shall distribute commands to other modules and/or users in accordance with ICD 10.3.

3.2.2.2.3 Telemetry. The CDH module shall accept telemetry inputs as listed in ICD 10.4.

3.2.2.2.4 Signal and power distribution. The CDH module harness shall provide all intramodule electrical connections in conformance to ICD 10.5.

3.2.3 Reliability. Compliance with reliability requirements shall be taken by prediction techniques in conformance with EOS-4.1. The allocated reliability for the CDH module baseline configuration operating under conditions specified herein for a TBD period is TBD.

Where mission objectives require changes to the baseline configuration, the reliability allocation shall be as specified in the mission specifications.

3.2.4 Maintainability. The CDH module shall be designed to emphasize accessibility and interchangeability. Field maintenance will be limited to checkout, removal, and replacement of equipment at the integral unit level. An integral unit is defined as a physical package containing factory-assembled parts.

3.2.5 Environmental conditions. The CDH module shall be designed to withstand or shall be protected against the worst probable combination of environments as specified in SP-11 and as implemented in SP-115.

3.2.6 Transportability. Transportability requirements shall be considered in the design of the CDH module such that it can be transported by all standard modes with a minimum of protection. Special packaging may be used, as required, to ensure that common carrier transportation does not impose design restrictions.

3.3 Design and construction

3.3.1 Parts materials and processes

3.3.1.1 Selection of parts, materials, and processes. Selection of parts, materials, and processes (PMP) shall be to the requirements identified in MIL-STD-143 Group I. Such selections shall be identified as standard PMP. When PMP selection cannot be made within this requirement, the item is nonstandard and justification must be made in accordance with MIL-STD-749. Control of this action shall be effected by the EOS parts, materials, and processes control board (PMPCB), with the PMPCB approving all selections, both standard and nonstandard, in accordance with EOS-4.1.

3.3.1.2 Program authorized parts list. All selections of electronic parts shall be identified and authorized by EOS-3.3-1. This list will reflect all PMPCP electronic part selections.

3.3.1.3 Program authorized materials list. All selections of materials shall be identified and authorized by EOS-3.3-2. This list will reflect all PMPCB material selections.

3.3.1.4 Program authorized processes list. All selections of processes shall be identified and authorized by EOS-3.3-3. This list will reflect all PMPCB process decisions.

3.3.1.5 Dissimilar metals. To avoid electrolytic corrosion, dissimilar metals shall not be used in direct contact unless protection against corrosion has been provided in accordance with MIL-E-8983 and MIL-STD-454, Requirement 16.

3.3.1.6 Magnetic materials. Magnetic materials shall be used only if necessary for equipment operation. Those magnetic materials which are used shall have minimum permanent, induced, and transient magnetic fields.

3.3.1.7 Fungus-inert materials. Materials which are fungus-inert, in accordance with MIL-E-8983, and MIL-STD-454, Requirement 4, shall be used.

3.3.1.8 Flammable toxic and unstable materials. Flammable, toxic, and unstable materials shall not be used.

3.3.1.9 Finish. The surface of each component of the subsystem shall be adequately finished to prevent deterioration from exposure to the specified environments that might jeopardize fulfillment of the specified performance. The finish shall also meet the applicable bonding requirements. Thermal properties of the finishes used on the component shall be compatible with the requirements given in detail in applicable equipment specifications. The requirements for finishes identified in MIL-E-8983 shall be implemented.

3.3.1.10 Outgassing. Low outgassing polymeric materials shall be used where sensitive thermal control and other surfaces are in direct line-of-sight and where temperature differences can exist between such surfaces.

3.3.2 Electromagnetic radiation

3.3.2.1 EMC/EMI requirements. The command and data handling module, and all internal units, equipments, and/or components comprising a part thereof, shall be designed for compliance with the radiated emission and susceptibility requirements of NASA/JSC Specification SL-E-0002 as modified or amended by EOS-3.3-4, and EOS-3.3-5. The conducted emission and susceptibility requirements of the aforementioned documents are applicable only at the module to spacecraft structure interface. EMI/EMS levels at interfaces within the module shall be controlled to the extent necessary to ensure self-compatibility of the module subsystem with a safety margin of at least six (6) dB.

3.3.2.2 Electrical bonding

3.3.2.2.1 Structural bonds. All metallic members of the basic CDH module radiator panel and support structure shall be electrically bonded to each adjacent member to form an electrically continuous, equipotential ground reference plane. The maximum allowable DC impedance across any one structural joint shall be 2.5 milliohms with an overall design goal of 10 milliohms, or less, between any two diametrically opposite points on the module. The general methods of MIL-B-5087, Class R, may be used for implementation of these requirements.

3.3.2.2.2 Equipment mounting pads. Surfaces on the module radiator panel and structure which are intended for unit, equipment, or component mounting shall be free of paint, anodize, or other nonconductive finishes. The maximum DC impedance between the baseplate of any electrically active unit, equipment or component, and the module radiator panel shall be 2.5 milliohms.

3.3.2.2.3 Electrical connectors. All interface electrical connectors, both plug and receptical, which form a part of the unit and cable RF shielding system within the module shall have electrically conductive body shells, free of nonconductive finishes. They shall also have provisions for terminating (grounding) the shields on the interconnecting electrical harness. The maximum DC impedance between the shield termination point and the baseplate of the parent unit shall be 2.5 milliohms.

3.3.2.2.4 Electrostatic bonds. Electrically passive components or appurtenant structures which are attached to the basic module structure through thermal isolators shall be provided with electrostatic bonds having a DC impedance of 1.0 ohms, or less. The general methods of MIL-B-5087, Class S, may be used for implementation of this requirement.

3.3.2.3 Electrical systems grounding

3.3.2.3.1 Primary DC power. All primary DC power returns shall be isolated from chassis/case/structure by a minimum of one (1) megohm resistance. These returns shall be routed, along with the DC power input lead, via unshielded twisted-pair wiring to the module to spacecraft interface connector for eventual grounding in the power module. Power conversion/DC isolator units or subunits shall be provided at each primary power user terminal to maintain the required DC isolation between the primary and secondary power distribution systems.

3.3.2.3.2 Secondary DC power. Secondary DC power distribution networks shall, in general, use multiple point grounding within the CDH module with returns through the module radiator panel and/or support structure. These networks shall be initially grounded adjacent to the secondary transformer winding in the power converter and at each user element.

3.3.2.3.3 Secondary AC power. Secondary AC power networks shall use single-point grounding with a two-wire, twisted shielded pair distribution. The ground point may be either at the source or load end, whichever is shown by circuit analysis to be most beneficial to compatible module operation. Structural returns shall not be used for secondary AC power.

3.3.2.3.4 Intramodule signal/control circuits (high level). All high level ($\geq$5 volt logic, bilevel or analog) signal or control circuits which do not exit the CDH module shall be multiple-point grounded at both the source and load end of each circuit branch to unit or component chassis by the shortest most direct path. Circuits sharing space on a common printed circuit board should not share common grounding traces on the board or common hardwire jumpers to chassis ground logs. Preferably, each such board should have a dedicated ground plane layer to which all components requiring ground returns can be directly connected. This ground plane should, in turn, be directly bonded to unit chassis or frame through grounding pads at each hold-down fastener.

3.3.2.3.5 Intermodule signal/control circuits (high level). All high-level signal or control circuitry which exit the CDH module shall be grounded at the final driving element. Two-wire, twisted shielded pair distribution shall be used for each such circuit between the source unit and the module to spacecraft interface connector. The load elements in the external module will be DC isolated from structural grounds. Any signal or control circuitry which enters the CDH module shall be DC isolated from chassis/case/structure ground by a minimum of one (1) megohm resistance. Any such circuits shall also be provided with two-wire, twisted shield pairs between the interface connector and the load unit.

3.3.2.3.6 Analog circuits (low level). Any low-level ($<$5 volts) analog circuits, which are shown by circuit analysis or test to be sensitive to circulating currents in the module or spacecraft structure, shall be single-point grounded either at the source or load element, whichever is most appropriate for the circuit under consideration. Wherever possible, balanced differential circuitry should be used. In the case of low-level circuits which enter or exit the CDH module, the location of the circuit ground point shall be coordinated with the systems integration contractor.

3.3.2.3.7 Data bus. The command and telemetry data bus system shall be differentially driven and balanced to structural ground in the CDH module. This system shall be transformer-coupled at each remote terminal. Each individual data bus wire entering the CDH module shall be DC isolated from chassis/case/structure by a minimum of one (1) megohm resistance.

3.3.2.3.8 Wire shields. External shields shall be provided for all interconnecting wires between units, equipments, or components in the CDH module and between each input/output connector and the module to spacecraft interface connectors except for the primary DC power lines. In general, these shields shall be multipoint grounded at each end and at each intermediate interface. An exception to this rule will be allowed for low-level analog circuitry where single-point shield grounding may be necessary. If possible, such circuitry should be provided with two, mutually-isolated shields, the inner shield being single-point grounded and the outer shield multipoint grounded.

3.3.2.3.9 EMI filter components. High performance EMI filters will be required at the primary DC power input and return terminals of each power converter unit or subunit in the CDH module in order to ensure compliance with the electromagnetic interference requirements of the applicable specifications. These filters should have two stages: 1) an AF ripple filter stage balanced line to line; and 2) a pair of RF feedthrough filters bulkhead mounted behind the input connector. The combined AF/RF filter circuit should be designed for the minimum capacitance from either line to chassis necessary to achieve compliance with the specifications. Excessive line to ground capacitance will tend to negate the beneficial effects of the twisted pair wiring used in the primary DC power harness.

3.3.3 Nameplates and product marking. Each subsystem shall be identified in accordance with MIL-STD-130 as implemented by EOS-3.3-6. Where practical, a minimum character height of 0.09 inch shall be used. Marking shall include the manufacturer's name or initials, assembly part number and revision status, assembly serial number, contract number, and others as delineated in the component equipment specifications.

3.3.4 Workmanship. The module and its component equipment shall be constructed, finished, and assembled in accordance with MIL-STD-454, Requirement 9 and the specifications and drawings specified herein.

3.3.5 Interchangeability and replaceability. The CDH module shall be designed to permit removal and replacement of components with a minimum of disturbance of associate or adjacent equipment. Design for service and access shall conform to the principles and requirements of MIL-STD-470.

3.3.6 Safety. The CDH module shall be designed to meet or exceed the requirements of EOS-3.3-7, as implemented by EOS-4.1. The design criteria shall include but shall not be limited to those set forth in MIL-STD-882.

3.3.7 Human performance/human engineering. MIL-STD-1472A shall be used for the design of man/machine interfaces.

3.4 Documentation. Documentation shall be as specified in EOS-3.3-8.

4. QUALITY ASSURANCE PROVISIONS

4.1 General. The quality assurance program controls shall be in accordance with the requirements of MIL-Q-9858 as augmented by EOS-4.1.

4.1.1 Responsibility for inspection and test. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified herein. Except as otherwise specified, the supplier may use his own facilities or any commercial laboratory acceptable to the government. The government reserves the right to perform any of the inspections set forth in the specifications where such inspections are deemed necessary to ensure that supplies and services conform to prescribed requirements.

4.2 Quality conformance inspections

4.2.1 Category I tests

4.2.1.1 Development tests. Development tests shall be conducted to verify the module performance parameters, proper interfacing between components of the module, and proper interfacing between the module and other spacecraft modules. The development tests shall be conducted at the unit and/or integrated module levels. The type of developmental evaluation or test to be applied to verify satisfaction of the requirements defined in this specification are outlined in Table 1. The tests shall be conducted in accordance with EOS-4.2 and EOS-4.3.

4.2.1.2 Qualification tests. Module qualification shall be accomplished by means of qualification tests on the individual units and/or as a normal consequence of having the units installed in the qualification test space vehicle during its qualification testing. The types of qualification evaluation or test to be applied to verify satisfaction of the requirements of this specification are outlined in Table 1. Qualification test verification methods and requirements shall be as defined in the EOS-4.2.

4.2.1.2.1 Components. As a minimum, the following types of component qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, are measured before and after the environmental tests.
- Random vibration. Each component will be subjected to a random vibration from 20 to 2000 Hz in all three axes while

Table 1. Verification Cross Reference Index

VERIFICATION CROSS REFERENCE INDEX

LEGEND: N/A - Not Applicable; I - Inspection; A - Analysis; S - Similarity; T - Test

SECTION 3 REQUIREMENTS	Not Applicable	Acceptance			Qualification				SECTION 4 VERIFICATION REQUIREMENTS
		I	A	T	I	A	S	T	
(TBD)									

electrically powered in the maximum normal load condition. Performance will be continuously monitored to detect failures (either hard or trends).

- Thermal vacuum. Thermally cycle the component beyond expected orbital temperatures while at an orbital vacuum condition. The component is operating continuously and detailed performance parameters are measured at each temperature extreme.

- Electromagnetic interference (EMI) and susceptibility (EMS). Engineering EMI and EMS data are measured to provide diagnostic information for the module EMI/EMS qualification.

4.2.1.2.2 Module. As a minimum, the following types of module qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, and electrical interface characteristics are measured before and after the environmental tests.

- Acoustics. Acoustics testing will be performed with the module mounted in a receptacle which provides acoustics shielding similar to the Observatory structure. The module will be heavily instrumented to verify design adequacy of interconnections (electrical and mechanical) between the module and other Observatory elements. The test data will be used to confirm estimates of the vibration environment for components mounted on radiators and establish procedures for acceptance testing. The module will be electrically powered and continuously monitored via the data bus to detect failures and out-of-tolerance performance trends.

- Thermal vacuum. The objectives of the module thermal qualification test are to determine: 1) maximum and minimum operating temperatures for the module for worst-case environments and operating duty cycles, 2) temperature levels and temperature variations of the module adjacent to the interface fittings, and 3) heater power requirements for cold-case conditions.

  The test will be conducted in a thermal-vacuum chamber with liquid nitrogen cold walls. The module will be mounted on a fixture simulating the spacecraft frame. The properties of this fixture shall permit its temperature to be held constant at any level between 0 and 100°F.

  A heating source for the radiators will approximate the absorbed flux of the external environment. This can be done with electrical heaters, infrared lamps, or other techniques where the absorbed heating can be determined accurately.

- Power bus and data bus will be tested in excess of their operational limits to determine design margins and compliance with the interface specification.

- Detailed performance data will be measured to determine module specification values.

- Thermistor/heater control and calibration will be determined.

- EMI and EMS. EMI measurements will determine the levels radiated on each module electrical interface and verify there is adequate design margin to ensure non-interference with other modules. EMS measurements will verify that each module has adequate susceptibility margin to prevent performance degradation resulting from other module allowable radiation levels. The general test methods of MIL-STD-462, as modified or amended by EOS-3.3-5, shall be used during this test program except that, wherever practical, spectrum analyzers or other forms of rapid display and analysis equipment may be used in lieu of equipment requiring manual tuning and data collection.

4.2.1.2.3 Inspection sequence. The inspection sequence shall be governed by the following:

- Examination of the module shall be performed prior to functional testing.

- Functional tests shall be performed prior to, during, where appropriate, and following environmental testing. The functional testing conducted at the conclusion of an environmental test may serve as the functional testing to be performed prior to the next environmental test.

- Environmental testing may be performed in any sequence.

4.2.1.2.4 Failure criteria. The module shall exhibit no failure, malfunction or out-of-tolerance performance degradation as a result of the examinations and test specified herein. Any such failure, malfunction or out-of-tolerance performance degradation shall be cause for rejection.

4.2.2 Test conditions. Unless otherwise specified, the conditions under which all inspections are accomplished shall be specified below:

4.2.2.1 Atmospheric and environmental. All examinations and tests shall be conducted under the local prevailing pressure at the time of test, at a temperature of 75 ±5°F and at a relative humidity of 55 percent or less.

4.2.2.2 Deviations of atmospheric conditions. When tests are performed with atmospheric conditions substantially different from the specified values, proper allowances for changes in instrument readings shall be made to compensate for the deviation from the specified conditions.

4.2.3 Equipment warmup time. The equipment warmup time shall be less than 1 minute.

4.2.4 Temperature stabilization. Temperature stabilization shall have been achieved when temperature of the equipment mounting structure varies not more than 5°F during a period of 15 minutes.

4.2.5 Measurements. Measuring instruments used to determine functional parameter values (such as voltage, frequency, current, flow, pressure, etc.) shall indicate true values with an accuracy determined by the tolerance allowed for the parameter variation itself, such that the measuring instrument shall not introduce an uncertainty greater than 10 percent of the allowable variation of the measured parameter. However, no such measurement accuracy shall be required to exceed 0.5 percent of the required value of the parameter unless otherwise specified.

Except as specifically noted in the inspection methods, tolerances for environmental test conditions shall be defined as follows:

Temperature: ±5°F

Barometric pressure: ±10 percent

Relative humidity: +5, -10 percent

Time: ±5 percent

Vibration amplitude (sine): ±10 percent

Vibration frequency: ±2 percent

4.3 Category II tests. (Not applicable)

4.4 Analyses. The following requirements of Section 3 shall be verified by review of analytical data:

3.2.3 Reliability. To be verified by analysis in accordance with Section TBD of EOS-4.1.

3.3.6 Safety. To be verified by analysis in accordance with Section TBD of EOS-4.1.

5. PREPARATION FOR DELIVERY

5.1 General. The CDH module as specified herein shall be protected from degradation by environments anticipated during shipment, handling, and storage. Standard commercial packaging practices are acceptable provided they fulfill these requirements.

5.2 Preservation and packaging

5.2.1 Cleaning. The CDH shall be clean and free of contaminants which might impair its use prior to packaging.

5.2.2 Attaching parts. When attaching parts such as nuts, bolts, washers, etc., accompanying the CDH module they shall be preserved, bagged, appropriately identified, and attached to, or adjacent to, the fitting for which they are intended.

5.2.3 Electrical connectors. All electrical connectors shall be capped with protected dust caps. Caps used shall be a friction fitting or threaded type which do not require tape or a mechanical device to secure.

5.2.4 Critical surfaces. External machined surfaces and mounting surfaces of the CDH module shall be protected with protective pads. Materials used for pads shall not cause item deterioration.

5.2.5 Wrapping. The CDH module shall be wrapped or bagged using antistatic polyethylene film.

5.2.6 Cushioning. When required for protection, the CDH module shall be cushioned using a suitable resilient foam cushioning material such as polyethylene, polyurethane, or polystyrene.

5.3 Packing. The CDH module shall be packed in a suitable shipping container designed for one item only. The shipping container used shall provide protection to the item against corrosion, deterioration, and damage during shipment from the source of supply to the receiving activity. Containers shall comply with applicable tariffs and regulations for particular modes of transportation when so shipped.

5.3.1 Storage conditions. The CDH module shall not be adversely affected by storage within its container at temperatures between 60 and 90°F and relative humidities of 60 percent or less.

5.3.2 Shipping conditions. The CDH module shall be capable of withstanding the following environments:

Temperature: +160°F in an unsheltered area (125°F +35°F due to solar radiation) and -40°F in accordance with restricted air transport criteria

Humidity: Up to 100 percent in an unsheltered area

Rough Handling: Capable of withstanding and physically protecting the unit from rough handling during shipping by common carrier.

5.4 Marking for shipment. Each CDH module and shipping container shall be marked with the following:

(a) Item nomenclature

(b) Part number

(c) Contract or purchase order number

(d) Manufacturer's name

(e) Manufacturer's part number and serial number (on item container only)

(f) Quantity

(g) Date of manufacture (on item container only)

(h) FRAGILE - HANDLE WITH CARE (when applicable)

(i) SPACE VEHICLE MATERIAL - DO NOT OPEN IN RECEIVING INSPECTION (when applicable - shipping container only)

(j) Actual weight.

5.4.1 Documentation. All required reliability and test documentation such as test reports, certifications, shipping invoices, etc., shall be either packed in the CDH module container or attached to the exterior surface of the shipping container. Attachment shall be such a manner as to preclude loss of this data during handling and shipment by common carrier.

6. NOTES

6.1 Definition of spacecraft configuration

6.1.1 Minimum redundancy configuration. The minimum redundancy configuration is defined as the spacecraft configuration which contains the minimum redundancy of units necessary to ensure that no plausible single-point failure will prevent Observatory retrieval by the Space Shuttle System. For purposes of this specification this configuration is identified as the baseline spacecraft configuration.

6.1.2 Nominal redundancy configuration. The nominal redundancy configuration is defined as the spacecraft configuration which includes standby redundant units for most of the electronic assemblies to provide a "typical" redundancy level for long-life spacecrafts.

TITLE

PRIME ITEM DEVELOPMENT SPECIFICATION
FOR THE

ELECTRICAL POWER MODULE

DATE 20 SEPT 1974 NO. SP-1113

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

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SPECIFICATION SP-1113

ELECTRICAL POWER MODULE

1. SCOPE

This specification establishes the requirements for the design, development, and performance of the electrical power module for the EOS multimission spacecraft.

2. APPLICABLE DOCUMENTS

2.1 Government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

SPECIFICATIONS

NASA

SP-11	Observatory Segment Specification
SP-1111	Structure Assembly Specification
SP-115	EOS Environmental Specification
SP-1112	Communication and Data Handling Specification

Military

MIL-B-5087	Bonding, Electrical, and Lightning Protection for Aerospace Systems
MIL-E-8983A	Electronic Equipment, Aerospace Extended Space Environment, General Requirements
MIL-Q-9858A	Quality Program Requirements

STANDARDS

Military

MIL-STD-143B	Standard and Specifications, Order of Precedence for Selection
MIL-STD-454C	Standard General Requirement for Electronic Equipment
MIL-STD-749B	Preparation and Submission of Data for Approval of Nonstandard Electronic Parts

MIL-STD-1472A	Human Engineering Design Criteria for Military Systems, Equipment and Facilities
MIL-STD-882 15 July 1969	System Safety Program for Systems and Associated Subsystems and Equipment, Requirements for
MIL-STD-462	EMI/EMS Test Methods
MIL-STD-130	Name Plates
MIL-STD-470	Maintainability Program Requirements (For Systems and Equipments)

OTHER PUBLICATIONS

NASA

SL-E-0002	Electromagnetic Compatibility Control Plan
NASA STDN101.1 X-560-63-2	STDN User's Guide Aerospace Data Systems Standards
EOS-3.3-1	Program Authorized Parts List
EOS-3.3-2	Program Authorized Materials List
EOS-3.3-3	Program Authorized Processes List
EOS-3.3-4	Electromagnetic Compatibility Control Plan
EOS-3.3-5	EMI/EMS Limits and Test Methods
EOS-3.3-6	Marking of Parts and Assemblies

2.2 Non-government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. If no revision or date is shown, the latest released issue of the applicable document shall apply. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

DRAWINGS

20.5	Electrical Power Module Wire List
10.4	Satellite Telemetry Allocation, ICD
10.3	Satellite Command Allocation, ICD

10.2	Satellite Primary Power Allocation, ICD
20.1	Electrical Power Module Envelope, ICD
TBS	Electrical Power Module Assembly Drawing

OTHER PUBLICATIONS

EOS-3.3-7	Safety Design Criteria
EOS-3.3-8	Configuration Management Plan
EOS-4.1	System Effectiveness Program Plan
EOS-4.2	Integrated Test Plan

3. REQUIREMENTS

3.1 General

3.1.1 Function. The EOS electrical power module (EPM) shall provide the following major functions:

- Energy-storage
- Battery charge regulation
- Power distribution control

The EPM shall be capable of supplying and controlling all power within tolerance from full load to minimum load. The EPM shall meet these requirements for orbital average mission loads from 0.3 to 2 kW by the addition or removal of energy storage and power control equipment. Such additional equipment shall be identical in configuration and manufacture to the basic EPM equipment. Modification of the basic EPM to accommodate the equipment for any mission shall not exceed installation of additional harness, and adjustment or replacement of current sensitive control and fault protection devices.

3.1.2 Operation. The EPM is required to provide electrical power to the spacecraft and the payload. The configuration and major interfaces of the EPM are illustrated in Figure 1. Primary unregulated power is routed to the EPM from the solar array and drive module via the spacecraft harness. Power during eclipse and peak loads in excess of solar array capability shall be provided by nickel-cadmium batteries contained within the EPM. Isolation between primary power input lines and between batteries shall be obtained by the use of diodes, and isolation of the power outputs from the EPM to the spacecraft and payload shall be obtained by the use of switches and/or circuit breakers. Primary power from the solar array shall be regulated within the EPM when appropriate for battery charge

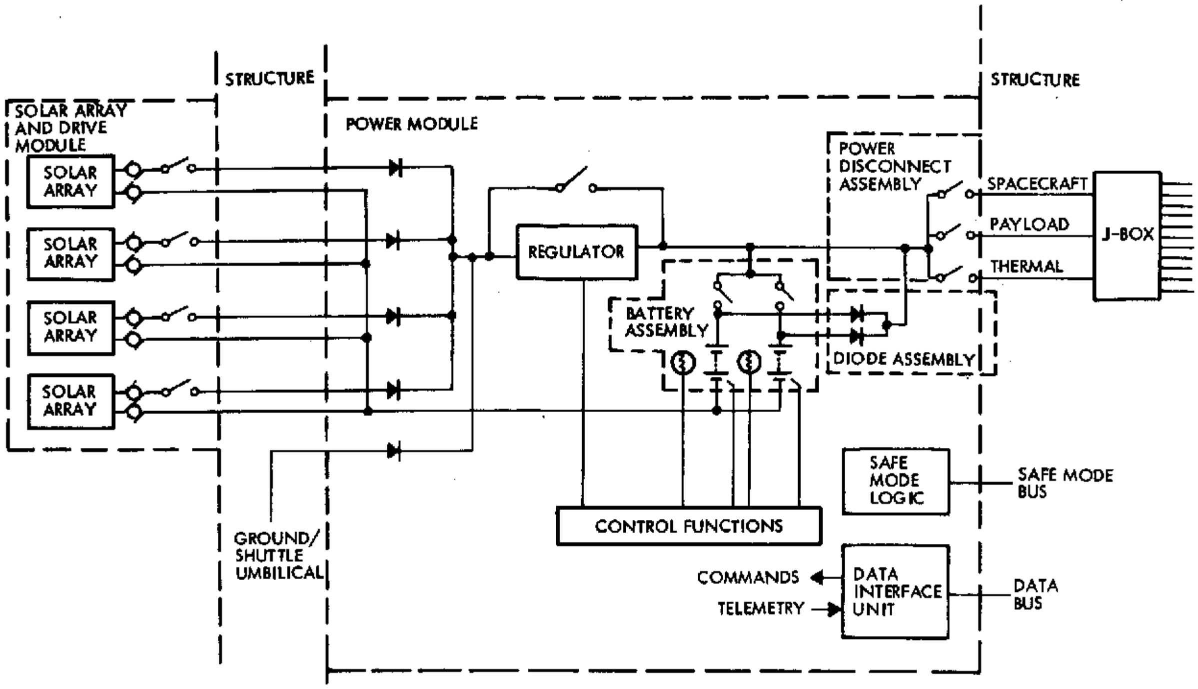

Figure 1. Power Module and Interface Block Diagram

control. The EPM shall also provide the capability for sensing the fault conditions and enabling the safe mode within the Observatory.

Ground/shuttle umbilical interfaces to the EPM shall be provided to enable total control of power and power distribution from a remote control panel. This capability shall include features which permit the removal of power from any or all module interfaces within the Observatory during shuttle refurbishment operations.

3.1.3 Equipment list

3.1.3.1 Contractor-furnished equipment. The major components of the baseline EPM are listed in Table 1.

3.1.4 Government furnished equipment (GFE). The data interface unit will be GFE. One unit is required for either the baseline or the growth versions of the EPM.

3.1.4 Item definition

3.1.4.1 Item diagram. The block diagram of Figure 2 illustrates the relationship of the components necessary to provide the EPM functions:

3.1.4.1.1 Diode assembly. The diode assembly shall provide isolation from shorts occurring on EPM input power paths. Isolation between batteries shall also be provided.

Table 1. Major Component List

Component Name	Quantity per Baseline EPM
Power control unit	1
Battery assembly	1
Diode assemblies	2
Power disconnect assembly	1
Secondary power and bus protection assembly	1
Power module harness	1
Power module thermal control	1
Power module structure	1

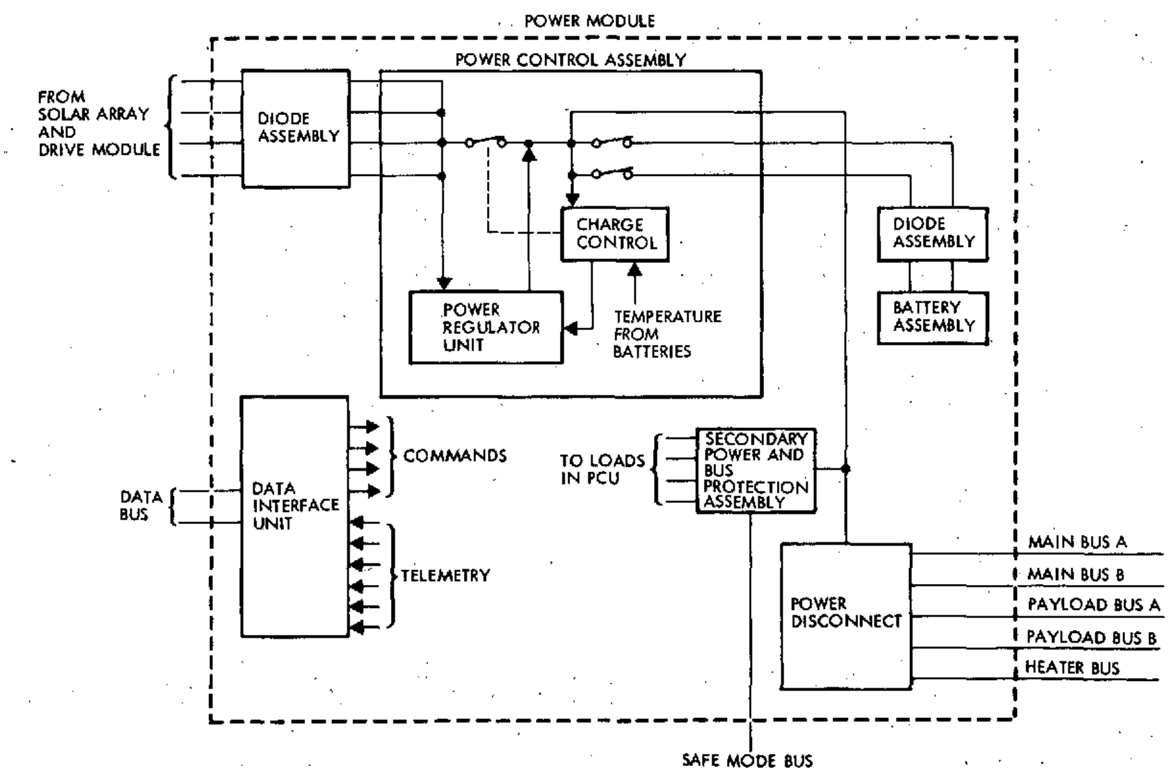

Figure 2. Electrical Power Module Block Diagram

3.1.4.1.2 Power disconnect assembly. The power disconnect assembly contains the relays, breakers, and electronic drive circuitry necessary to provide all power distribution control and output power bus fault isolation.

3.1.4.1.3 Battery assembly. The battery assembly contains the nickel-cadmium batteries, temperature sensing electronics, voltage and current telemetry signal conditioning.

3.1.4.1.4 Secondary power and bus protection assembly. This assembly contains all secondary power conversion equipment necessary to the operation of the EPM equipment. Input power is derived from the load side of the power distribution switchgear in the EPM. Fuses required to provide fault isolation of redundant and/or noncritical electronics are mounted on this assembly.

3.1.4.1.5 Data interface unit (DIU). The data interface unit provides the digital-to-bilevel and analog-to-digital conversion necessary for the EPM interface with the on-board computer. The data interface unit drives power directly from the main bus and is functionally independent of the EPM.

The data interface unit shall contain all conversion, conditioning, and bus protection equipment necessary for the operation of the data interface components, and shall be a complete assembly delivered ready for installation into the EPM.

3.1.4.1.6 Power control assembly. The power control assembly shall contain sensing and control necessary to implement power system operational modes. The functions shall include analog signal processing of battery voltage, current, and temperature signals, battery switching control, control logic, under/overvoltage protection, power conditioning control logic, main bus power conditioning, and fault protection logic. The power control assembly shall be capable of operating in parallel with a second power control assembly for EPM configurations which have a minimum of one and a maximum of four batteries charging in parallel.

3.1.5 Interface definition

3.1.5.1 Internal interfaces. The interfaces between assemblies of the EPM shall be interconnected by means of the harness assembly entirely within the EPM.

3.1.5.1.1 Battery assembly/power control unit. Battery charge power shall be routed from the regulated battery charge bus through charge enable/disable switches to the battery assembly. Current, voltage and temperature monitors within the battery assembly shall be routed to the power control unit (PCU) for control of the battery charge and for transmission through the data bus for ground monitoring. Each battery will also provide one adhydrode cell signal to the PCU for use in battery charge control when selected by command. This signal will also be telemetered to the ground via the data bus.

3.1.5.1.2 Battery assembly/diode assembly. The battery discharge power shall be routed from the batteries to the primary power bus through diodes in the diode assembly. A diode shall be provided for each battery and oriented to prevent the flow of current from the primary power bus to the battery when the associated charge enable/disable switch is operated open.

3.1.5.1.3 Diode assembly/PCU. The unregulated primary power bus from the solar array and drive module and the ground/shuttle power umbilical line shall be routed through the diode assembly for isolation before entering the PCU.

3.1.5.1.4 Secondary power and bus protection/PCU. The secondary voltages necessary to the operation of the PCU functions will be derived within the secondary power and bus protection assembly and routed to the appropriate locations within the PCU.

3.1.5.1.5 Data interface unit/PCU. The command inputs and telemetry outputs of the EPM shall be routed between the PCU and the DIU for transmission via the data bus.

3.1.5.2 External interfaces

3.1.5.2.1 Primary power. Unregulated primary power shall be routed to the EPM from the solar array and drive module during periods of sunlight.

Primary power shall be distributed to the spacecraft modules on two independently controlled and fault isolation main buses. The main bus redundant primary power shall be routed from the power disconnect assembly within the EPM to a J-box on the spacecraft structure. The redundant primary power lines to each spacecraft module shall originate from the spacecraft J-box.

Primary power shall be distributed to the payload modules on two independently controlled and fault isolated payload buses. The payload primary power buses shall be routed from the power disconnect assembly within the EPM to a J-box located at the spacecraft/payload transition ring. The primary power lines to the payload modules will originate from one of the two buses within the transition ring J-box.

Primary power for the structure and module heaters of the Observatory shall be distributed on an independently controlled and fault isolated bus. The heater primary power shall be routed to the spacecraft and transition ring J-boxes for distribution to the required locations.

3.1.5.2.2 Secondary power. Secondary power shall not be distributed outside the electrical power module.

3.1.5.2.3 Mechanical interfaces. The electrical power module shall meet the requirements of ICD 20.1.

3.1.5.2.4 Functional interfaces. The external interfaces of the EPM exclusive of primary power shall be limited to the data bus and the safe mode bus.

3.1.5.2.4.1 EPM/communication and data handling (CDH) module. The EPM shall include a data interface unit for receipt of commands from and transmission of telemetry to the CDH module via the Observatory data bus.

3.1.5.2.4.2 EPM/safe mode bus. The EPM shall incorporate logic to sense the requirement for and enable the Observatory safe mode as described in paragraph 3.2.1.3. The safe mode bus enable signal from the EPM shall be acted upon in the attitude determination module and solar array and drive module.

3.1.6 Government-furnished module equipment. Certain items of module equipment which are identical or nearly identical in function will be fabricated by the integration contractor and provided as GFE to the remaining module contractors. This equipment shall include:

Item	Reference
Data interface unit	3.2.1.3.4 of SP-1112
Power control unit	3.2.1.9 of SP-1113

All module structures and integration of the modules will be provided by the integrating contractor.

3.2 Characteristics

3.2.1 Performance

3.2.1.1 General. The electrical power module shall be capable of controlling and regulating adequate power for steady state and transient loads to the Observatory during prelaunch, ascent, on-orbit operation, and shuttle servicing for at least TBD years. During normal on-orbit operation, the EPM shall provide power to enable all spacecraft and payload modules to operate with the required duty cycle. Power during periods of illumination will be generated by the solar array and delivered to the EPM from the solar array and drive module. Power during periods of eclipse will be derived from the batteries within the EPM. Peak loads exceeding the illuminated array capability may be supported by the solar array and batteries in combination provided that the limits of battery depth-of-discharge specified in paragraph 3.2.1.5 are not exceeded for more than TBD orbits.

The electrical power module shall consist of a basic equipment complement capable of controlling and regulating at least 0.3 kW, and incorporating energy storage of TBD kW-hr (100 percent depth-of-discharge). Power control and regulation capability up to 2.0 kW shall be obtainable by the addition of energy storage and power control equipment to the basic EPM equipment complement.

As a minimum the EPM shall use redundancy to ensure that the Observatory safe mode can be enabled and maintained until servicing can be accomplished. No single-point failure mode shall exist that could cause mission failure or prevent enabling and maintaining the safe mode. Current limiting or fuse devices shall be used to protect the primary power buses from short circuits occurring within the control and monitoring components of the EPM. Total or partial failure or inactivation of a battery shall not degrade spacecraft operation during illuminated periods.

3.2.1.2 Primary power buses

3.2.1.2.1 Power. The EPM shall supply power to the spacecraft, payload, and heater buses in accordance with the requirements specified for each mission of EOS.

3.2.1.2.2 Bus characteristics. The bus characteristics at the power module/structure interface connector shall be as follows:

Regulation

28 ± TBD volts

Ripple

≤0.5 volts from 5 Hz to 100 kHz

Transient

Amplitude ≤1.0 volts peak

Duration ≤0.1 second

Impedance*

≤0.15 ohm from 1 Hz to 5 kHz

≤0.5 ohm from 5 kHz to 100 kHz

≤1.0 ohm from 100 kHz to 1 MHz

*Under following load conditions:
1. a) Solar array source simulation disconnected
   b) Regulator "OFF" (shorted)
   c) 50% ±30% maximum discharge load (resistive)
2. a) Solar array simulator connected
   b) Fully charged batteries floating on power bus
   c) Sunlit period average load (resistive)

3.2.1.3 Safe mode bus. The EPM shall provide logic to sense anomalous sun pointing by monitoring solar array current. This logic shall be activated if either of two conditions occurs:

(a) Loss of solar array current for a period exceeding the maximum eclipse time.

(b) Two consecutive sunlight periods are experienced without the batteries reaching the voltage limit (BVLS).

Activation of the safe mode logic shall result in a change of state on the safe mode bus from open (essentially infinite resistance to ground) to short (essentially zero resistance to ground).

The EPM shall incorporate the capability to override and/or reset the safe mode logic by ground command.

3.2.1.4 Power control assembly. The power control assembly of the EPM shall control and maintain the selected operational modes of the electrical power system throughout the mission life. The primary function of the power control assembly shall be to ensure the normal sequences of battery charge/discharge operation consisting of: 1) battery discharge to load during eclipse, 2) shunt charge of the batteries with the output of the solar array directly connected to the batteries and Observatory loads, and 3) regulated control of the solar array output to the batteries and Observatory loads.

Secondary functions of the power control assembly shall be primary power bus under/overvoltage detection, Observatory safe mode monitoring and initiation, and analog signal processing of battery voltage, current, and temperature signals.

3.2.1.5 Battery assembly. The battery assembly shall contain a minimum of two (baseline configuration) and a maximum of four, 22-cell batteries with an individual capacity not to exceed 40 ampere-hours. Each battery shall include one adhydrode cell to detect battery overcharge and fault conditions. Electronics contained in the battery assembly shall be limited to that necessary to provide measurements of battery voltages, current and temperature. The signal outputs shall be buffered so that short circuits or applied voltages up to 35 volts shall not cause a battery failure or decreased battery life.

3.2.1.6 Diode assembly. The diode assembly of the EPM shall provide isolation of the solar array power buses originating in the solar array and drive module, and isolation of the individual battery discharge outputs from the battery assembly. Isolation for the ground/shuttle primary power input shall also be provided in the diode assembly.

3.2.1.7 Power disconnect assembly. The power disconnect assembly of the EPM shall provide for the reliable interruption and restoration of the primary power to the spacecraft, payload, and heater buses. Protection shall be incorporated to prevent disconnecting both

- The module shall be capable of operation when the heat sink temperature is ±20°F greater than the most severe predicted operating temperatures, where heat sink is defined as the structure or panel to which the electronic black boxes and other module equipment is mounted. These limits will be termed heat sink qualification temperatures. Less severe temperature limits can be used for components that might be damaged by the qualification temperatures if a waiver is obtained from the contractor.

- The module shall be designed so that the nominal set point temperature of the heat sink is 70°F with electrical heaters turned off.

- Electrical heaters shall be incorporated to maintain the orbit-average temperature at the module attachment locations above 60°F with the heater response approximating a sine pulse over one orbital period (rather than a step-input pulse).

- All module heat dissipation shall be radiated to space from the outboard facing panel.

- The surfaces of the module, except for the panel radiator areas, shall be thermally insulated with multilayer insulation, such that the effective emissivity, $\epsilon \leq 0.01$.

3.2.1.11.2 Module/structure assembly thermal interfaces. The design of the module thermal control system shall consider the following interface constraints:

- The structure assembly/module attach point temperature will be 70 ±10°F.

- Each module attachment fitting on the structure assembly will have a thermal resistance >5 hr-°F/BTU.

- The effective emittance, ϵ, of the structure assembly/module insulation barrier will be ≤ 0.02.

3.2.1.11.3 Heater power constraints. Module thermal control system heater power shall not exceed 18 watts under normal operating conditions, and 25 watts under the most severe cold operating conditions that consider predictable variations in duty cycle and heating environment as well as parameter uncertainties in thermal properties, heating environment, insulation heat loss, etc.

In the battery, battery temperature and temperature gradients shall meet the following requirements:

Maximum temperature difference between:

End and center cells $\leq 3.0^{\circ}$F

Adjacent cells	$\leq 0.5^{\circ}F$*
Top of cell and baseplate	$\leq 4.5^{\circ}F$
Batteries	$\leq \pm 2.5^{\circ}$ from the average temperature of all batteries*
Range of average battery temperatures:	30 to $55^{\circ}F$

3.2.1.12 Module structure. The module structure shall support all equipment listed in paragraph 3.1.3 and shall be capable of supporting additional equipment listed in paragraph 3.2.1.16 for modular expansion or complete redundancy.

No amplification of the vibration or acoustic environment shall be caused by the module structure which may result in degradation of the spacecraft performance.

The module structure, when mounted on the spacecraft structure, shall withstand the launch, ascent, and on-orbit loads as defined in SP-1111.

The structure shall not cause a change in alignment of the spacecraft axes by more than TBD arc seconds.

The factors of safety shall be no less than 1.00 for limit loads and 1.25 for ultimate loads except where loads may be dangerous to personnel, the ultimate loads shall be 1.50.

3.2.1.13 Useful life. The design of the EPM shall be such that wearout of any item or depletion of expendables will not occur prior to a useful life of TBD years. Useful life is defined as the operating time of the equipment counted from the time of launch vehicle liftoff.

3.2.1.14 Storage life. The EPM (except batteries) shall have a minimum storage life of 3 years. Storage life critical components may be refurbished.

Battery cells shall have a minimum storage life of 3 years. The cells shall be stored wet, discharged, and in the shorted condition.

3.2.1.15 Telemetry. Telemetry data shall be primarily limited to those functions necessary for control and operation of the power module during flight. This telemetry shall include but not be limited to functions such as bus and battery voltages and currents, the status of bistable or multimode circuits or relays, temperatures, etc. Telemetry indications of equipment status should be as direct as indication as practicable.

*Measured at similar points on each battery and cell.

3.2.1.16 Expansion capability. A capability shall be provided within the limits of module structure, size, and power available for expanding the baseline configuration to include the addition of redundant units or the addition of new components to perform additional functions.

3.2.2 Physical characteristics

3.2.2.1 Mechanical

3.2.2.1.1 Envelope. The module envelope shall be as shown in ICD 20.1.

3.2.2.1.2 Module volume. The module shall have a volume of approximately 33 cubic feet and a maximum load carrying capability of 600 pounds of equipment. Components may be mounted to the outboard facing panel, nonoutboard surfaces, and the module frame members.

The outboard facing panel may be modified with local cutouts to facilitate assembly or it may be divided into separate equipment heat sink surfaces. Internal stiffness bulkheads and/or mounting panels may be added as required. However, prior to any modifications to the module structure, the subsystem contractor shall perform a structural analysis to ensure structural integrity.

3.2.2.1.3 Module weight. The total weight of the EPM for the baseline and expanded configurations shall not exceed the listed weights.

Baseline Module	Expanded Module
TBD	TBD

3.2.2.1.4 Module center of gravity. The center of gravity of the module shall be located within TBD.

3.2.2.1.5 Attach-points. The attach-points between the module and the spacecraft shall be as shown in ICD 20.1.

3.2.2.1.6 Module/structure interface connector. The module/structure interface connector shall be provided by the system contractor and shall be mounted on the side face of the module. It is required that the connector position be maintained as specified to ensure interchangeability of modules.

3.2.2.1.7 Equipment expansion. The components shall be arranged in the module such that the expansion capability requirements of 3.2.1.16 can be accommodated with minimum impact on the baseline configuration design.

3.2.2.2 Electrical

3.2.2.2.1 Power. Total power required for the EPM shall not exceed TBD watts. Allocation of this power is as follows:

Baseline configuration requirement: TBD watts

Redundancy and expansion capability: TBD watts

Power consumption of the EPM units shall be within the power allocations in ICD 10.2.

3.2.2.2.2 Commands. Commands for controlling EPM operation shall be as listed in ICD 10.3

3.2.2.2.3 Telemetry. The EPM telemetry measurements shall be as listed in ICD 10.4.

3.2.2.2.4 Signal and power distribution. The EPM harness shall provide all intramodule electrical connections in conformance to ICD 20.5.

3.2.3 Reliability. The EPM shall be capable of performing, as specified, for at least TBD years in orbit. This shall include all redundancy incorporated including alternate and backup modes. Demonstration of compliance with these requirements shall be through reliability analysis as called out in EOS Document EOS-4.1, System Effectiveness Program Plan.

3.2.4 Maintainability. The EPM shall be designed in accordance with the requirements of MIL-STD-1472, paragraph 5.9, as implemented by EOS-4.1.

3.2.5 Environmental conditions. The EPM shall be designed to withstand or shall be protected against the worst probably combination of environments as specified in SP-11 and as implemented in SP-115.

3.2.6 Transportability. Transportability requirements shall be considered in the design of the electrical power module such that it can be transported by all standard modes with a minimum of special packing or precautionary measures, except that transportation of the battery assembly shall conform to the requirements of paragraph 5.3.2.1 (Shipping Conditions, Battery Assembly).

3.3 Design and construction

3.3.1 Parts, materials and processes

3.3.1.1 Selection of parts, materials, and processes. Selection of parts, materials, and processes (PMP) shall be to the requirements identified in MIL-STD-143 Group 1. Such selections shall be identified as standard PMP. When PMP selection cannot be made within this requirement, the item is nonstandard and justification must be made in accordance with MIL-STD-749. Control of this action shall be effected by the EOS parts, materials, and processes control board (PMPCB), with the PMPCB approving all selections, both standard and nonstandard, in accordance with EOS-4.1.

3.3.1.2 Program authorized parts list. All selections of electronic parts shall be identified and authorized by EOS-3.3-1. This list will reflect all PMPCP electronic part selections.

3.3.1.3 Program authorized materials list. All selections of materials shall be identified and authorized by EOS-3.3-2. This list will reflect all PMPCB material selections.

3.3.1.4 Program authorized processes list. All selection of processes shall be identified and authorized by EOS-3.3-3. This list will reflect all PMPCB process decisions.

3.3.1.5 Dissimilar metals. To avoid electrolytic corrosion, dissimilar metals shall not be used indirect contact unless protection against corrosion has been provided in accordance with MIL-E-8983 and MIL-STD-454, Requirement 16.

3.3.1.6 Magnetic materials. Magnetic materials shall be used only if necessary for equipment operation. Those magnetic materials which are used shall have minimum permanent, induced, and transient magnetic fields.

3.3.1.7 Fungus-inert materials. Materials which are fungus-inert, in accordance with MIL-E-8983 and MIL-STD-454, Requirement 4, shall be used.

3.3.1.8 Flammable toxic and unstable materials. Flammable, toxic, and unstable materials shall not be used.

3.3.1.9 Finish. The surface of each component shall be adequately finished to prevent deterioration from exposure to the specified environments that might jeopardize fulfillment of the specified performance. The finish shall also meet the applicable bonding requirements. Thermal properties of the finishes used on the component shall be compatible with the requirements given in detail in applicable equipment specifications. The requirements for finishes identified in MIL-E-8983 shall be implemented.

3.3.1.10 Outgassing. Low outgassing polymeric materials shall be used where sensitive thermal control and other surfaces are in direct line of sight and where temperature differences can exist between such surfaces.

3.3.2 Electromagnetic radiation

3.3.2.1 EMC/EMI requirements. The electrical power module, and all internal units, equipment and/or components comprising a part thereof, shall be designed for compliance with the radiated emission and susceptibility requirements of NASA/JSC Specification SL-E-0002 as modified or amended by EOS-3.3-4 and EOS-3.3-5. The conducted emission and susceptibility requirements of the aforementioned documents are applicable only at the module-to-spacecraft structure interface. EMI/EMS levels at interfaces within the module shall be controlled to the extent necessary to ensure self-compatibility of the module subsystem with a safety margin of at least 6 dB.

3.3.2.2 Electrical bonding

3.3.2.2.1 Structural bonds. All metallic members of the basic electrical power module radiator panel and support structure shall be electrically continuous, equi-potential ground reference plane. The maximum allowable DC impedance across any one structural joint shall be 2.5 milliohms with an overall design goal of 10 milliohms, or less, between any two diametrically opposite points on the module. The general methods of MIL-B-5087, Class R, may be used for implementation of these requirements.

3.3.2.2.2 Equipment mounting pads. Surfaces on the module radiator panel and structure which are intended for unit, equipment, or component mounting shall be free of paint, anodize, or other nonconductive finishes. The maximum DC impedance between the baseplate of any electrically active unit, equipment or component and the module radiator panel and structure shall be 2.5 milliohms.

3.3.2.2.3 Electrical connectors. All interface electrical connectors both plug and receptacle, which form a part of the unit and cable RF shielding system within the module shall have electrically conductive body shells, free of nonconductive finishes. They shall also have provisions for terminating (grounding) the shields on the interconnecting electrical harness. The maximum DC impedance between the shield termination point and the baseplate of the parent unit shall be 2.5 milliohms.

3.3.2.2.4 Electrostatic bonds. Electrically passive components or appurtenant structures which are attached to the basic module structure through thermal isolators shall be provided with electrostatic bonds having a DC impedance of 1.0 ohm or less. The general methods of MIL-B-5087, Class S, may be used for implementation of this requirement.

3.3.2.3 Electrical systems grounding

3.3.2.3.1 Primary DC power. The electrical power module shall provide the single-point structural ground for the primary DC power distribution subsystem. This subsystem will be DC isolated from structural grounds in all user modules as well as in the solar array. The physical location of the ground point may be within, or immediately adjacent to the battery assembly. Interconnecting primary power wiring between units and/or assemblies of the electrical power module shall be unshielded, twisted pairs.

3.3.2.3.2 Secondary DC power. Secondary DC power distribution networks shall, in general, use multiple point grounding within the electrical power module with returns through the module radiator panel and/

or support structure. These networks shall be initially grounded adjacent to the secondary transformer winding in the power converter and at each user element.

3.3.2.3.3 Secondary AC power. Secondary AC power networks shall use single point grounding with a two-wire, twisted shielded pair distribution. The ground point may be either at the source or load end, whichever is shown by circuit analysis to be most beneficial to compatible module operation. Structural returns shall not be used for secondary AC power.

3.3.2.3.4 Intramodule signal/control circuit (high level). All high level (≥5 volt logic, bilevel or analog) signal or control circuits which do not exit the electrical power module shall be multiple point grounded at both the source and load end of each circuit branch to the unit or component chassis by the shortest most direct path. Circuits sharing space on a common printed circuit board should not share common grounding traces on the board or common hardwire jumpers to chassis ground logs. Preferably, each such board should have a dedicated ground plane layer to which all components requiring ground returns can be directly connected. This ground plane should, in turn, be directly bonded to unit chassis or frame through grounding pads at each hold-down fastener.

3.3.2.3.5 Intermodule signal/control circuits (high level). All high level signal or control circuitry which exits the electrical power module shall be grounded at the final driving element. Two-wire, twisted shielded pair distribution shall be used for each such circuit between the source unit and the module-to-spacecraft interface connector. The load elements in the external module will be DC isolated from structural grounds. Any signal or control circuitry which enters the electrical power module shall be DC isolated from chassis/case/structure ground by a minimum of 1 megohm resistance. Any such circuits shall also be provided with two-wire, twisted shield pairs between the interface connector and the load unit.

3.3.2.3.6 Analog circuits (low level). Any low level (<5 volts) analog circuits, which are shown by circuit analysis or test to be sensitive to circulating currents in the module or spacecraft structure, shall be single-point grounded either at the source or load element, whichever is most appropriate for the circuit under consideration. Wherever possible, balanced differential circuitry should be used. In the case of low level circuits which enter or exit the electrical power module, the location of the circuit ground point shall be coordinated with the systems integration contractor.

3.3.2.3.7 Data bus. The command and telemetry data bus system shall be differentially driven and balanced to structural ground in the communication and data handling module. This system shall be transformer-coupled at each remote terminal. Each individual data bus wire entering the electrical power module shall be DC isolated from chassis/case/structure by a minimum of 1 megohm resistance.

3.3.2.3.8 Wire shields. External shields shall be provided for all interconnecting wires between units, equipment or components in the electrical power module and between each input/output connector and the module-to-spacecraft interface connectors except for input and output primary DC power lines. In general, these shields shall be multi-point grounded at each end and at each intermediate interface. An exception to this rule will be allowed for low level analog circuitry where single-point shield grounding may be necessary. If possible, such circuitry should be provided with two mutually-isolated shields, the inner shield being single-point grounded and the outer shield multi-point grounded.

3.3.2.3.9 EMI filter components. High performance EMI filters will be required at the primary DC power input and return terminals of each power converter unit or sub-unit in the electrical power module to ensure compliance with the electromagnetic interference requirements of the applicable specifications. These filters should have two stages: (1) an AF ripple filter stage balanced line to line; and (2) a pair of RF feedthrough filters bulkhead-mounted behind the input connector. The combined AF/RF filter circuit should be designed for the minimum capacitance from either line to chassis necessary to achieve compliance with the specifications. Excessive line-to-ground capacitance will tend to negate the beneficial effects of the twisted pair wiring used in the primary DC power harness. Similar filtering may be required at the input and output terminals of the pulsewidth switching regulator assembly; however, because of fault isolation requirements, line-to-case feed-through RF filters should not be used.

3.3.3 Nameplates and product marking. Each unit shall be identified in accordance with MIL-STD-130 as implemented by EOS-3.3-6. Where practical, a minimum character height of 0.09 inch shall be used. Marking shall include the manufacturer's name or initials, assembly part number and revision status, assembly serial number, contract number, and others as delineated in the component equipment specifications.

3.3.4 Workmanship. The EPM and its component equipment shall be constructed, finished, and assembled in accordance with MIL-STD-454, Requirement 9, and the specifications and drawings specified herein.

3.3.5 Interchangeability and replaceability. The electrical power module shall be designed to permit removal and replacement of components with a minimum of disturbance of associated or adjacent equipment. Design for service and access shall conform to the principles and requirements of MIL-STD-470.

3.3.6 Safety. The electrical power module shall be designed to meet or exceed the requirements of EOS-3.3-7, as implemented by EOS-4.1, System Effectiveness Program Plan. The design criteria include but are not limited to those set forth in MIL-STD-882.

3.3.7 Human performance/human engineering. MIL-STD-1472A, "Human Design Criteria for Military Systems," shall be used for the design of man/machine interfaces.

3.4 Documentation. Documentation shall be as defined in EOS-3.3-8.

4. QUALITY ASSURANCE PROVISIONS

4.1 General. The quality assurance program controls shall be in accordance with the requirements of MIL-Q-9858 as augmented by EOS-4.1.

4.1.1 Responsibility for inspection and test. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified herein. Except as otherwise specified, the supplier may use his own facilities or any commercial laboratory acceptable to the government. The government reserves the right to perform any of the inspections set forth in the specifications where such inspections are deemed necessary to ensure that supplies and services conform to prescribed requirements.

4.2 Quality conformance inspections

4.2.1 Category I tests

4.2.1.1 Development tests. Development tests shall be conducted to verify the module performance parameters, proper interfacing between components of the module, and proper interfacing between the module and other spacecraft modules. The development tests shall be conducted at the unit and/or integrated module levels. The type of developmental evaluation or test to be applied to verify satisfaction of the requirements defined in this specification are outlined in Table 1. The tests shall be conducted in accordance with EOS-4.2

4.2.1.2 Qualification tests. Module qualification shall be accomplished by means of qualification tests on the individual units and/or as a normal consequence of having the units installed in the qualification test space vehicle during its qualification testing. The types of qualification evaluation or test to be applied to verify satisfaction of the requirements of this specification are outlined in Table 1. Qualification test verification methods and requirements shall be as defined in the EOS-4.2.

4.2.1.2.1 Components. As a minimum, the following types of component qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, are measured before and after the environmental tests.
- Random vibration. Each component will be subjected to a random vibration from 20 to 2000 Hz in all three axes while electrically powered in the maximum normal load condition. Performance will be continuously monitored to detect failures (either hard or trends).
- Thermal vacuum. Thermally cycle the component beyond expected orbital temperatures while at an orbital vacuum condition. The component is operating continuously and detailed performance parameters are measured at each temperature extreme.

Table 1. Verification Cross Reference Index

VERIFICATION CROSS REFERENCE INDEX									
LEGEND: N/A - Not Applicable; I - Inspection; A - Analysis; S - Similarity; T - Test									
SECTION 3 REQUIREMENTS	Not Applicable	Acceptance			Qualification				SECTION 4 VERIFICATION REQUIREMENTS
		I	A	T	I	A	S	T	
(TBD)									

SAMPLE

- Electromagnetic interference (EMI) and susceptibility (EMS). Engineering EMI and EMS data are measured to provide diagnostic information for the module EMI/EMS qualification.

4.2.1.2.2 Module. As a minimum, the following types of module qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, and electrical interface characteristics are measured before and after the environmental tests.

- Acoustics. Acoustics testing will be performed with the module mounted in a receptacle which provides acoustics shielding similar to the Observatory structure. The module will be heavily instrumented to verify design adequacy of interconnections (electrical and mechanical) between the module and other Observatory elements. The test data will be used to confirm estimates of the vibration environment for components mounted on radiators and establish procedures for acceptance testing. The module will be electrically powered and continuously monitored via the data bus to detect failures and out-of-tolerance performance trends.

- Thermal vacuum. The objectives of the module thermal qualification test are to determine: 1) maximum and minimum operating temperatures for the module for worst-case environments and operating duty cycles, 2) temperature levels and temperature variations of the module adjacent to the interface fittings, and 3) heater power requirements for cold case conditions.

  The test will be conducted in a thermal-vacuum chamber with liquid nitrogen cold walls. The module will be mounted on a fixture simulating the spacecraft frame. The properties of this fixture shall permit its temperature to be held constant at any level between 0 and 100°F.

  A heating source for the radiators will approximate the absorbed flux of the external environment. This can be done with electrical heaters, infrared lamps, or other techniques where the absorbed heating can be determined accurately.

  - Power bus and data bus will be tested in excess of their operational limits to determine design margins and compliance with the interface specification.

  - Detailed performance data will be measured to determine module specification values.

  - Thermistor/heater control and calibration will be determined.

- EMI and EMS. EMI measurements will determine the levels radiated on each module electrical interface and verify there is adequate design margin to ensure non-interference with other modules. EMS measurements will verify that each module has adequate susceptibility margin to prevent performance degradation resulting from other module allowable radiation levels. The general test methods of MIL-STD-462, as modified or amended by EOS-3.3-5, shall be used during this test program except that, wherever practical, spectrum analyzers or other forms of rapid display and analysis equipment may be used in lieu of equipment requiring manual tuning and data collection.

4.2.1.2.3 Inspection sequence. The inspection sequence shall be governed by the following:

- Examination of the module shall be performed prior to functional testing.

- Functional tests shall be performed prior to, during, where appropriate, and following environmental testing. The functional test may serve as the functional testing to be performed prior to the next environmental test.

- Environmental testing may be performed in any sequence.

4.2.1.2.4 Failure criteria. The module shall exhibit no failure, malfunction or out-of-tolerance performance degradation as a result of the examinations and test specified herein. Any such failure, malfunction or out-of-tolerance performance degradation shall be cause for rejection.

4.2.2 Test conditions. Unless otherwise specified, the conditions under which all inspections are accomplished shall be specified below:

4.2.2.1 Atmospheric and environmental. All examinations and tests shall be conducted under the local prevailing pressure at the time of test, at a temperature of 75 $\pm 5^{\circ}F$ and at a relative humidity of 55 percent or less.

4.2.2.2 Deviations of atmospheric conditions. When tests are performed with atmospheric conditions substantially different from the specified values, proper allowances for changes in instrument readings shall be made to compensate for the deviation from the specified conditions.

4.2.3 Equipment warmup time. The equipment warmup time shall be less than 1 minute.

4.2.4 Temperature stabilization. Temperature stabilization shall have been achieved when temperature of the equipment mounting structure varies not more than $5^{\circ}F$ during a period of 15 minutes.

4.2.5 Measurements. Measuring instruments used to determine functional parameter values (such as voltage, frequency, current, flow, pressure, etc.) shall indicate true values with an accuracy determined by the tolerance allowed for the parameter variation itself, such that the measuring instrument shall not introduce an uncertainty greater than 10 percent of the allowable variation of the measured parameter. However, no such measurement accuracy shall be required to exceed 0.5 percent of the required value of the parameter unless otherwise specified.

Except as specifically noted in the inspection methods, tolerances for environmental test conditions shall be defined as follows:

Temperature: $\pm 5^{\circ}F$

Barometric pressure: ±10 percent

Relative humidity: +5, -10 percent

Time: ±5 percent

Vibration amplitude (sine): ±10 percent

Vibration frequency: ±2 percent

4.3 Category II tests. (Not applicable)

4.4 Analyses. The following requirements of Section 3 shall be verified by review of analytical data:

3.2.3 Reliability. To be verified by analysis in accordance with Section TBD of EOS-4.1.

3.3.6 Safety. To be verified by analysis in accordance with Section TBD of EOS-4.1.

5. PREPARATION FOR DELIVERY

5.1 General. The electrical power module as specified herein shall be protected from degradation by environments anticipated during shipment, handling, and storage. Standard commercial packaging practices are acceptable provided they fulfill these requirements.

5.2 Preservation and packaging

5.2.1 Cleaning. The EPM shall be clean and free of contaminants which might impair its use prior to packaging.

5.2.2 Attaching parts. When attaching parts, such as nuts, bolts, washers, etc., accompanying the EPM, they shall be preserved, bagged, appropriately identified, and attached to or adjacent to the fitting for which they are intended.

5.2.3 Electrical connectors. All electrical connectors shall be capped with protected dust caps. Caps used shall be a friction fitting or threaded type which do not require tape or a mechanical device to secure.

5.2.4 Critical surfaces. External machined surfaces and mounting surfaces of the EPM shall be protected with protective pads. Materials used for pads shall not cause item deterioration.

5.2.5 Wrapping. The EPM shall be wrapped or bagged using anti-static polyethylene film.

5.2.6 Cushioning. When required for protection, the EPM shall be cushioned using a suitable resilient foam cushioning material such as polyethylene, polyurethane, or polystyrene.

5.3 Packing. The EPM shall be packed in a suitable shipping container designed for one item only. The shipping container used shall provide protection to the item against corrosion, deterioration, and damage during shipment from the source of supply to the receiving activity. Containers shall comply with applicable tariffs and regulations for particular modes of transportation when so shipped.

5.3.1 Storage conditions. The EPM (exclusive of the battery assembly) shall not be adversely affected by storage within its container at temperatures between 60°F and 90°F and relative humidities of 60 percent or less.

5.3.1.1 Storage conditions, battery assembly. Each battery shall be stored wet, discharged and in the shorted condition by the following method:

(a) Each battery shall be discharged below 0.1 volt by clipping a 1-ohm resistor across the cell terminals.

(b) Remove the 1-ohm resistor and immediately short the cell by wrapping an uncoated copper wire around the cell terminals.

(c) Each battery shall be placed in a polyethylene bag and an inert drying agent shall be added to exclude moisture. The bag shall be heat sealed. The cell serial number shall be clearly visible from the outside of the bag.

(d) Each battery shall be packaged in a manner to avoid damage during shipment.

(e) The batteries shall be stored in a manner such that the temperature remains between 32 and 68°F.

5.3.2 Shipping conditions. The EPM (exclusive of the battery assembly) shall be capable of withstanding the following environments:

Temperature:	+160°F in an unsheltered area (125 +35°F due to solar radiation) and -40°F in accordance with restricted air transport criteria
Humidity:	Up to 100 percent in an unsheltered area

Rough handling:	Capable of withstanding and physically protecting the unit from rough handling during shipping by common carrier

5.3.2.1 Shipping conditions, battery assembly. Each battery shall be stored, for short duration, and shipped in a manner such that the temperature does not exceed 89°F. A temperature change indicator shall be placed in each shipping container to verify that this temperature is not exceeded. For periods of storage in excess of one week the cells shall be stored in a manner such that the temperature remains between 32 and 68°F.

5.4 Marking for shipment. Each EPM and shipping container shall be marked with the following:

(a) Item nomenclature

(b) Systems part number

(c) Contractor or purchase order number

(d) Manufacturer's name

(e) Manufacturer's part number and serial number (on item container only)

(f) Quantity

(g) Date of manufacture (on item container only)

(h) Fragile — Handle With Care (when applicable)

(i) Space Vehicle Material — Do Not Open In Receiving Or Receiving Inspection (when applicable — shipping container only)

(j) Actual weight

5.4.1 Documentation. All required reliability and test documentation such as test reports, certifications, shipping invoices, etc., shall be either packed in the EPM container or attached to the exterior surface of the shipping container. Attachment shall be such as to preclude loss of these data during handling and shipment by common carrier.

6. NOTES

6.1 Definition of spacecraft configuration

6.1.1 Minimum redundancy configuration. The minimum redundancy configuration is defined as the spacecraft configuration which contains the minimum redundancy of units necessary to ensure that no plausible single-point failure will prevent Observatory retrieval by the

Space Shuttle System. For purposes of this specification this configuration is identified as the baseline spacecraft configuration.

6.1.2 Nominal redundancy configuration. The nominal redundancy configuration is defined as the spacecraft configuration which includes standby redundant units for most of the electronic assemblies to provide a "typical" redundancy level for long-life spacecrafts.

TITLE

PRIME ITEM DEVELOPMENT SPECIFICATION
FOR THE

ATTITUDE DETERMINATION MODULE

DATE 20 SEPT 1974 NO. SP-1114

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

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SPECIFICATION SP-1114

ATTITUDE DETERMINATION MODULE

1. SCOPE

1.1 Scope. This specification establishes the functional requirements for performance, design, development, test and qualification of the attitude determination module (ADM) for the Earth Observatory Satellite. The ADM provides on-board attitude reference and data for post facto attitude determination.

2. APPLICABLE DOCUMENTS

2.1 Government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

SPECIFICATIONS

NASA

SP-11	Observatory Segment Specification
SP-115	Observatory Environmental Specification
SP-1111	Structure Assembly Specification
SP-1112	Communication and Data Handling Module
SP-1113	Electric Power Module

Military

MIL-B-5087	Bonding, Electrical, and Lightning Protection for Aerospace Systems
MIL-E-8988A	Electronic Equipment, Aerospace Extended Space Environment, General Requirements
MIL-Q-9858	Quality Program Requirements

STANDARDS

Military

MIL-STD-143B	Standard and Specifications, Order of Precedence for Selection
MIL-STD-454C	Standard General Requirement for Electronic Equipment
MIL-STD-749B	Preparation and Submissions of Data for Approval of Nonstandard Electronic Parts

MIL-STD-1472A	Human Engineering Design Criteria for Military Systems, Equipment, and Facilities
MIL-STD-882 15 July 1969	System Safety Program for Systems and Associated Subsystems and Equipment, Requirements for
MIL-STD-462	EMI/EMS Test Methods
MIL-STD-130	Name Plates
MIL-STD-470	Maintainability Program Requirements (for Systems and Requirements)

OTHER PUBLICATIONS

NASA

SL-E-0002	Electromagnetic Compatibility Control Plan
NASA STDN 101.1 X-560-63-2	STDN User's Guide, Aerospace Data Systems Standards
- - - -	Electromagnetic Compatibility Requirements for Space Systems
EOS-3.3-1	Program Authorized Parts List
EOS-3.3-2	Program Authorized Materials List
EOS-3.3-3	Program Authorized Processes List
EOS-3.3-4	Electromagnetic Compability Control Plan
EOS-3.3-5	EMI/EMS Limits and Test Methods
EOS-3.3-6	Marking of Parts and Assemblies

2.2 Non-government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. If no revision or date is shown, the latest released issue of the applicable document shall apply. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

DRAWINGS

60.5	AD Module Wire List
10.4	Satellite Telemetry Allocation, ICD

10.3	Satellite Command Allocation, ICD
10.2	Satellite Primary Power Allocation, ICD
60.1	AD Module Envelope, ICD
TBD	AD Module Assembly Drawing

OTHER PUBLICATIONS

EOS-3.3-7	Safety Design Criteria
EOS-3.3-8	Configuration Management Plan
EOS-4.1	System Effectiveness Program Plan
EOS-4.2	Integrated Test Plan

3. REQUIREMENTS

3.1 General

3.1.1 Function. The ADM shall contain the spacecraft attitude sensors and associated signal processing electronics required to provide three-axis attitude steering reference for attitude control in conjunction with the actuation module and on-board computer. The ADM shall supply telemetered data for ground determination of spacecraft attitude.

The ADM shall provide the attitude steering reference for control of the spacecraft during sun acquisition, earth hold mode, orbit injection/ trim mode, normal mode, and safe mode.

The ADM must be capable of supporting earth, sun, and inertially oriented missions. The orbital altitude range is 300 to 22,000 nautical miles. The ADM configuration is shown in Figure 1.

3.1.2 Operation. The ADM shall provide attitude and rate information for spacecraft attitude control and attitude determination during the following modes of operation.

3.1.2.1 Sun acquisition mode. The sun acquisition mode (Figure 2) consists of reorienting the spacecraft to align the -z axis with the sun for maximum solar array power and a benign thermal condition. The pointing reference is a sun sensor(s) with 2π steradian field-of-view. During this mode spacecraft rates will be reduced to less than ± 0.03 deg/sec from separation induced rates of up to 1 deg/sec per axis and random initial attitude.

3.1.2.2 Earth hold mode. The earth hold mode (Figure 3) consists of reorienting the spacecraft yaw (+z) axis to point at the earth prior to firing of the orbit injection engine. Orientation around the yaw axis is controlled to align the orbit injection engine into the orbit plane. This

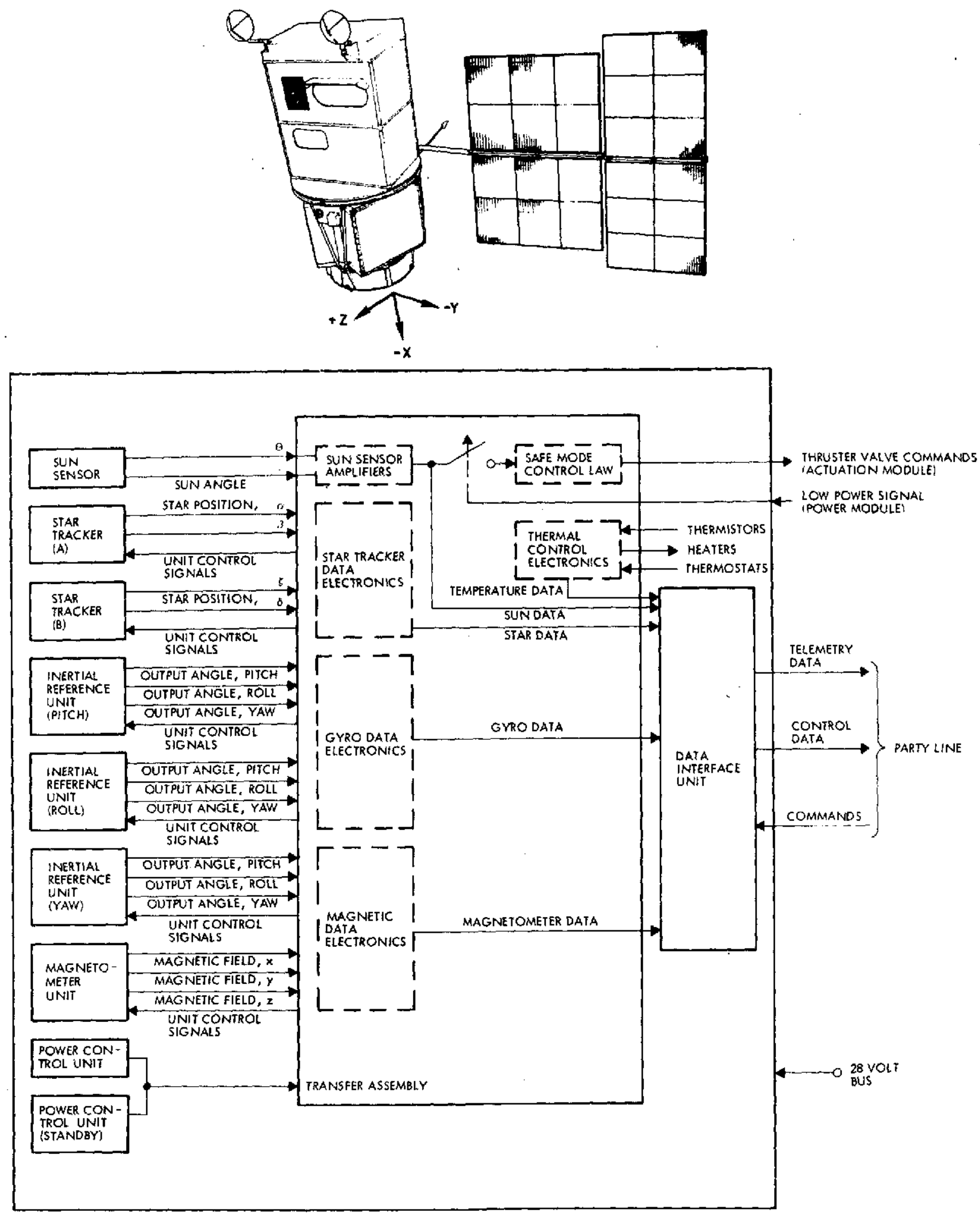


Figure 1. Attitude Determination Module Block Diagram

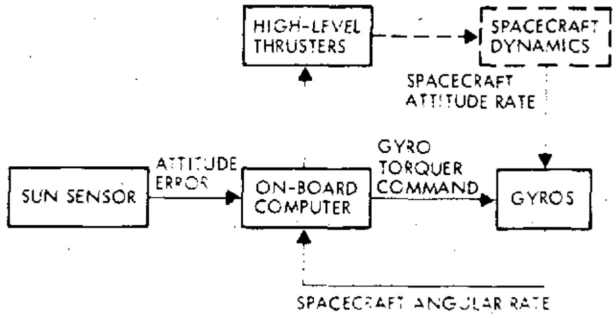


Figure 2. Sun Acquisition Mode

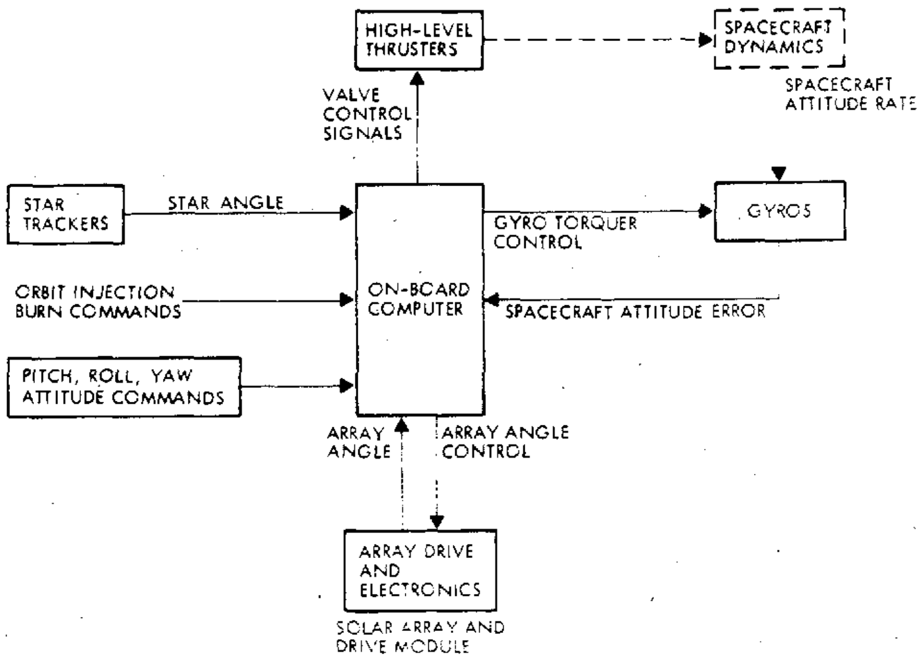


Figure 3. Earth Hold Mode

attitude is held under gyro control in response to pitch, roll, and yaw attitude ground commands derived from the ground computation of spacecraft altitude.

3.1.2.3 Orbit injection/trim mode. The orbit injection/trim mode (Figure 4) consists of maintaining the spacecraft attitude held fixed at the desired orientation while orbit injection or orbit adjust engine burns are executed. The spacecraft attitude is held fixed at the desired orientation by means of the high level thrusters and the gyro reference. During subsequent orbit trim burns, the spacecraft attitude is controlled in the same manner.

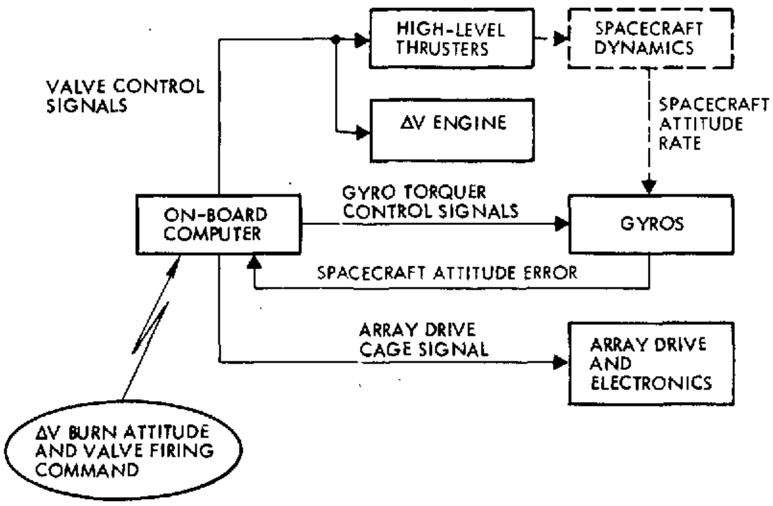


Figure 4. Orbit Injection/Trim Mode

3.1.2.4 Normal mode. The normal mode (Figure 5) requires long term, accurate spacecraft attitude control as dictated by payload requirements. Specific mission may require earth, stellar inertial, or sun pointing. The normal mode attitude reference is a body-fixed rate integrating gyro and body-fixed star trackers for periodic update and collection of gyro drift.

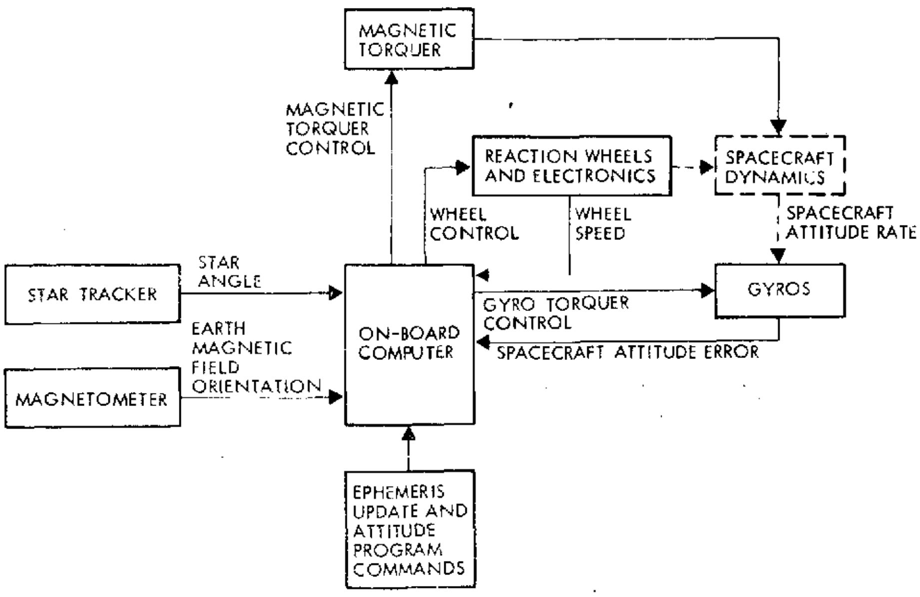


Figure 5. Normal Mode

3.1.2.5 Safe mode. The safe mode (Figure 6) is a back-up mode which consists of automatically reorienting the spacecraft for maximum sun illumination of the solar array. In this mode the solar array drive is rotated to position the array surface to be parallel to the spacecraft (+x) axis and the spacecraft is reoriented to point the (-z) axis at the sun. The safe mode is entered when logic in the power module gives a persistent indication of negative energy balance. Attitude control is by means of the sun sensor and the low level thrusters. Special-purpose safe mode attitude control electronics replace the on-board computer for attitude control and the gyros, reaction wheels, and magnetic torquers are disabled.

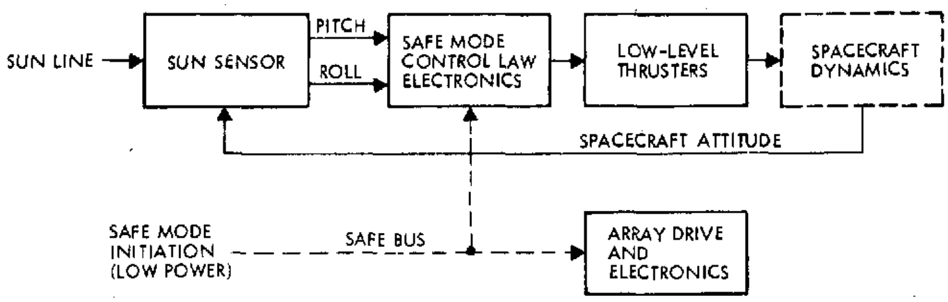


Figure 6. Safe Mode

3.1.3 Government-furnished module equipment. Certain items of module equipment which are identical or nearly identical in function will be fabricated by the integration contractor and provided as GFE to the remaining module contractors. This equipment shall include:

Item	Reference
Data interface unit	3.2.1.3.4 of SP-1112
Power control unit	3.2.1.9 of SP-1113

All module structures and integration of the modules will be provided by the integrating contractor.

3.1.4 Equipment list. The ADM shall incorporate the following equipment and quantities.

Component Name ADM Equipment Group	Quantity per Module
Inertial reference unit	6
Star tracker	3
Star tracker shade	3

Magnetometer	1
Sun sensor	1
Transfer assembly	2
Data interface unit	2
Power conditioning unit	2
Harness	1

3.2 Characteristics

3.2.1 Performance

3.2.1.1 Electrical interface requirements. The module electrical interfaces are illustrated in Figure 7.

3.2.1.1.1 Data bus interface. A 4-wire, full duplex, party line data bus shall be used for intermodule data transfer. One pair of wires shall be used for data from the ADM modules (supervisory line), the other pair shall be used for data to the ADM module (reply line). Characteristics of the data bus signals shall be as defined in SP-1112.

3.2.1.1.2 Primary power. The ADM shall receive primary power from the power module on two redundant lines as follows:

Voltage: 28 ±7 volts

Current: TBD amperes maximum

Transients:

Load switching: ±1 volt (100 ms or less)

Fault correction: Down to +20 VDC or up to +39 VDC for 100 ms or less

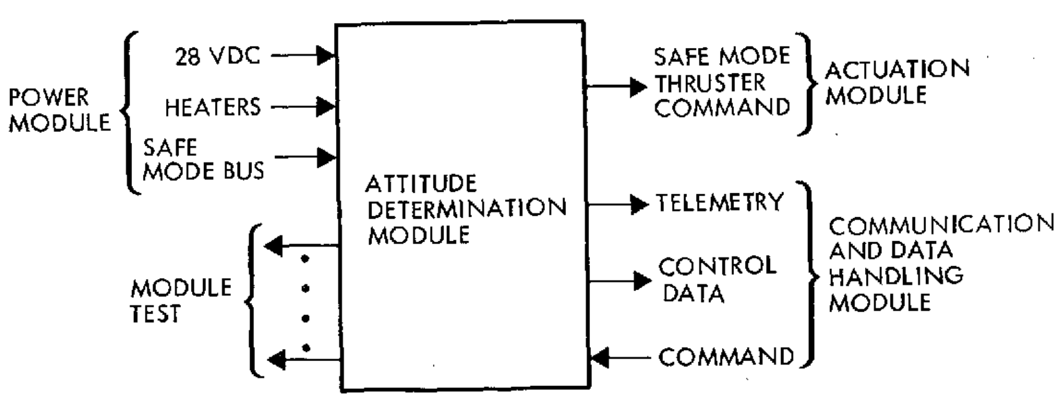


Figure 7. Attitude Determination Module — Electrical Interface

3.2.1.1.3 Module heaters. A connection to heater power shall be provided. The line shall operate at 28 volt nominal with TBD amperes maximum current.

3.2.1.1.4 Safe mode bus. A connection to the safe mode bus shall be provided. Normal mode shall be initiated by a +5 volt level (10 ma maximum module sink current). Safe mode shall be indicated by a 0 volt level.

3.2.1.1.5 Module test connector. A test connector shall be provided for module test of the following signals: TBD

3.2.1.2 Module definition. The ADM shall consist of a single enclosure containing the sensors and associated electronics used for preforming the attitude sensing functions delineated in paragraph 3.1.2. The equipment contained with the ADM is described below.

3.2.1.2.1 Inertial reference unit (IRU). The IRU shall contain a single-degree-of-freedom gas bearing gyro and associated torque rebalance loop. Six IRU's, two per spacecraft axis, shall be installed in the ADM with three being normally used at any one time.

Internal switching shall control the application of 1) primary DC input power, 2) spin motor power, and 3) torque-rebalance loop and heater power, separately for each gyro. Also, if torque-rebalance loop cross-strapping is used, the necessary switching logic shall be incorporated.

Gyro temperature control loops shall be included, if required. Each IRU assembly (gyro and electronics) shall be plug-in type and interchangeable.

3.2.1.2.1.1 Operating modes. Each IRU shall have only the following modes

(a) Power off. This mode is entered on receipt of the OFF command and persists until the receipt of the STANDBY or ON commands. With the exception of the DC power required for the command logic circuitry, no other power enters or leaves the channel.

(b) Standby. This mode is entered on receipt of the STANDBY command and persists until the receipt of the ON or OFF commands. In this mode only the torque rebalance loop and the gyro heater, if any, are energized. Input DC power is applied to the channel.

(c) Operate. This mode is entered on receipt of the ON command and persists until the receipt of the STANDBY or OFF commands. In this mode the gyro spin motor, torque rebalance loop, and gyro heater, if any, shall be energized. The channel shall be completely operational.

(d) Warmup time. Each channel of the IRU shall meet the requirements of this specification within 90 minutes of the application of spin motor power. The gyro spin motors shall reach synchronous speed within 120 seconds of the application of power.

3.2.1.2.1.2 Gyro-torquer loop characteristics

(a) Configuration. The gyro loops shall be torque rebalanced, using either digital torque rebalance electronics or analog torque rebalance electronics with analog to digital output converters.

(b) Frequency response. The closed loop frequency response shall not be less than 10 Hz (-3 dB).

(c) Maximum input rate. Each channel shall meet all requirements of this specification with input rates of ±1.0 deg/sec, about any axis, including the effects of gyro output axis rotation to 1.0 deg/sec and shall meet reduced accuracy requirements, as specified for rates of ±1.0 to ±2.5 deg/sec. The RGA shall meet all the requirements of this specification after exposure to slewing rates of 30 deg/sec about each of the input axes, in the operation mode.

(d) Stability.

- Gain margin: worst case greater than 6 dB
- Phase margin: worst case greater than 30 deg

3.2.1.2.1.3 Attitude/measurement

(a) Quantization and range. Each gyro channel shall measure angular displacement (incremental attitude) about the gyro input axis with a quantization of 0.1 arc sec (nominal) per count over the input range of ±2.5 deg/sec.

(b) Data output. The output from each gyro loop shall be a 16-bit digital data word in signed, two's complement form. The data shall be read out serially, least significant bit first, from one gyro at a time under external control. Each readout shall occur every 320 msec. Each gyro loop will be provided with a Read pulse at least 1.1 msec before a readout begins. For digital loops each data word shall represent an integral number of limit cycle periods. The digital data output shall saturate, but not overflow, for rates that exceed the capacity of the 16-bit data word.

3.2.1.2.1.4 Electrical outputs. For each gyro, the following signals shall be available on external ADM connectors for operational and test/telemetry usage.

(a) Attitude data. The attitude data output shall be as described in paragraph 3.2.1.2.1.3 of this specification.

(b) Analog rate signal. The range of the analog rate signal shall be 0 to 5 volts with a scale factor of 1 volt/deg/sec. The zero rate point shall be 2.5 volts and the bandwidth shall be less than 1 Hz. This output shall be configured such that the output voltage shall not exceed ±10 volts in the event of a failure.

(c) Spin motor monitor (SMM). The spin motor monitor output shall be a bilevel signal, which indicates whether the gyro spin motor has reached synchronous speed. The amplitude shall be 5.0 VDC ± TBS when the spin motor is at synchronous speed and 0.0 VDC ± TBS when the spin motor is not at synchronous speed.

(d) Gyro temperature. In addition to the temperature sensors required for gyro stabilization, a gyro temperature monitoring sensor shall be provided. This temperature sensor shall be incorporated into an electrical circuit whose output voltage shall be between 0 and 5 volts, depending on the gyro temperature. The gyro temperature output scale factor shall be 20° F/volt and 0.0 volts shall correspond to 80° F. In the event of a failure, the circuit shall be configured such that this output voltage can never exceed ±10 volts.

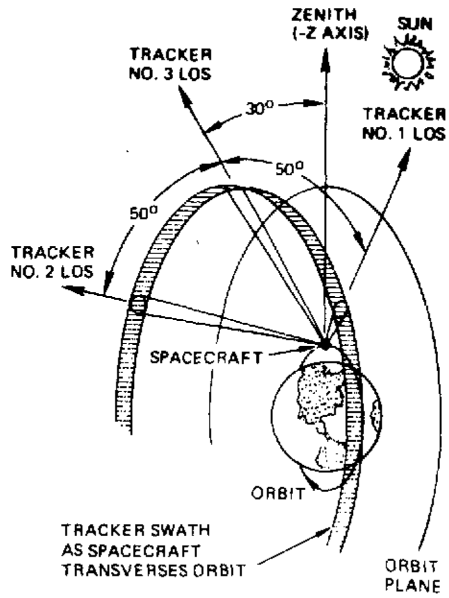
(e) Gyro preamplifier output. The gyro preamplifier output voltage shall be available for test and monitoring. This output shall be buffered (protected) such that a failure cannot occur if this output is permanently shorted to ground. Fuses shall not be used for protection.

(f) Precision reference voltages. All precision reference voltages shall be available for test and monitoring. These outputs shall be buffered (protected) such that a failure cannot occur if these outputs are permanently shorted to ground. Fuses shall not be used for protection.

3.2.1.2.2 Star tracker assembly (STA). The STA is used to provide precise measurements of the positions of start relative to STA coordinates. Three STA's shall be installed in the ADM with the orientation as shown in Figure 8. The trackers are oriented such that their lines of sight lie on a cone which makes an angle of 60 degrees to the positive pitch (y) axies. Tracker number 3 has its line of sight in the (-Z, -Y) plane. Trackers 1 and 2 are located 50 degrees around the cone from Tracker 3. In a two-tracker configuration (minimum) Tracker 3 is deleted. The three STA's are required to operate simultaneously.

The STA shall consist of optics for collecting the energy from the stars in the field of view (FOV) and forming their image. The image shall be presented to a photodetector which generates suitable signals The signals shall be conditioned for presentation as outputs as specified herein. The unit shall be generally self contained and providing a bright object sensor and shutter to protect its detector from damage due to excess illumination. The glare shields which are required for the applications of the STA to the spacecraft shall be incorporated within the ADM.

The STA shall be capable of searching for stars within its field of view; acquiring a star in accordance with star brightness established by command; tracking the acquired star until the receipt of a suitable command, or until the star leaves the FOV; and then repeating the sequence. The unit shall contain means to prevent damage to the STA in the event a bright object approaches the field of view. The unit shall provide outputs indicating the operating mode, the rectangular coordinates (U, V) in the FOV of a star which is being tracked, the magnitude of the star being tracked; and a variety of status and telemetry signals.



- STAR TRACKER FIELD OF VIEW: 8 × 8 DEG
- STAR MAGNITUDE: +2 TO +6

Figure 8. Star Tracker Geometry

3.2.1.2.2.1 Search performance. The STA shall search a FOV at least $-4° \leq U \leq 4° \leq V \leq 4°$.

(a) Search mode. The STA shall commence searching in accordance with the requirements of Table 1. If the search is executed as a raster scan as shown in Figure 9, the U axis shall be aligned with the high speed scan (equivalent to the horizontal sweep in television), and the V axis shall be aligned with the slow speed scan.

(b) Frame time. The time to search the field of view in the absence of any star shall be no greater than 4 seconds. The STA shall be able to acquire and track with vehicle scan rates of ω_z = 0.2 to 0.4 deg/sec.

(c) Acquisition. When a star of adequate intensity is encountered, the STA shall generate an acquisition signal. Adequate intensity is defined as intensity in excess of a commanded threshold. This signal shall stop the search scan and enable the track mode within 50 milliseconds of star encounter.

3.2.1.2.2.2 Tracking requirements. The STA shall track a star which has been acquired anywhere within the search FOV.

Table 1. Search Mode Requirements

Prior Status	Event	Search Mode Action
STA OFF	ON command received	STA commences searching at $U \leq -4°$, $V \geq 4°$ within 1 second of application of power
STA ON in search mode	SEARCH command received	None
STA ON in track mode	Search command received or STA loses track or star leaves FOV ±5 percent	STA commences searching at U_O, V_O where U_O, V_O are the coordinates of the point at which the star being tracked was last observed. Acquisition shall be inhibited for (TBS) scan lines.

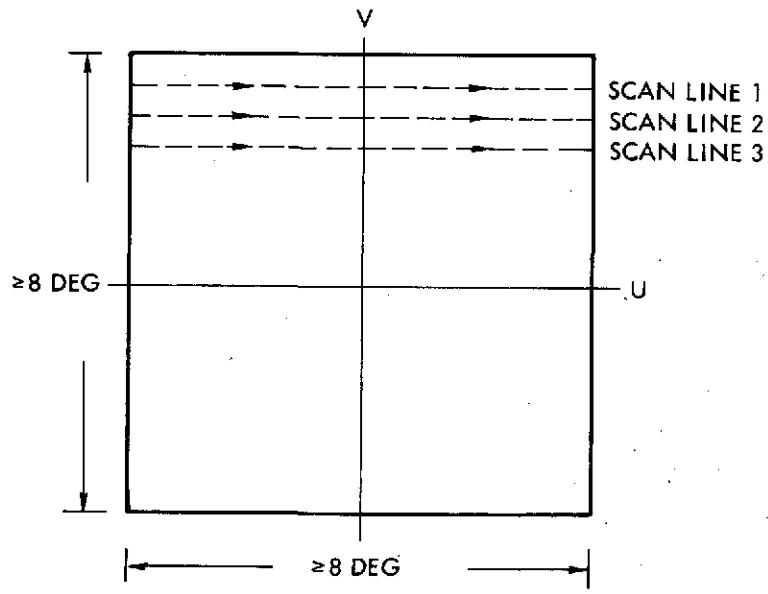


Figure 9. Projection of the STA Angular Field of View on the Focal Plane of the Optical System

(a) Star presence signal. The tracking function which is initiated in the track mode shall include the continuing determination whether or not there is a star within the tracking field of view which meets the commanded threshold setting. Existence of an adequate signal shall be indicated by the star presence bilevel output being true. The supplier shall furnish star magnitude calibration data.

(b) Track mode. A bilevel signal shall be true when the STA is in the track mode. The STA shall automatically revert to the search mode under the following conditions.

- The star moves outside the search field of view ± 5 percent
- The star presence signal fails to go true within 0. 1 sec of the track mode signal going true.
- The star presence signal having become true properly, goes false.
- The bright object signal becomes true indicating a bright object moves near the FOV.

(c) Star position signal. The STA shall generate analog voltages E_u and E_v proportional to the coordinates of U and V, respectively, of the star within the FOV. These signals shall be $-5 \text{ volt} \leq E_u \leq 5 \text{ volt} \leq -5 \text{ volt} \leq E_v \leq 5 \text{ volt}$ for star position within the search FOV. These voltages shall be provided as both wideband and narrowband outputs.

(d) Star position accuracy. The STA shall be calibrated with respect to cross talk, nonlinearity, stellar magnitude, temperature effects, magnetic field effects, dynamic lag effects, etc.

(e) Tracking statistics. The probability that the star presence signal will go false while the star is within the search FOV (±5 percent tolerance) shall be less than 10^{-2} per second for a (GOV) star at or above the commanded threshold against the scattered light irradiance values specified in TBD.

3. 2. 1. 2. 2. 3 Protection requirements. The STA shall be capable of sensing excess irradiance and protecting itself from damage, and of accepting a signal which will actuate the protective mechanism. The self-protection circuitry shall be energized whenever the primary power is applied without regard to on/off command.

3. 2. 1. 2. 2. 4 Optical/mechanical characteristics

(a) Electro-optical null axis. The electro-optical null (0) axis shall be aligned with respect to an optical alignment cube on the ADM within 10 arc sec, around the U and V axes.

(b) Angle measurement axes. The orientation of the U axis (rotation about the 0 axis) shall be defined with respect to a reference surface of the ADM with an error of not more than 15 arc sec RMS and parallel to the reference surface to within ±15 arc min.

(c) Coordinate systems. The coordinate system of the STA shall be denoted U, V, O. The O axis shall be the optical axis. The U and V axes shall each be perpendicular to the O axis. The +U axis shall be in the direction of scan of the high speed scan of the search raster. The -V direction shall be the direction of the slow speed scan of the search raster.

(d) Shade interface. The STA shall be installed with a shade incorporated in the ADM. The optics shall be designed so that the performance requirements specified herein can be met with an interfering collimated beam of light of irradiance $10^{-9}W/cm^2$ (5800° K black body) falling on the optics from any angular direction equal to or greater than 1 degree, outside the search FOV.

3.2.1.2.2.5 Thermistors. The STA shall provide temperature sensing thermistors. The location and characteristics of the thermistors are (TBS). The thermistors shall be calibrated prior to installation. The calibration data shall be submitted in the data package for each STA.

3.2.1.2.3 Star tracker shade. The star trackers shall each be fitted with a shade to provide attenuation of sunlight or earthshine for specified angle outside of the STA optical axis.

3.2.1.2.3.1 Design. The shade shall be conical in shape with outer diameter less than TBD inches and overall length less than TBD inches. The shade shall be thermally isolated from the lens of the STA.

3.2.1.2.3.2 Attenuation. The shade shall provide 10^8 attenuation of sunlight or earthshine for angles greater than 45 degrees of the STA optical axis.

3.2.1.2.4 Triaxial magnetometer assembly (TMA). The TMA shall provide electrical signals proportional to the three orthogonal components of the magnetic field at the magnetometer location. A single TMA shall be installed within the ADM.

3.2.1.2.4.1 Output signal characteristics. The three-axis outputs of the TMA shall be proportional and in the range of 0 to +5 VDC.

(a) Range. The operating range for this assembly shall be +2.0 to -2.0 gauss for any one axis.

(b) Accuracy. The accuracy for this assembly shall be ±1 percent of full scale (FS = ±2.0 gauss).

(c) Linearity. The nonlinearity of the reading shall not be more than ±1 percent of 1.25 volts/gauss over the range given in paragraph 3.6.1.1.1.

(d) Resolution. The resolution of the reading shall be less than 0.004 gauss.

(e) Output. The output signal from the assembly shall be nominally +2.500 VDC for zero gauss field. This bias voltage shall be provided as a separate output.

(f) Bias Tolerance. The bias output tolerance shall be ±0.025 VDC, with less than 5 millivolts peak-to-peak ripple.

(g) Scale factor. For maximum field of +2.0 gauss, the output shall be +4.95 ±0.05 VDC. For minimum field of -2.0 gauss, the output shall be +0.05 ±0.05 VDC. Ripple shall be less than 40 millivolts peak-to-peak.

(h) Frequency response. The frequency response of the assembly shall be greater than 50 Hz.

3.2.1.2.5 Sun sensor (SS). The ADM shall contain coarse and fine sun sensor installed to point along the -z axis of the spacecraft. The sun sensor outputs shall indicate the sun attitude relative to two orthogonal axes. The fine sensor shall provide the same information as the coarse sensor but with greater accuracy and over a smaller field of view.

3.2.1.2.5.1 Coarse sun sensor field of view. The coarse sun sensor field-of-view shall be 2π steradian.

3.2.1.2.5.2 Coarse sun sensor accuracy. The coarse sun sensor null accuracy shall be within ±2 degrees (1σ). The accuracy off null shall be within ±2 percent (1σ) of the calibration curve supplied with each unit.

3.2.1.2.5.3 Fine sun sensor field of view. The fine sun sensor operating field of view shall be at least ±15 degrees (square).

3.2.1.2.5.4 Fine sun sensor accuracy. The fine sun sensor null accuracy shall be within ±1 arc min (1σ). The accuracy off null shall be within 2 percent (1σ) of the calibration curve supplied with each unit.

3.2.1.2.6 Transfer assembly (TA). The TA shall contain the special-purpose electronics for processing and buffering the ADM sensor signals in a form suitable to the DIU. The TA shall also contain electronics for module thermal control and safe mode attitude control circuitry. Two TA's shall be installed with one being standby redundant. The TA construction shall be based on "slices" containing multilayer printed circuit boards.

3.2.1.2.6.1 Output signals. The TA output signals requirements are as follows: TBD.

3.2.1.2.6.2 Input signals. The TA input signal requirements are as follows: TBD.

3.2.1.2.7 Data interface unit (DIU). The DIU provides the interfacing electronics between the TA and the spacecraft command and data handling module. Two DIU's shall be installed in the ADM with one of these units being standby redundant. DIU is of standard design and is described in SP-1112.

3.2.1.2.8 Power conditioning unit. A power conditioning unit shall be provided to supply power to all ADM module units except the data interface unit. Power conditioning for the module shall be accomplished with the following components:

(a) Bus protection assembly

(b) Secondary power converter

3.2.1.2.8.1 Bus protection assembly. The bus protection assembly shall provide the following functions:

(a) Fusing for the +28 volt module primary power

(b) Fusing for the +28 volt heater power

(c) Safe mode bus logic.

3.2.1.2.8.2 Module primary power fusing. Redundant fusing shall be provided for each secondary power converter as follows:

Converter No. 1: 3.7 ±0.5 amp

Converter No. 2: 3.7 ±0.5 amp

3.2.1.2.8.3 Heater power fusing. Redundant fusing shall be provided for each heater line as follows:

Heater No. 1: TBD amp

Heater No. 2: TBD amp

3.2.1.2.8.4 Safe mode bus logic. Logic shall be provided to turn off all module components except heaters and equipment and for the safe mode, when the safe mode line voltage is below 0.4 volt.

3.2.1.2.9 Harness. The module harness shall provide all electrical interfaces between module assemblies and to the module/structure interface and test connectors. The harness shall be of modular design for maximum system flexibility. Installation or removal of the harness should be possible without removing electrical assemblies. It shall be possible to remove electrical assemblies without removing the harness. Cable strain relief or back-shell potting shall be employed at all harness terminations.

Wire sizes shall be selected to hold round trip voltage drops between source and load to one percent or less of the supply voltage. The minimum wire size for power and control circuitry shall be AWG No. 20. The minimum wire size for data or test circuitry shall be AWG No. 22. Under worst case conditions, wire temperature shall not exceed the temperature rating of the wire insulation.

3.2.1.3 Thermal. The module thermal control system design constraints are presented in the following paragraphs.

3.2.1.3.1 Module thermal requirements. The module thermal design shall satisfy the following on-orbit requirements:

- The module shall be capable of operation when the heat sink temperature is $\pm 20^{\circ}$F greater than the most severe predicted operating temperatures, where heat sink is defined as the structure or panel to which the electronic black boxes and other module equipment is mounted. These limits will be termed heat sink qualification temperatures. Less severe temperature limits can be used for components that might be damaged by the qualification temperatures if a waiver is obtained from the contractor.

- The module shall be designed so that the nominal set point temperature of the heat sink is 70°F with electrical heaters turned off.

- Electrical heaters shall be incorporated to maintain the orbit-average temperature at the module attachment locations above 60°F with the heater response approximating a sine pulse over one orbital period (rather than a step-input pulse).

- All module heat dissipation shall be radiated to space from the outboard facing panel.

- The surfaces of the module, except for the panel radiator areas, shall be thermally insulated with multilayer insulation, such that the effective emissivity, $\epsilon \leq 0.01$.

3.2.1.3.2 Module/structure assembly thermal interfaces. The design of the module thermal control system shall consider the following interface constraints:

- The structure assembly/module attach point temperature will be $70 \pm 10^{\circ}$F.

- Each module attachment fitting on the structure assembly will have a thermal resistance greater than 5 hr-°F/BTU.

- The effective emittance, ϵ, of the structure assembly/module insulation barrier will be ≤ 0.02.

3.2.1.3.3 Heater power constraints. Module thermal control system heater power shall not exceed 0 watts under normal operating conditions, and 9 watts under the most severe cold operating conditions that consider predictable variations in duty cycle and heating environment as well as parameter uncertainties in thermal properties, heating environment, insulation heat loss, etc.

3.2.1.4 Module structure. The module structure shall support all equipment listed in paragraph 3.1.3 and shall be capable of supporting additional equipment listed in paragraph 3.2.1.8 for modular expansion or complete redundancy.

No amplification of the vibration or acoustic environments shall be caused by the module structure which may result in degradation of the spacecraft performance.

The module structure, when mounted on the spacecraft structure, shall withstand the launch, ascent, and on-orbit loads as defined in SP-1111.

The structure shall not cause a change in alignment of the spacecraft axes by more than TBD arc sec.

The factors of safety shall be no less than 1.00 for limit loads and 1.25 for ultimate loads except where loads may be dangerous to personnel, the ultimate loads shall be 1.50.

3.2.1.5 Useful life. The design of the ADM shall be such that wearout of any item or depletion of expendables will not occur prior to a useful life of TBD years. Useful life is defined as the operating time of the equipment counted from the time of launch vehicle liftoff.

3.2.1.6 Storage life. The ADM shall have a minimum storage life of 3 years. Storage life critical components may be refurbished.

3.2.1.7 Telemetry. Data necessary for post-facto attitude determination and telemetry data shall be primarily limited for those functions necessary for control and operation of the ADM during flight. Telemetry indications of equipment status should be as direct as indication as practicable.

3.2.1.8 Expansion capability. A capability shall be provided within the limits of module structure, size, and power available for expanding the baseline configuration to include the addition of redundant units or the addition of new components to perform additional functions.

3.2.2 Physical characteristics

3.2.2.1 Mechanical

3.2.2.1.1 Envelope. The module envelope shall be as shown in ICD 60.1.

3.2.2.1.2 Module volume. The module shall have a volume of approximately 33 cubic feet and a maximum load carrying capability of 600 pounds of equipment. Components may be mounted to the outboard facing panel, nonoutboard surfaces, and the module frame members.

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The outboard facing panel may be modified with local cutouts to facilitate assembly or it may be divided into separate equipment heat sink surfaces. Internal stiffness bulkheads and/or mounting panels may be added as required. However, prior to any modifications to the module structure, the subsystem contractor shall perform a structural analysis to ensure structural integrity.

3.2.2.1.3 Module weight. The total weight of the ADM for the baseline and expanded configurations shall not exceed the listed weights.

Baseline Module	Expanded Module
TBD	TBD

3.2.2.1.4 Module center of gravity. The center of gravity of the module shall be located within TBD.

3.2.2.1.5 Attach-points. The attach-points between the module and the spacecraft shall be as shown in ICD 60.1.

3.2.2.1.6 Module/structure interface connector. The module/structure interface connector shall be provided by the system contractor and shall be mounted on the side face of the module. It is required that the connector position be maintained as specified to ensure interchangeability of modules.

3.2.2.1.7 Equipment expansion. The components shall be arranged in the module such that the expansion capability requirements of paragraph 3.2.1.8 can be accommodated with minimum impact on the baseline configuration design.

3.2.2.2 Electrical

3.2.2.2.1 Power. Total power required for the ADM shall not exceed TBD watts. Allocation of this power is as follows:

Baseline configuration requirement: TBD watts

Redundancy and expansion capability: TBD watts

Power consumption of the ADM units shall be within the power allocations in ICD 10.2.

3.2.2.2.2 Commands. Commands for controlling ADM operation shall be as listed in ICD 10.3.

3.2.2.2.3 Telemetry. The ADM telemetry measurements shall be as listed in ICD 10.4.

3. 2. 2. 2. 4 Signal and power distribution. The ADM harness shall provide all intramodule electrical connections in conformance to ICD 60. 5.

3. 2. 3 Reliability. The ADM shall be capable of performing, as specified, for at least TBD years in orbit. This shall include all redundancy incorporated including alternate and backup modes. Demonstration of compliance with these requirements shall be through reliability analysis as called out in EOS Document EOS-4. 1, System Effectiveness Program Plan.

3. 2. 4 Maintainability. The ADM shall be designed in accordance with the requirements of MIL-STD-1472, paragraph 5. 9, as implemented by EOS-4. 1.

3. 2. 5 Environmental conditions. The ADM shall be designed to withstand or shall be protected against the worst probably combination of environments as specified in SP-11 and as implemented in SP-115.

3. 2. 6 Transportability. Transportability requirements shall be considered in the design of the electrical power module such that it can be transported by all standard modes with a minimum of special packing or precautionary measures.

3. 3 Design and construction

3. 3. 1 Parts, materials, and processes

3. 3. 1. 1 Selection of parts, materials, and processes. Selection of parts, materials, and processes (PMP) shall be to the requirements identified in MIL-STD-143 Group 1. Such selections shall be identified as standard PMP. When PMP selection cannot be made within this requirement, the item is nonstandard and justification must be made in accordance with MIL-STD-749. Control of this action shall be effected by the EOS parts, materials, and processes control board (PMPCB), with the PMPCB approving all selections, both standard and nonstandard, in accordance with EOS-4. 1.

3. 3. 1. 2 Program authorized parts list. All selections of electronic parts shall be identified and authorized by EOS-3. 3-1. This list will reflect all PMPCP electronic part selections.

3. 3. 1. 3 Program authorized materials list. All selections of materials shall be identified and authorized by EOS-3. 3-2. This list will reflect all PMPCB material selections.

3. 3. 1. 4 Program authorized processes list. All selection of processes shall be identified and authorized by EOS-3. 3-3. This list will reflect all PMPCB process decisions.

3. 3. 1. 5 Dissimilar metals. To avoid electrolytic corrosion, dissimilar metals shall not be used indirect contact unless protection against corrosion has been provided in accordance with MIL-E-8983 and MIL-STD-454, Requirement 16.

3.3.1.6 Magnetic materials. Magnetic materials shall be used only if necessary for equipment operation. Those magnetic materials which are used shall have minimum permanent, induced, and transient magnetic fields.

3.3.1.7 Fungus-inert materials. Materials which are fungus-inert, in accordance with MIL-E-8983 and MIL-STD-454, Requirement 4, shall be used.

3.3.1.8 Flammable toxic and unstable materials. Flammable, toxic and unstable materials shall not be used.

3.3.1.9 Finish. The surface of each component shall be adequately finished to prevent deterioration from exposure to the specified environments that might jeopardize fulfillment of the specified performance. The finish shall also meet the applicable bonding requirements. Thermal properties of the finishes used on the component shall be compatible with the requirements given in detail in applicable equipment specifications. The requirements for finishes identified in MIL-E-8983 shall be implemented.

3.3.1.10 Outgassing. Low outgassing polymeric materials shall be used where sensitive thermal control and other surfaces are in direct line of sight and where temperature differences can exist between such surfaces.

3.3.2 Electromagnetic radiation

3.3.2.1 EMC/EMI requirements. The electrical power module, and all internal units, equipment and/or components comprising a part thereof, shall be designed for compliance with the radiated emission and susceptibility requirements of NASA/JSC Specification SL-E-0002 as modified or amended by EOS-3.3-4 and EOS-3.3-5. The conducted emission and susceptibility requirements of the aforementioned documents are applicable only at the module-to-spacecraft structure interface. EMI/EMS levels at interfaces within the module shall be controlled to the extent necessary to ensure self-compatibility of the module subsystem with a safety margin of at least 6 dB.

3.3.2.2 Electrical bonding

3.3.2.2.1 Structural bonds. All metallic members of the basic electrical power module radiator panel and support structure shall be electrically continuous, equi-potential ground reference plane. The maximum allowable DC impedance across any one structural joint shall be 2.5 milliohms with an overall design goal of 10 milliohms, or less, between any two diametrically opposite points on the module. The general methods of MIL-B-5087, Class R, may be used for implementation of these requirements.

3.3.2.2.2 Equipment mounting pads. Surfaces on the module radiator panel and structure which are intended for unit, equipment, or component mounting shall be free of paint, anodize, or other nonconductive finishes. The maximum DC impedance between the baseplate of any electrically active unit, equipment or component and the module radiator panel and structure shall be 2.5 milliohms.

3.3.2.2.3 Electrical connectors. All interface electrical connectors both plug and receptacle, which form a part of the unit and cable RF shielding system within the module shall have electrically conductive body shells, free of nonconductive finishes. They shall also have provisions for terminating (grounding) the shields on the interconnecting electrical harness. The maximum DC impedance between the shield termination point and the baseplate of the parent unit shall be 2.5 milliohms.

3.3.2.2.4 Electrostatic bonds. Electrically passive components or appurtenant structures which are attached to the basic module structure through thermal isolators shall be provided with electrostatic bonds having a DC impedance of 1.0 ohm or less. The general methods of MIL-B-5087, Class S, may be used for implementation of this requirement.

3.3.2.3 Electrical systems grounding

3.3.2.3.1 Primary DC power. The electrical power module shall provide the single-point structural ground for the primary DC power distribution subsystem. This subsystem will be DC isolated from structural grounds in all user modules as well as in the solar array. The physical location of the ground point may be within, or immediately adjacent to the battery assembly. Interconnecting primary power wiring between units and/or assemblies of the electrical power module shall be unshielded, twisted pairs.

3.3.2.3.2 Secondary DC power. Secondary DC power distribution networks shall, in general, use multiple point grounding within the electrical power module with returns through the module radiator panel and/or support structure. These networks shall be initially grounded adjacent to the secondary transformer winding in the power converter and at each user element.

3.3.2.3.3 Secondary AC power. Secondary AC power networks shall use single point grounding with a two-wire, twisted shielded pair distribution. The ground point may be either at the source or load end, whichever is shown by circuit analysis to be most beneficial to compatible module operation. Structural returns shall not be used for secondary AC power.

3.3.2.3.4 Intramodule signal/control circuit (high level). All high level (<5 volt logic, bilevel or analog) signal or control circuits which do not exit the electrical power module shall be multiple point grounded at both the source and load end of each circuit branch to the unit or component chassis by the shortest most direct path. Circuits sharing space on a common printed circuit board should not share common grounding traces on the board or common hardwire jumpers to chassis ground logs. Preferably, each such board should have a dedicated ground plane layer to which all components requiring ground returns can be directly connected. This ground plane should, in turn, be directly bonded to unit chassis or frame through grounding pads at each hold-down fastener.

3.3.2.3.5 Intermodule signal/control circuits (high level). All high level signal or control circuitry which exists the electrical power module shall be grounded at the final driving element. Two-wire, twisted shielded pair distribution shall be used for each such circuit between the source unit and the module-to-spacecraft interface connector. The load elements in the external module will be DC isolated from structural grounds. Any signal or control circuitry which enters the electrical power module shall be DC isolated from chassis/case/structure ground by a minimum of 1 megohm resistance. Any such circuits shall also be provided with two-wire, twisted shield pairs between the interface connector and the load unit.

3.3.2.3.6 Analog circuits (low level). Any low level (< 5 volts) analog circuits, which are shown by circuit analysis or test to be sensitive to circulating currents in the module or spacecraft structure, shall be single-point grounded either at the source or load element, whichever is most appropriate for the circuit under consideration. Wherever possible, balanced differential circuitry should be used. In the case of low level circuits which enter or exit the electrical power module, the location of the circuit ground point shall be coordinated with the systems integration contractor.

3.3.2.3.7 Data bus. The command and telemetry data bus system shall be differentially driven and balanced to structural ground in the communication and data handling module. This system shall be transformer-coupled at each remote terminal. Each individual data bus wire entering the electrical power module shall be DC isolated from chassis/case/structure by a minimum of 1 megohm resistance.

3.3.2.3.8 Wire shields. External shields shall be provided for all interconnecting wires between units, equipment or components in the electrical power module and between each input/output connector and the module-to-spacecraft interface connectors except for input and output primary DC power lines. In general, these shields shall be multi-point grounded at each end and at each intermediate interface. An exception to this rule will be allowed for low level analog circuitry where single-point shield grounding may be necessary. If possible, such circuitry should be provided with two mutually-isolated shields, the inner shield being single-point grounded and the outer shield multi-point grounded.

3.3.2.3.9 EMI filter components. High performance EMI filters will be required at the primary DC power input and return terminals of each power converter unit or sub-unit in the electrical power module to ensure compliance with the electromagnetic interference requirements of the applicable specifications. These filters should have two stages: (1) an AF ripple filter stage balanced line to line; and (2) a pair of RF feedthrough filters bulkhead-mounted behind the input connector. The combined AF/RF filter circuit should be designed for the minimum capacitance from either line to chassis necessary to achieve compliance with the specifications. Excessive line-to-ground capacitance will tend to negate the beneficial effects of the twisted pair wiring used in the primary DC power harness. Similar filtering may be required at the input and output terminals of the pulsewidth switching regulator assembly; however, because of fault isolation requirements, line-to-case feedthrough RF filters should not be used.

3.3.3 Nameplates and product marking. Each unit shall be identified in accordance with MIL-STD-130 as implemented by EOS-3.3-6. Where practical, a minimum character height of 0.09 inch shall be used. Marking shall include the manufacturer's name or initials, assembly part number and revision status, assembly serial number, contract number, and others as delineated in the component equipment specifications.

3.3.4 Workmanship. The ADM and its component equipment shall be constructed, finished, and assembled in accordance with MIL-STD-454, Requirement 9, and the specifications and drawings specified herein.

3.3.5 Interchangeability and replaceability. The electrical power module shall be designed to permit removal and replacement of components with a minimum of disturbance of associated or adjacent equipment. Design for service and access shall conform to the principles and requirements of MIL-STD-470.

3.3.6 Safety. The electrical power module shall be designed to meet or exceed the requirements of EOS-3.3-7, as implemented by EOS-4.1, System Effectiveness Program Plan. The design criteria include but are not limited to those set forth in MIL-STD-882.

3.3.7 Human performance/human engineering. MIL-STD-1472A, "Human Design Criteria for Military Systems," shall be used for the design of man/machine interfaces.

3.4 Documentation. Documentation shall be as defined in EOS-3.3-8.

4. QUALITY ASSURANCE PROVISIONS

4.1 General. The quality assurance program control shall be in accordance with the requirements of MIL-Q-9858 as augmented by EOS-4.1.

4.1.1 Responsibility for inspection and test. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified herein. Except as otherwise specified, the supplier may use his own or any commercial facilities acceptable to the government. The government reserves the right to perform any of the inspections set forth in the specifications where such inspections are deemed necessary to ensure that supplies and services conform to prescribed requirements.

4.2 Quality conformance inspections

4.2.1 Category I tests

4.2.1.1 Development tests. Development tests shall be conducted to verify the module performance parameters, proper interfacing between components of the module, and proper interfacing between the module and other spacecraft modules. The development tests shall be conducted at the unit and/or integrated module levels. The type of developmental evaluation or test to be applied to verify satisfaction of the requirements defined in this specification are outlined in Table 1. The tests shall be conducted in accordance with EOS-4.2 and EOS-4.6.

As each payload instrument is integrated, interface signal characteristics shall be measured to verify design margins and compatible operation with the spacecraft data bus and power bus. These tests shall be conducted in accordance with EOS-4.2.

4.2.1.2 Qualification tests. Module qualification shall be accomplished by means of qualification tests on the individual units and/or as a normal consequence of having the units installed in the qualification test space vehicle during its qualification testing. The types of qualification evaluation or test to be applied to verify satisfaction of the requirements of this specification are outlined in Table 1. Qualification test verification methods and requirements shall be as defined in EOS-4.2.

4.2.1.2.1 Components. As a minimum, the following types of component qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, are measured before and after the environmental tests.

- Random vibration. Each component will be subjected to a random vibration from 20 to 2000 Hz in all three axes while electrically powered in the maximum normal load condition. Performance will be continuously monitored to detect failures (either hard or trends).

- Thermal/vacuum. Thermally cycle the component beyond expected orbital temperatures while at an orbital vacuum condition. The component is operating continuously and detailed performance parameters are measured at each temperature extreme.

- Electromagnetic interference (EMI) and susceptibility (EMS). Engineering EMI and EMS data are measured to provide diagnostic information for the module EMI/EMS qualification.

4.2.1.2.2 Module. As a minimum, the following types of module qualification tests shall be included:

Table 1. Verification Cross Reference Index

VERIFICATION CROSS REFERENCE INDEX									
LEGEND: N/A - Not Applicable; I - Inspection; A - Analysis; S - Similarity; T - Test									
SECTION 3 REQUIREMENTS	Not Applicable	Acceptance			Qualification				SECTION 4 VERIFICATION REQUIREMENTS
		I	A	T	I	A	S	T	
(TBD)									

SAMPLE

- Functional. Performance parameters, electrical and/or mechanical, and electrical interface characteristics are measured before and after the environmental tests.

- Acoustics. Acoustics testing will be performed with the module mounted in a receptacle which provides acoustics shielding similar to the Observatory structure. The module will be heavily instrumented to verify design adequacy of interconnections (electrical and mechanical) between the module and other Observatory elements. The test data will be used to confirm estimates of the vibration environment for components mounted on radiators and establish procedures for acceptance testing. The module will be electrically powered and continuously monitored via the data bus to detect failures and out-of-tolerance performance trends.

- Thermal vacuum. The objectives of the module thermal qualification test are to determine: 1) maximum and minimum operating temperatures for the module for worst-case environments and operating duty cycles, 2) temperature levels and temperature variations of the module adjacent to the interface fittings, and 3) heater power requirements for cold-case conditions.

  The test will be conducted in a thermal-vacuum chamber with liquid nitrogen cold walls. The module will be mounted on a fixture simulating the spacecraft frame. The properties of this fixture shall permit its temperature to be held constant at any level between 0 and 100°F.

  A heating source for the radiators will approximate the absorbed flux of the external environment. This can be done with electrical heaters, infrared lamps, or other techniques where the absorbed heating can be determinated accurately.

  - Power bus and data bus will be tested in excess of their operational limits to determine design margins and compliance with the interface specification.

  - Detailed performance data will be measured to determine module specification values.

  - Thermistor/heater control and calibration will be determined.

- EMI and EMS. EMI measurements will determine the levels radiated on each module electrical interface and verify there is adequate design margin to ensure noninterference with other modules. EMS measurements will verify that each module has adequate susceptibility margin to prevent performance degradation resulting from other module allowable radiation levels. The general test methods of MIL-STD-462, as modified or amended by EOS -3.3-5, shall be used during this test program except that, wherever practical, spectrum analyzers or other forms of rapid display and analysis equipment may be used in lieu of equipment requiring manual tuning and data collection.

4.2.1.2.3 Observatory. As a minimum, the following types of Observatory qualification tests shall be included:

- Functional. System performance parameters will be measured before and after the environmental tests. The functional tests consist of: 1) integrated system test (IST), 2) detailed subsystem tests, 3) detailed instrument tests, 4) power load determination (before only), 5) deployment tests, 6) alignments determination, 7) leak test, and 8) solar array illumination. In addition, weight and center-of-gravity design is measured.

- Low-frequency sine vibration. The Observatory will be subjected to sine vibration at levels less than 1.1 times limit load over the range of 5 to 100 Hz in all three axes. The Observatory will be heavily instrumented to verify the analytical model used for loads prediction and design adequacy of primary and secondary structure and module connections. The Observatory will be electrically powered in the launch mode. RF telemetry data will be monitored continuously to verify the electrical system performance and design adequacy of the electrical interconnections between modules.

- Acoustics. The Observatory will be subjected to an acoustics test while mounted vertically on an integration and test pedestal. Extensive instrumentation will be used to: 1) verify design adequacy of the solar array and other nonmodule components, 2) confirm estimates of the vibration environment for components mounted on walls other than radiators, and 3) confirm adequacy of the module receptacle used for module-level acoustics test. The Observatory will be electrically powered in the maximum normal load condition and continuously monitored via the RF telemetry link to detect performance degradation.

- Shock. All ordnance (separation system, pin pullers on the array, antennas, and module supports) will be fired to verify design adequacy of all Observatory components. The Observatory will be electrically powered in the appropriate mode during the firing. RF telemetry will be monitored to detect performance degradation.

- Thermal vacuum. The Observatory thermal control test has the basic objectives to evaluate the module testing concept and the thermal control system. Evaluation of modular testing requires correlation of system and module test results. Evaluation of the thermal control system entails the following: 1) structure thermal control, 2) module/structure interaction, and 3) required heater power.

The thermal vacuum test will be conducted in a thermal vacuum environmental chamber with an LN_2 cold wall. The -Z side will be irradiated with a heat source that can be accurately defined (solar simulation will not be necessary). The +Z side will face the cold wall; no attempt will be made to simulate the external energy input (earth emission, albedo, and solar). This test method will allow an accurate thermal definition of the chamber environment. The test conditions will include a cooldown phase to evaluate heat leaks, a steady-state phase to evaluate heater requirements, and a transient phase to evaluate interface interactions.

Throughout the test, detailed temperature data will be measured to verify the thermal analytical model. Design adequacy of the thermal insulation, heater control, and thermistor placement will be determined. In addition, thermistor calibration and methods for thermal evaluation and control by the ground station will be analyzed. At each thermally-stabilized level, integrated systems test and sensor (ADM and instrument) aliveness tests will be performed. Telemetry data will be monitored continuously to verify design margins of all subsystems.

- Electromagnetic compatibility. Noise levels will be measured on critical signals (including interfaces) utilizing the module test connectors. This will verify module design adequacy and and validate the EMI/EMS interface criteria established for module-level qualification.

4.2.1.2.4 Qualification by similarity. With concurrence of the procuring agency, qualification by similarity shall be acceptable at the component level where the component design or mounting has not significantly changed and the component has been previously qualified under applicable environmental conditions.

4.2.1.2.5 Inspection sequence. The inspection sequence shall be governed by the following:

- Examination of the module shall be performed prior to functional testing.

- Functional tests shall be performed prior to, during, where appropriate, the following environmental testing. The functional testing conducted at the conclusion of an environmental test may serve as the functional testing to be performed prior to the next environmental test.

- Environmental testing may be performed in any sequence.

4.2.1.2.6 Failure criteria. The module shall exhibit no failure, malfunction or out-of-tolerance performance degradation as a result of the examinations and test specified herein. If a failure occurs during the performance of any test, the test shall be suspended and the discrepancy, failure reporting, analysis, and corrective action procedures as set forth in EOS-4.1 shall be followed.

4.2.2 Test conditions. Unless otherwise specified, the conditions under which all inspections are accomplished shall be specified below:

4.2.2.1 Atmospheric and environmental. All examinations and tests shall be conducted under the local prevailing pressure at the time of test, at a temperature of 75 ±5°F and at a relative humidity of 55 percent or less.

4.2.2.2 Deviations of atmospheric conditions. When tests are performed with atmospheric conditions substantially different from the specified values, proper allowances for changes in instrument readings shall be made to compensate for the deviation from the specified conditions.

4.2.3 Equipment warmup time. The equipment warmup time shall be less than 1 minute.

4.2.4 Temperature stabilization. Temperature stabilization shall have been achieved when temperature of the equipment mounting structure varies not more than 5°F during a period of 15 minutes.

4.2.5 Measurements. Measuring instruments used to determine functional parameter values (such as voltage, frequency, current, flow, pressure, etc.) shall indicate true values with an accuracy determined by the tolerance allowed for the parameter variation itself, such that the measuring instrument shall not introduce an uncertainty greater than 10 percent of the allowable variation of the measured parameter. However, no such measurement accuracy shall be required to exceed 0.5 percent of the required value of the parameter unless otherwise specified.

Test instrumentation and equipment shall comply with the requirements of MIL-C-45662.

Except as specifically noted in the inspection methods, tolerances for environmental test conditions shall be defined as follows:

Temperature: ±5°F

Barometric pressure: ±10 percent

Relative humidity: +5, -10 percent

Time: ±5 percent

Vibration amplitude (sine): ±10 percent

Vibration frequency: ±2 percent

4.3 Category II tests. (Not applicable)

4.4 Analyses. The following requirements of Section 3 shall be verified by review of analytical data:

3.2.3 Reliability. To be verified by analysis in accordance with Section TBD of EOS-4.1.

3.3.6 Safety. To be verified by analysis in accordance with Section TBD of EOS-4.1.

5. PREPARATION FOR DELIVERY

5.1 General. The ADM as specified herein shall be protected from degradation by environments anticipated during shipment, handling, and storage. Standard commercial packaging practices are acceptable provided they fulfill these requirements.

5.2 Preservation and packaging

5.2.1 Cleaning. The ADM shall be clean and free of contaminants which might impair its use prior to packaging.

5.2.2 Attaching parts. When attaching parts, such as nuts, bolts, washers, etc., accompanying the ADM, they shall be preserved, bagged, appropriately identified, and attached to or adjacent to the fitting for which they are intended.

5.2.3 Electrical connectors. All electrical connectors shall be capped with protected dust caps. Caps used shall be a friction fitting or threaded type which do not require tape or a mechanical device to secure.

5.2.4 Critical surfaces. External machined surfaces and mounting surfaces of the ADM shall be protected with protective pads. Materials used for pads shall not cause item deterioration.

5.2.5 Wrapping. The ADM shall be wrapped or bagged using anti-static polyethylene film.

5.2.6 Cushioning. When required for protection, the ADM shall be cushioned using a suitable resilient foam cushioning material such as polyethylene, polyurethane, or polystyrene.

5.3 Packing. The ADM shall be packed in a suitable shipping container designed for one item only. The shipping container used shall provide protection to the item against corrosion, deterioration, and damage during shipment from the source of supply to the receiving activity. Containers shall comply with applicable tariffs and regulations for particular modes of transportation when so shipped.

5.3.1 Storage conditions. The ADM shall not be adversely affected by storage within its container at temperatures between 60°F and 90°F and relative humidities of 60 percent or less.

5.3.2 Shipping conditions. The ADM shall be capable of withstanding the following environments:

Temperature:	+160°F in an unsheltered area (125 +35°F due to solar radiation) and -40°F in accordance with restricted air transport criteria
Humidity:	Up to 100 percent in an unsheltered area
Rough handling:	Capable of withstanding and physically protecting the unit from rough handling during shipping by common carrier

5.4 Marking for shipment. Each ADM and shipping container shall be marked with the following:

(a) Item nomenclature

(b) Systems part number

(c) Contractor or purchase order number

(d) Manufacturer's name

(e) Manufacturer's part number and serial number (on item container only)

(f) Quantity

(g) Date of manufacture (on item container only)

(h) Fragile — Handle With Care (when applicable)

(i) Space Vehicle Material — Do Not Open In Receiving Or Receiving Inspection (when applicable — shipping container only)

(j) Actual weight

5.4.1 Documentation. All required reliability and test documentation such as test reports, certifications, shipping invoices, etc., shall be either packed in the ADM container or attached to the exterior surface of the shipping container. Attachment shall be such as to preclude loss of these data during handling and shipment by common carrier.

TITLE

PRIME ITEM DEVELOPMENT SPECIFICATION
FOR THE

ACTUATION MODULE

DATE 20 SEPT 1974 NO. SP-1115

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

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SPECIFICATION SP-1115

ACTUATION MODULE

1. SCOPE

1.1 Scope. This specification establishes the functional requirements for performance, design, development, test, and quality assurance for the actuation module (AM). The AM is used to perform functions delinated herein for the Earth Observation Spacecraft, and unmanned earth orbiting spacecraft.

2. APPLICABLE DOCUMENTS

2.1 Document. The following documents of the exact issue specified or if not specified, the latest issue in effect form a part of the specification to the extent specified herein. The requirements of this specification shall supersede those of the documents referenced in those cases where differences occur between this specification and the referenced documents.

SPECIFICATIONS

NASA

SP-11	Observatory Segment Specification
SP-115	Observatory Environmental Criteria Specification
SP-1111	Structure Assembly Specification
SP-1112	Communication and Data Handling Module Specification
SP-1113	Electrical Power Module

Military

MIL-E-8983A	Electronic Equipment, Aerospace Extended Space Environment, General Requirements
MIL-B-5087	Bonding, Electrical, and Lightning Protection for Aerospace Systems
MIL-Q-9858	Quality Program Requirements

STANDARDS

Military

MIL-STD-143B	Standard and Specifications, Order of Precedence for Selection
MIL-STD-454C	Standard General Requirement for Electronic Equipment
MIL-P-26536C	Propellant Hydrazine
MIL-P-27401B	Propellant, Nitrogen Pressurizing Agent

MIL-STD-1522	Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems
MIL-STD-749B	Preparation and Submissions of Data for Approval of Nonstandard Electronic Parts
MIL-STD-1472A	Human Engineering Design Criteria for Military Systems, Equipment, and Facilities
MIL-STD-882 15 July 1969	System Safety Program for Systems and Associated Subsystems and Equipment, Requirements for
MIL-STD-462	EMI/EMS Test Methods
MIL-STD-130D	Identification Marking of U.S. Military Property
MIL-STD-470	Maintainability Program Requirements for Systems

OTHER PUBLICATIONS

NASA

SL-E-0002	Electromagnetic Compatibility Control Plan
NASA STDN101.1 X-560-63-2	STDN User's Guide Aerospace Data Systems Standards
- - - -	Electromagnetic Compatibility Requirements for Space Systems
EOS-3.3-1	Program Authorized Parts List
EOS-3.3-2	Program Authorized Materials List
EOS-3.3-3	Program Authorized Processes List
EOS-3.3-4	Electromagnetic Compatibility Control Plan
EOS-3.3-5	EMI/EMS Limits and Test Methods
EOS-3.3-6	Marking of Parts and Assemblies

2.2 Non-Government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. If no revision or date is shown, the latest released issue of the applicable document shall apply. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

DRAWINGS

50.5	Actuation Module Wire List
10.4	Satellite Telemetry Allocation, ICD
10.3	Satellite Command Allocation, ICD
10.2	Satellite Primary Power Allocation, ICD
50.1	Actuation Module Envelope, ICD
TBD	Actuation Module Assembly Drawing

OTHER PUBLICATIONS

EOS-3.3-7	Safety Design Criteria
EOS-3.3-8	Configuration Management Plan
EOS-4.1	System Effectiveness Program Plan
EOS-4.2	Integrated Test Plan

3. REQUIREMENTS

3.1 General

3.1.1 Functional requirements. The AM shall contain the mission-peculiar attitude and control equipment required to rotate the spacecraft in three axes to achieve the desired attitude in response to commands from the command and data handling module (CDHM) or the attitude determination module (ADM). In addition, the ADM shall contain propulsion equipment to provide ΔV capability to achieve orbit transfer/adjust maneuvers.

Two separate propulsion systems are described in the specification. System I shall be used for applications which require ΔV for periodic orbit adjust maneuvers and attitude control of the vehicle during these maneuvers and as a backup to the reaction wheel main attitude control system. System II shall be used for applications which require additional ΔV for orbit transfer maneuvers to and from the operational altitude as well as for orbit adjust and attitude control.

3. 1. 2 Operations. In normal operation the AM controls the spacecraft attitude and provides ΔV for orbit transfer/adjust maneuvers based on commands from the CDHM which processes sensor data provided by the ADM. In the backup failure mode control is by means of the sun sensor and low-level thrusters. Special-purpose hardware imbedded in the ADM replaces the data processing function of the CDHM and the reaction wheel and torquers are disabled in the AM. Details of the various normal and failure modes of operation are described in the ADM specification. A typical layout of the AM is shown in Figure 1 and block diagrams are presented in Figures 2 through 4.

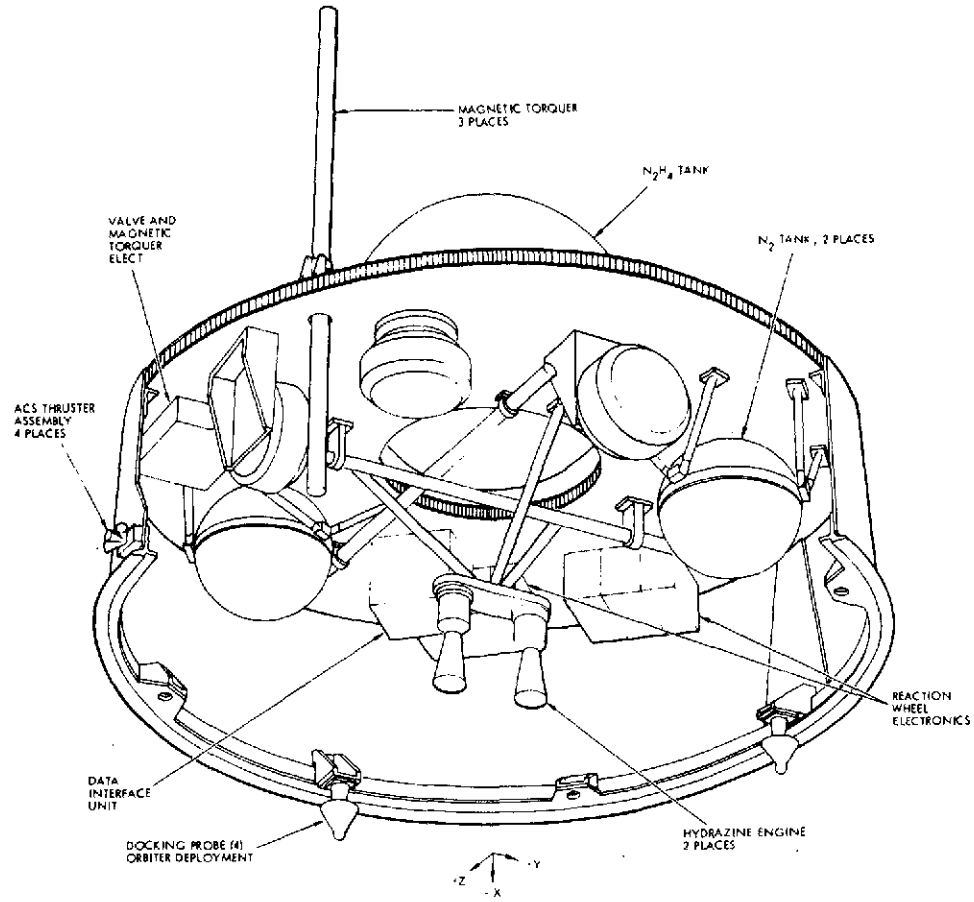

Figure 1. Typical Actuation Module Layout

3. 1. 3 Major components list. The AM shall consist of the complement of equipment listed in Table 1. The equipment consists of the attitude control equipment group and the reaction control and propulsion equipment group. The latter group consists of components for System I and System II where functions have been described in Section 3. 1. 1.

3. 1. 4 Interfaces. The electrical interface of the AM is illustrated in Figure 5. Basic inputs to the module come from the power module, command and data handling module, and the attitude determination module. Outputs of the AM are for telemetry and module test points.

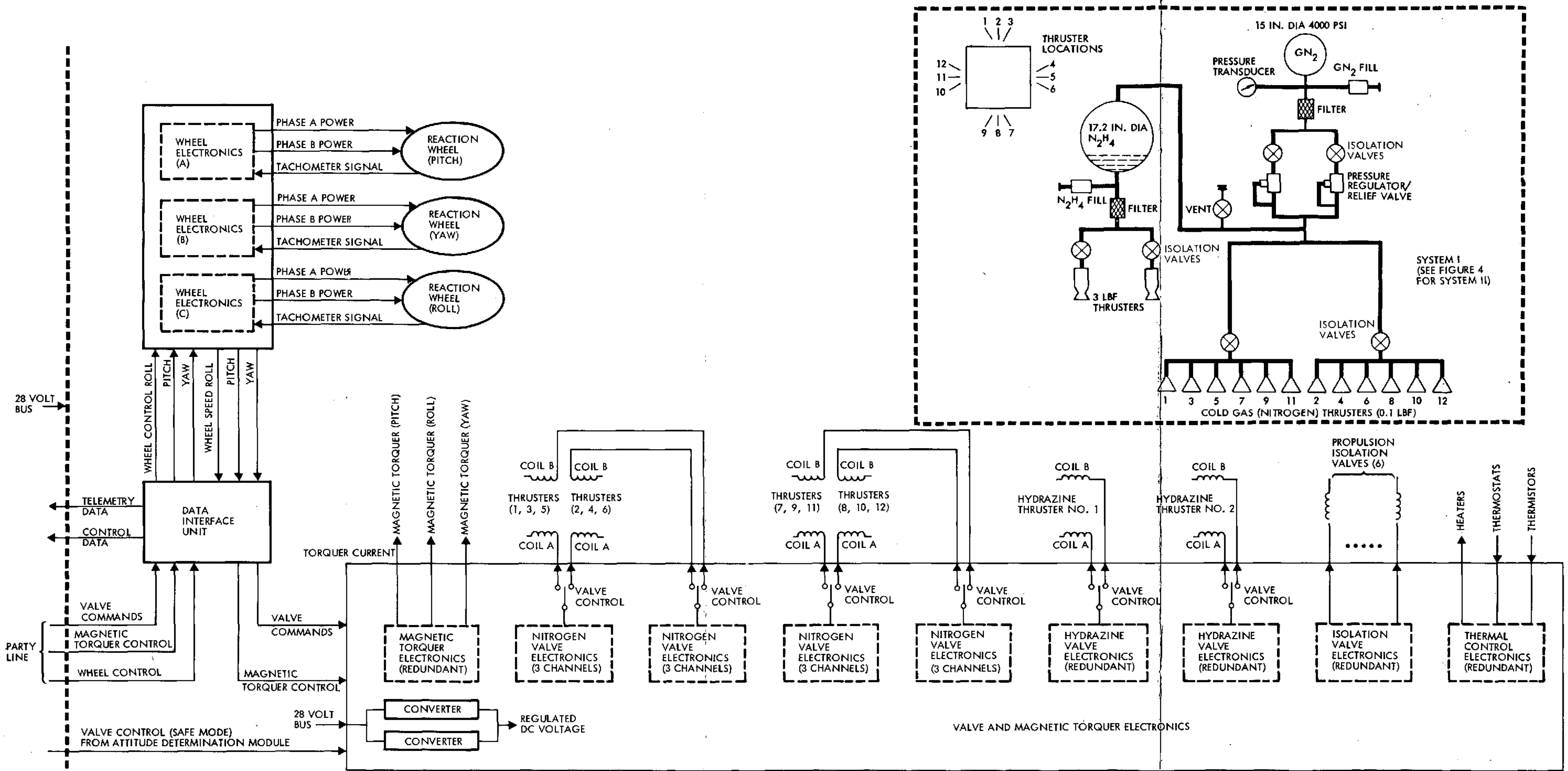

Figure 2. Actuation Module (Minimum Redundancy)

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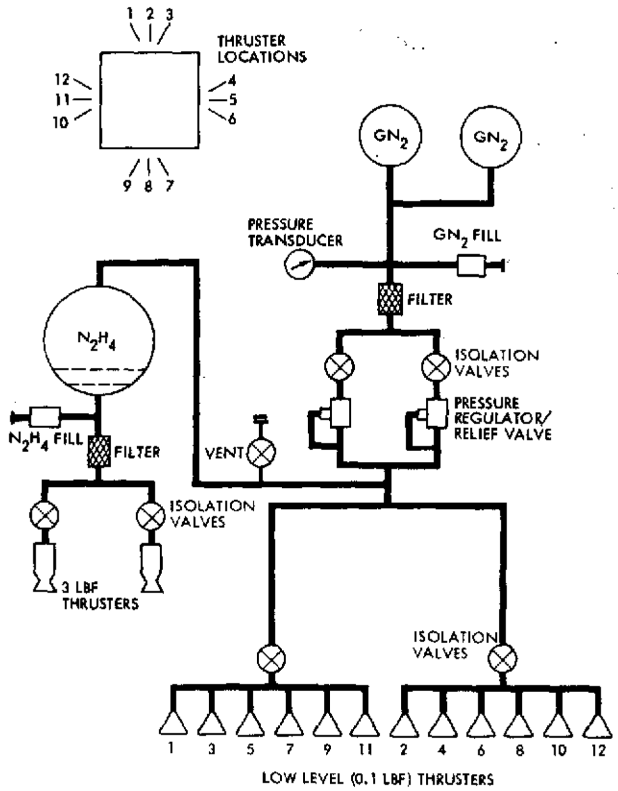

Figure 3. Orbit Adjust – Single Level RCS Configuration (System I) (Minimum redundancy removes 1 to 3 pound thruster)

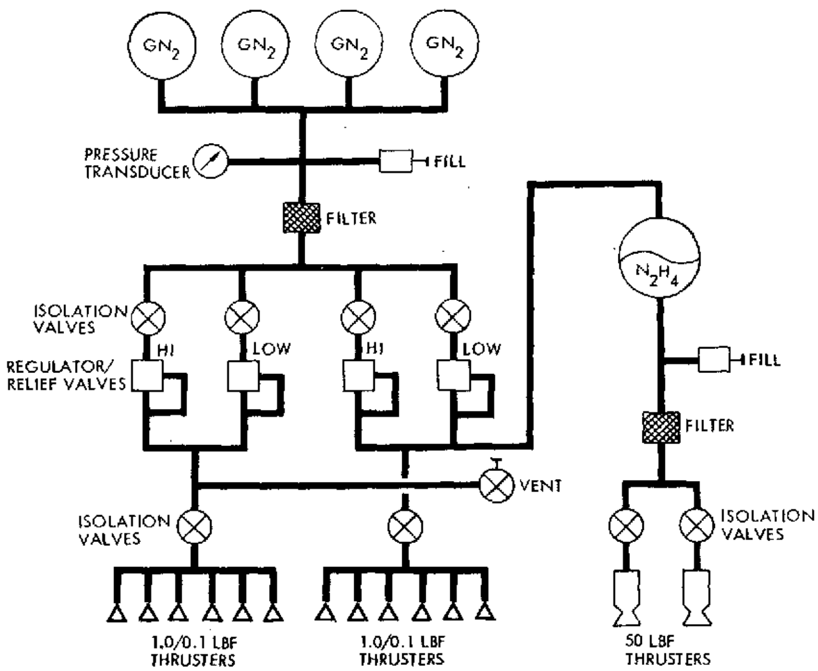

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Figure 4. System II Schematic

Table 1. Major Components List

Equipment	Number	
Attitude Control Group		
Reaction wheels*	3	
Magnetic torquers*	3	
Reaction wheel electronics	3	
Value and magnetic torquer electronics	1	
Data interface unit	1	
Reaction Control and Propulsion*	System 1	System 2
N_2H_4 tank	1	1
GN_2 tank	2	4
Fill	2	2
Pressure transducer	1	1
Filter	2	2
Isolation valves	6	8
Regulators	2	4
N_2H_4 thruster (3.0 lbf)	2	
N_2H_4 thruster (50 lbf)	-	2
GN_2 thrusters (1.0 and 0.1 lbf)	12	12
Vent	1	1
GN_2	18.7 lb	68 lb
N_2H_4	96.3 lb	866 lb
Harness		
Thermal control		
Module structure		

*Mission-Peculiar Units

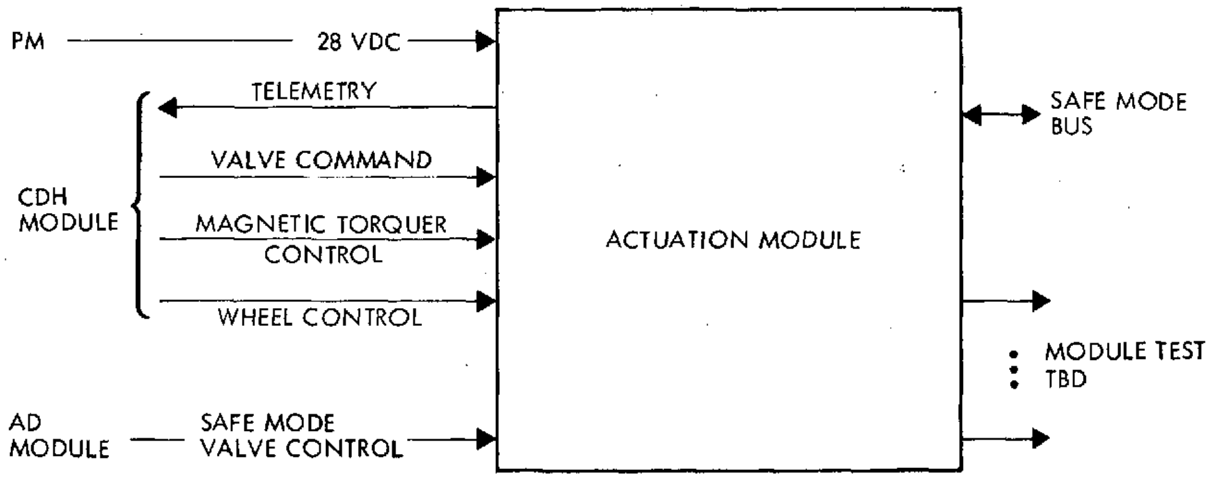

Figure 5. Actuation Module — Electrical Interface

3.1.5 Government-furnished module equipment. Certain items of module equipment which are identical or nearly identical in function will be fabricated by the integration contractor and provided as GFE to the remaining module contractors. This equipment shall include:

Item	Reference
Data interface unit	3.2.1.3.4 of SP-1112
Power control unit	3.2.1.9 of SP-1113

All module structures and integration of the modules will be provided by the integrating contractor.

3.2 Characteristics

3.2.1 Performance

3.2.1.1 Attitude control equipment group

3.2.1.1.1 Reaction wheel assembly. The reaction wheel assembly (RWA) shall consist of a motor driven inertial wheel contained within a sealed case. Three RWA shall be mounted, one along each of the three control axes. The RWA shall be used to absorb spacecraft momentum.

Primary input to the RWA will be a signal proportional to spacecraft attitude and attitude rate based on data from the ADM which is processed by the on-board computer in the CDHM. Output signals shall be diagnostic in nature to determine the proper momentum has been generated.

(a) Input signal. The input signal shall be the output of the reaction wheel electronics assembly.

(b) Output signal

Momentum:	7. 2 ft-lb-sec at 1250 rpm
Torque:	20 in. oz
Response:	(torque-speed) TBD
Tachometer:	4.0 ± 0.4 volts at 1000 rpm clockwise
	4.0 ± 0.4 volts at 1000 rpm counterclockwise

(c) Alignment. The reaction wheel shall be within ±15 arc minutes of the normal to the reaction wheel mounting surface.

(d) Component interface characteristics

Size:	120 x 4. 7 in.
Weight:	16. 5 lb
Power:	53 watts

3. 2. 1. 1. 2 Magnetic torquers. Three orthogonal magnetic torquers shall be installed in the AM. The electromagnetic torquers shall be an iron rod wrapped uniformly with a large number of turns of wire conductor. The magnetic torquers shall be energized by commands from the CDHM which is formed by processing signals from the ADM. The magnetic field produced in the rod interacts with the earth's magnetic field to produce the required torque for attitude control.

(a) Magnetic Moment. Each electromagnet shall have the following magnetic moment properties.

Linear magnetic field strength:	120,000 pole-cm at 12 VDC at 0. 5 watt power
Scale Factor:	4200 ± 420 pole-cm/ma
Residual Moment:	less than 1200 pole-cm

These requirements shall be verified by analysis using data supplied from magnetic field tests.

(b) Winding and core

Core material:	Allegheny Ludlum electrical steel 4750	at 72 ± 10° F
Windings:	40,300 turns (No. 32 copper wire)	
Winding resistance shall be 3000 ± 150 ohms.		

(c) Magnetic field. The magnetic field of the unit shall be as indicated in Table 2.

Table 2. Magnetic Field Magnitude

Current (ma)	Field (Milli-Oersteds)
+30	TBD ±10%
+15	TBD ±10%
0	TBD ±10%
-15	TBD ±10%
-30	TBD ±10%

(d) Insulation resistance. Insulation resistance between the coil and the case shall be greater than 10 megohms at 200 VDC.

(e) Thermal requirements. The unit shall be designed to survive the qualification test temperature of -15° F to 135° F and shall perform within the requirement over the operating temperature range of 0° F to 120° F.

(f) Design. The design and dimensions shall be:

Diameter: 1.2 in.

Length: 40 in.

Weight: 8.7 lb

(g) Alignment. The magnetic torquers shall be mounted within TBD of the normal to the magnetic torquer mounting surfaces.

3.2.1.1.3 Valve and magnetic torquer electronics. The valve and magnetic electronics shall provide controlled commands to the thruster valves, magnetic torquers, and module thermal control. The electronic unit shall have the following physical characteristics:

Weight: 1.9 lb

Size: 6 x 8 x 1.6 in.

Power: 3 watt

3.2.1.1.3.1 Magnetic torquer electronics. The magnetic torquer electronics (MTE) shall generate a drive current (Ic) with magnitude and polarity determined by the magnetic moment command. Two complete drivers per each control axis shall be provided and shall operate in an active/standby mode. The current shall be proportional to the magnetic moment command, and the driver circuit shall operate efficiently.

Operating current range:	40 ma
Input impedance:	10/k ohms
Output impendance:	10 ohms maximum
Duty cycle:	0 to 100 percent

3.2.1.1.3.2 Valve drive electronics. The valve drive electronics (VDE) shall provide thruster coil power (28 VDC) in response to commands from the on-board computer or commands from the ADM when operating in the safe Mode. The VDE shall have the capability of driving 14 thruster valve loads plus 9 latching and isolation valves. Valve selection and modes of operation shall be made commandable by inputs. Each valve shall be capable of independent enable and disable. Each of the redundant solenoid valve coils shall be driven by a valve driver circuit.

(a) Input signal. Input signal to the VDE shall be the output of the CDHM and the ADM when operating in the back-up safe Mode.

(b) Electrical

Output voltage:	28 VDC
Output impedance:	1.0 ohms or less
Input voltage:	20 volt minimum
Power:	7 watts average (RCS)

(c) Performance. Rise and fall time shall be 150 microseconds Valve turn-off inductive transient voltage shall be clamped to 45 volts.

3.2.1.1.4 Reaction wheel electronics. The reaction wheel electronics (RWE) shall consist of redundanct pairs to drive each of the three reaction wheels. The RWE shall supply two-phase power to the reaction wheel motor to response to serial digital input from the CDHM. In addition, the RWE shall process the tachometer output of the RWA to a logical signal. Telemetry processing for RWA temperature, motor current, and voltage are required.

(a) Input signal. Input signals to the RWE shall be the output of the CDHM.

(b) Electrical

Square wave 2 phase

56 volts peak to peak

400 Hz

Output impedance 2 ohms

Input impedance 2,000 ohms

Phase A fixed; Phase B shall be reversible

Power 30 watts average

(c) Design

Weight: 12 lbs

Size: 6 x 8 x 10 in

3.2.1.1.5 Data interface unit. The data interface unit (DIU) provides the interfacing electronics between the AM and the CDHM. Two of the units shall be installed in the AM with one of these units being standby redundant of standard design and is described in paragraph 3.2.1.3.4 of SP-1112.

3.2.1.2 Propulsion equipment group

(a) Impulse. The System I orbit adjust engines shall deliver a minimum total impulse of TBD lb-sec to perform the ΔV maneuver. The System II orbit transfer and orbit adjust engines shall deliver a total impulse of TBD lb-sec. Attitude control shall be delivered by cold gas thrusters; total cold gas impulse for System I and System II shall be TBD lb-sec and TBD lb-sec, respectively. The system shall have a propellant storage capacity which allows for a TBD percent contingency growth in total impulse.

(b) Propellants and pressurant. Propellants shall be anhydrous hydrazine per MIL-P-26536 and pressurant shall be gaseous nitrogen per MIL-P-27401.

(c) Operating pressure. The initial gas storage pressure shall be 4000 psia maximum at 120°F. Regulated pressure shall be 350 psia.

(d) Leakage. The total leakage, exclusive of thruster valve seat internal leakage, shall not exceed 10 cc/hr GN_2. Valve seat leakage shall not exceed 5 cc/hr/thruster.

(e) Mass properties

Item	System I		System II	
	Qty	Unit Weight (lb)	Qty	Unit Weight (lb)
Pressurant tank	2	11.25	4	15
Propellant tank	1	8.4	1	65
Regulator	2	1.2	4	1.2
Fill and drain	2	0.5	2	0.5
Pressure transducer	1	0.5	2	0.5
Filters	2	0.5	2	0.5

Item	System I		System II	
	Qty	Unit Weight (lb)	Qty	Unit Weight (lb)
Latching & isolation valve	6	0.6	8	0.6
Vent valve	1	0.6	1	0.6
3.0 lbf thruster	2	0.6	-	-
50 lbf thruster	-	-	2	4.0
1.0/0.1 lbf thruster	12	0.7	12	0.7
Lines and brackets	-	5.0	-	5.0

(f) Power (System I and System II)

Item	Peak Power (watts/valve)	Average Power (watts/unit)
3 lbf thruster valve	6	
1.0/0.1 lbf thruster valve	6	
50 lbf thruster valve	60	
Pressure transducer	0.5*	0.5
Latching isolation valve	12	
Vent valve	13	

*watts/unit

(g) Valve operating signals. The thruster valves shall respond to valve driver signals having the following characteristics. The valve driver load shall be a combination of a TBD ohm resistance and a TBD ± henry inductance, operating in series.

On-state voltage: 28.0 ± 5 percent

Off-state load current: 375 microamperes max

Turn-off transient voltage: 50 ± 5 VDC minus primary voltage, with polarity opposite to that of On-State voltage

Turn-off voltage across load:

Fall time: ≤150 microseconds measured between +20 VDC and -10 VDC load voltage points

Rise time: ≤150 microseconds (10 to 90 percent of total change)

On-state pulsewidths:

50.0 lbf thruster	0.045 second to steady state
3.0 lbf thruster	0.045 second to steady state
1.0/0.1 lbf thruster	0.020 second (nominal)

3.2.1.2.1 Fill and drain valve. The fill and drain valves shall provide a direct interface with the ground servicing equipment for pressurizing and loading the propellant tank and pressurant tank. The valves shall be designed to minimize leakage and to preclude propellant spillage. The valves shall be provided with a cap to protect against contamination and to act as a redundant seal. The valves shall be in accordance with the requirements of Drawing TBD.

3.2.1.2.2 Propellant tank. The propellant tanks shall store both liquid hydrazine propellant and gaseous nitrogen pressurant. Separation of propellant and pressurant shall be maintained by an elastomeric diaphragm.

The System I tank shall have a capacity of TBD pounds of hydrazine and an operating pressure of 350 psia.

The System II tank shall have a capacity of TBD pounds of hydrazine and operating pressure of 350 psia.

3.2.1.2.3 Pressure transducer. The transducer shall monitor gas pressure in the pressurant tank. The pressure range shall be 0 to 4000 psi.

3.2.1.2.4 Latching isolation valve. The system shall use latching isolation valves to protect the system against failed-open or excessive internal leakage conditions in any of the redundant elements of the system. The valves shall employ electrical windings to open and close the valve and shall incorporate a latching mechanism to retain the poppet in the open and closed position. Each valve shall include an integral position indicator and permit flow from the outlet to inlet to prevent a severe pressure rise downstream of the closed valve.

3.2.1.2.5 Filter. The system shall use filters in the outlet line from the propellant tank and the gas tank. The gas filter shall be rated at TBD microns absolute. The propellant filter shall be rated at TBD microns absolute. The filter capacity for the liquid shall be TBD. The filter capacity for the gas shall be TBD.

3.2.1.2.6 Pressure regulator/relief valve. The pressure regulator/relief valve shall reduce the pressure of the gas flowing from the pressurant tank to the gas thrusters and to the propellant storage tank to a nominal pressure of 350 psia. The relief valve which is an integral part of the unit vents the downstream gas in the event that downstream pressure exceeds 360 psia.

3.2.1.2.7 Pressurant tank. The pressurant tank shall store gaseous nitrogen pressurant for pressurizing the hydrazine and for use as the working fluid in the 1.0/0.1 lbf thrusters. The System I tank shall be capable of storing TBD pounds of GN_2 at 4000 psi. The System II tank shall be capable of storing TBD pounds of GN_2 at 4000 psi.

3.2.1.2.8 ΔV thruster. System I shall use two redundant 3.0 lbf thrusters to perform ΔV maneuvers. System II shall use two redundant 50 lbf thrusters. The design and performance requirements shall conform to the requirements of TBD. Performance allocations are listed below:

Thrust vector: Geometric thrust vector shall be perpendicular to the nozzle exit plane within 0.75 degree.

Steady-state thrust and specific impulse: The thruster specific impulse (ISP) thrust, and a function of inlet pressure shall be determined. Steady-state thrust shall be predictable within ±10 percent.

End-of-life characteristics: Thruster performance characteristics over the mission duty cycle shall be defined.

Thrust variability: Engine-to-engine thrust variability shall be ≤5 percent as determined from engine acceptable hot-fire test data.

3.2.1.2.9 1.0/0.1 lbf thruster. The system shall contain twelve 1.0/0.1 lbf thrusters. The function of these thrusters shall be to provide impulse for attitude control. The thruster design and performance shall conform to TBD.

System I will operate the thrusters only at 0.1 lbf thrust. System II will operate the thrusters at 1.0 lbf during ΔV maneuvers and 0.1 lbf when used for control during the space craft operational phase. Performance allocations to the thruster are listed below (worst case per thruster).

Maximum initial starts: TBD

Maximum impulse bit (MIB): TBD

Maximum total impulse: TBD

Cyde life: TBD

A minimum impulse bit of TBD lb-sec produces a maximum torque bit of TBD ft-lb-sec with the thruster located TBD feet from the spacecraft center of gravity.

End-of-life characteristics: Thruster performance characteristics over the mission duty cycle shall be defined.

Impulse variability: ±20 percent for adjacent pulses over full range of operating conditions.

Thrust vector: Geometric thrust vector shall be perpendicular to the nozzle exit plane within 0.75 degree.

3.2.1.3 Harness. The module harness shall provide all electrical interfaces between module assemblies and to the module/structure interface and test connectors. The harness shall be of modular design for maximum system flexibility. Installation or removal of the harness should be possible without removing electrical assemblies. It shall be possible to remove electrical assemblies without removing the harness. Cable strain relief or back-shell potting shall be employed at all harness terminations.

Wire sizes shall be selected to hold round trip voltage drops between source and load to one percent or less of the supply voltage. The minimum wire size for power and control circuitry shall be AWG No. 20. The minimum wire size for data or test circuitry shall be AWG No. 22. Under worst case conditions, wire temperature shall not exceed the temperature rating of the wire insulation.

3.2.1.4 Thermal. The module thermal control system design constraints are presented in the following paragraphs.

3.2.1.4.1 Module thermal requirements. The module thermal design shall satisfy the following on-orbit requirements:

- The module shall be capable of operation when the heat sink temperature is $\pm 20^{\circ}F$ greater than the most severe predicted operating temperatures, where heat sink is defined as the structure or panel to which the electronic black boxes and other module equipment is mounted. These limits will be termed heat sink qualification temperatures. Less severe temperature limits can be used for components that might be damaged by the qualification temperatures if a waiver is obtained from the contractor.

- The module shall be designed so that the nominal set point temperature of the heat sink is 70°F with electrical heaters turned off.

- Electrical heaters shall be incorporated to maintain the orbit-average temperature at the module attachment locations above 60°F with the heater response approximating a sine pulse over one orbital period (rather than a step-input pulse).

- All module heat dissipation shall be radiated to space from the outboard facing panel.

- The surfaces of the module, except for the panel radiator areas, shall be thermally insulated with multilayer insulation, such that the effective emissivity, $\epsilon \leq 0.01$.

3.2.1.4.2 Module/structure assembly thermal interfaces. The design of the module thermal control system shall consider the following interface constraints:

- The structure assembly/module attach point temperature will be 70 ± 10°F.

- Each module attachment fitting on the structure assembly will have a thermal resistance >5 hr-°F/BTU.

- The effective emmitance, ϵ, of the structure assembly/module insulation barrier will be ≤ 0.02.

3.2.1.4.3 Heater power constraints. Module thermal control system heater power shall not exceed 0 watts under normal operating conditions, and 9 watts under the most severe cold operating conditions that consider predictable variations in duty cycle and heating environment as well as parameter uncertainties in thermal properties, heating environment, insulation heat loss, etc.

3.2.1.5 Module structure. The module structure shall support all equipment listed in paragraph 3.1.3 and shall be capable of supporting additional equipment listed in paragraph 3.2.1.9 for modular expansion or complete redundancy.

No amplification of the vibration or acoustic environments shall be caused by the module structure which may result in degradation of the spacecraft performance.

The module structure, when mounted on the spacecraft structure, shall withstand the launch, ascent, and on-orbit loads as defined in SP-1111.

The structure shall not cause a change in alignment of the spacecraft axes by more than TBD arc sec.

The factors of safety shall be no less than 1.00 for limit loads and 1.25 for ultimate loads except where loads may be dangerous to personnel, the ultimate loads shall be 1.50.

3.2.1.6 Useful life. The design of the AM shall be such that wearout of any item or depletion of expendables will not occur prior to a useful life of TBD years. Useful life is defined as the operating time of the equipment counted from the time of launch vehicle liftoff.

3.2.1.7 Storage life. The AM shall have a minimum storage life of 3 years. Storage life critical components may be refurbished.

3.2.1.8 Telemetry. Data necessary for post facto attitude determination and telemetry data shall be primarily limited for those functions necessary for control and operation of the AM during flight. Telemetry indications of equipment status should be as direct as indication as practicable.

3.2.1.9 Expansion capability. A capability shall be provided within the limits of module structure, size, and power available for expanding the baseline configuration to include the addition of redundant units or the addition of new components to perform additional functions.

3.2.2 Physical characteristics

3.2.2.1 Mechanical

3.2.2.1.1 Envelope. The module envelope shall be as shown in ICD 50.1.

3.2.2.1.2 Module volume. The module shall have a volume of approximately 33 cubic feet and a maximum load carrying capability of 600 pounds of equipment. Components may be mounted to the outboard facing panel, nonoutboard surfaces, and the module frame members.

The outboard facing panel may be modified with local cutouts to facilitate assembly or it may be divided into separate equipment heat sink surfaces. Internal stiffness bulkheads and/or mounting panels may be added as required. However, prior to an modifications to the module structure, the subsystem contractor shall perform a structural analysis to ensure structural integrity.

3.2.2.1.3 Module weight. The total weight of the AM for the baseline and expanded configurations shall not exceed the listed weights.

Baseline Module	Expanded Module
TBD	TBD

3.2.2.1.4 Module center of gravity. The center of gravity of the module shall be located within TBD.

3.2.2.1.5 Attach-points. The attach-points between the module and the spacecraft shall be as shown in ICD 50.1.

3.2.2.1.6 Module/structure interface connector. The module/structure interface connector shall be provided by the system contractor and shall be mounted on the side face of the module. It is required that the connector position be maintained as specified to ensure interchangeability of modules.

3.2.2.1.7 Equipment expansion. The components shall be arranged in the module such that the expansion capability requirements of paragraph 3.2.1.9 can be accommodated with minimum impact on the baseline configuration design.

3.2.2.2 Electrical

3.2.2.2.1 Power. Total power required for the AM shall not exceed TBD watts. Allocation of this power is as follows:

Baseline configuration requirement: TBD watts

Redundancy and expansion capability: TBD watts

Power consumption of the AM units shall be within the power allocations in ICD 10.2.

3.2.2.2.2 Commands. Commands for controlling AM operation shall be as listed in ICD 10.3.

3.2.2.2.3 Telemetry. The AM telemetry measurements shall be as listed in ICD 10.4.

3.2.2.2.4 Signal and power distribution. The AM harness shall provide all intramodule electrical connections in conformance to ICD 50.5.

3.2.3 Reliability. The AM shall be capable of performing, as specified, for at least TBD years in orbit. This shall include all redundancy incorporated including alternate and backup modes. Demonstration of compliance with these requirements shall be through reliability analysis as called out in EOS Document EOS-4.1, System Effectiveness Program Plan.

3.2.4 Maintainability. The AM shall be designed in accordance with the requirements of MIL-STD-1472, paragraph 5.9, as implemented by EOS-4.1.

3.2.5 Environmental conditions. The AM shall be designed to withstand or shall be protected against the worst probably combination of environments as specified in SP-11 and as implemented in SP-115.

3.2.6 Transportability. Transportability requirements shall be considered in the design of the electrical power module such that it can be transported by all standard modes with a minimum of special packing or precautionary measures.

3.3 Design and construction

3.3.1 Parts, materials and processes

3.3.1.1 Selection of parts, materials, and processes. Sel ection of parts, materials, and processes (PMP) shall be to the requirements identified in MIL-STD-143 Group 1. Such selections shall be identified as standard PMP. When PMP selection cannot be made within this requirement, the item is nonstandard and justification must be made in accordance with MIL-STD-749. Control of this action shall be effected by the EOS parts, materials, and processes control board (PMPCB), with the PMPCB approving all selections, both standard and nonstandard, in accordance with EOS-4.1.

3.3.1.2 Program authorized parts list. All selections of electronic parts shall be identified and authorized by EOS-3.3-1. This list will reflect all PMPCP electronic part selections.

3.3.1.3 Program authorized materials list. All selections of materials shall be identified and authorized by EOS-3.3-2. This list will reflect all PMPCB material selections.

3.3.1.4 Program authorized processes list. All selection of processes shall be identified and authorized by EOS-3.3-3. This list will reflect all PMPCB process decisions.

3.3.1.5 Dissimilar metals. To avoid electrolytic corrosion, dissimilar metals shall not be used indirect contact unless protection against corrosion has been provided in accordance with MIL-E-8983 and MIL-STD-454, Requirement 16.

3.3.1.6 Magnetic materials. Magnetic materials shall be used only if necessary for equipment operation. Those magnetic materials which are used shall have minimum permanent, induced, and transient magnetic fields.

3.3.1.7 Fungus-inert materials. Materials which are fungus-inert, in accordance with MIL-E-8983 and MIL-STD-454, Requirement 4, shall be used.

3.3.1.8 Flammable toxic and unstable materials. Flammable, toxic and unstable materials shall not be used.

3.3.1.9 Finish. The surface of each component shall be adequately finished to prevent deterioration from exposure to the specified environments that might jeopardize fulfillment of the specified performance. The finish shall also meet the applicable bonding requirements. Thermal properties of the finishes used on the component shall be compatible with the requirements given in detail in applicable equipment specifications. The requirements for finishes identified in MIL-E-8983 shall be implemented.

3.3.1.10 Outgassing. Low outgassing polymeric materials shall be used where sensitive thermal control and other surfaces are in direct line of sight and where temperature differences can exist between such surfaces.

3.3.2 Electromagnetic radiation

3.3.2.1 EMC/EMI requirements. The electrical power module, and all internal units, equipment and/or components comprising a part thereof, shall be designed for compliance with the radiated emission and susceptibility requirements of NASA/JSC Specification SL-E-0002 as modified or amended by EOS-3.3-4 and EOS-3.3-5. The conducted emission and susceptibility requirements of the aforementioned documents are applicable only at the module-to-spacecraft structure interface. EMI/EMS levels at interfaces within the module shall be controlled to the extent necessary to ensure self-compatibility of the module subsystem with a safety margin of at least 6 dB.

3.3.2.2 Electrical bonding

3.3.2.2.1 Structural bonds. All metallic members of the basic electrical power module radiator panel and support structure shall be electrically continuous, equi-potential ground reference plane. The maximum allowable DC impedance across any one structural joint shall be 2.5 milliohms with an overall design goal of 10 milliohms, or less, between any two diametrically opposite points on the module. The general methods of MIL-B-5087, Class R, may be used for implementation of these requirements.

3.3.2.2.2 Equipment mounting pads. Surfaces on the module radiator panel and structure which are intended for unit, equipment, or component mounting shall be free of paint, anodize, or other nonconductive finishes. The maximum DC impedance between the baseplate of any electrically active unit, equipment or component and the module radiator panel and structure shall be 2.5 milliohms.

3.3.2.2.3 Electrical connectors. All interface electrical connectors both plug and receptacle, which form a part of the unit and cable RF shielding system within the module shall have electrically conductive body shells, free of nonconductive finishes. They shall also have provisions for terminating (grounding) the shields on the interconnecting electrical harness. The maximum DC impedance between the shield termination point and the baseplate of the parent unit shall be 2.5 milliohms.

3.3.2.2.4 Electrostatic bonds. Electrically passive components or appurtenant structures which are attached to the basic module structure through thermal isolators shall be provided with electrostatic bonds having a DC impedance of 1.0 ohm or less. The general methods of MIL-B-5087, Class S, may be used for implementation of this requirement.

3.3.2.3 Electrical systems grounding

3.3.2.3.1 Primary DC power. The electrical power module shall provide the single-point structural ground for the primary DC power distribution subsystem. This subsystem will be DC isolated from structural grounds in all user modules as well as in the solar array. The physical location of the ground point may be within, or immediately adjacent to the battery assembly. Interconnecting primary power wiring between units and/or assemblies of the electrical power module shall be unshielded, twisted pairs.

3.3.2.3.2 Secondary DC power. Secondary DC power distribution networks shall, in general, use multiple point grounding within the electrical power module with returns through the module radiator panel and/or support structure. These networks shall be initially grounded adjacent to the secondary transfomer winding in the power converter and at each user element.

3.3.2.3.3 Secondary AC power. Secondary AC power networks shall use single point grounding with a two-wire, twisted shielded pair distribution. The ground point may be either at the source or load end, whichever is shown by circuit analysis to be most beneficial to compatible module operation. Structural returns shall not be used for secondary AC power.

3.3.2.3.4 Intramodule signal/control circuit (high level). All high level (<5 volt logic, bilevel or analog) signal or control circuits which do not exit the electrical power module shall be multiple point grounded at both the source and load end of each circuit branch to the unit or component chassis by the shortest most direct path. Circuits sharing space on a common printed circuit board should not share common grounding traces on the board or common hardwire jumpers to chassis ground logs. Preferably, each such board should have a dedicated ground plane layer to which all components requiring ground returns can be directly connected. This ground plane should, in turn, be directly bonded to unit chassis or frame through grounding pads at each hold-down fastener.

3.3.2.3.5 Intermodule signal/control circuits (high level). All high level signal or control circuitry which exists the electrical power module shall be grounded at the final driving element. Two-wire, twisted shielded pair distribution shall be used for each such circuit between the source unit and the module-to-spacecraft interface connector.

The load elements in the external module will be DC isolated from structural grounds. Any signal or control circuitry which enters the electrical power module shall be DC isolated from chassis/case/structure ground by a minimum of 1 megohm resistance. Any such circuits shall also be provided with two-wire, twisted shield pairs between the interface connector and the load unit.

3.3.2.3.6 Analog circuits (low level). Any low level (<5 volts) analog circuits, which are shown by circuit analysis or test to be sensitive to circulating currents in the module or spacecraft structure, shall be single-point grounded either at the source or load element, whichever is most appropriate for the circuit under consideration. Wherever possible, balanced differential circuitry should be used. In the case of low level circuits which enter or exit the electrical power module, the location of the circuit ground point shall be coordinated with the systems integration contractor.

3.3.2.3.7 Data bus. The command and telemetry data bus system shall be differentially driven and balanced to structural ground in the communication and data handling module. This system shall be transformer-coupled at each remote terminal. Each individual data bus wire entering the electrical power module shall be DC isolated from chassis/case/structure by a minimum of 1 megohm resistance.

3.3.2.3.8 Wire shields. External shields shall be provided for all interconnecting wires between units, equipment or components in the electrical power module and between each input/output connector and the module-to-spacecraft interface connectors except for input and output primary DC power lines. In general, these shields shall be multi-point grounded at each end and at each intermediate interface. An exception to this rule will be allowed for low level analog circuitry where single-point shield grounding may be necessary. If possible, such circuitry should be provided with two mutually-isolated shields, the inner shield being single-point grounded and the outer shield multi-point grounded.

3.3.2.3.9 EMI filter components. High performance EMI filters will be required at the primary DC power input and return terminals of each power converter unit or sub-unit in the electrical power module to ensure compliance with the electromagnetic interference requirements of the applicable specifications. These filters should have two stages: (1) an AF ripple filter stage balanced line to line; and (2) a pair of RF feedthrough filters bulkhead-mounted behind the input connector. The combined AF/RF filter circuit should be designed for the minimum capacitance from either line to chassis necessary to achieve compliance with the specifications. Excessive line-to-ground capacitance will tend to negate the beneficial effects of the twisted pair wiring used in the primary DC power harness. Similar filtering may be required at the input and output terminals of the pulsewidth switching regulator assembly; however, because of fault isolation requirements, line-to-case feedthrough RF filters should not be used.

3.3.3 Nameplates and product marking. Each unit shall be identified in accordance with MIL-STD-130 as implemented by EOS-3.3-6.

Where practical, a minimum character height of 0.09 inch shall be used. Marking shall include the manufacturer's name or initials, assembly part number and revision status, assembly serial number, contract number, and others as delineated in the component equipment specifications.

3.3.4 Workmanship. The AM and its component equipment shall be constructed, finished, and assembled in accordance with MIL-STD-454. Requirement 9, and the specifications and drawings specified herein.

3.3.5 Interchangeability and replaceability. The electrical power module shall be designed to permit removal and replacement of components with a minimum of disturbance of associated or adjacent equipment. Design for service and access shall conform to the principles and requirements of MIL-STD-470.

3.3.6 Safety. The electrical power module shall be designed to meet or exceed the requirements of EOS-3.3-7, as implemented by EOS-4.1, System Effectiveness Program Plan. The design criteria include but are not limited to those set forth in MIL-STD-882.

3.3.7 Human performance/human engineering. MIL-STD-1472A, "Human Design Criteria for Military Systems," shall be used for the design of man/machine interfaces.

3.4 Documentation. Documentation shall be as defined in EOS-3.3-8.

4. QUALITY ASSURANCE PROVISIONS

4.1 General. The quality assurance program controls shall be in accordance with the requirements of MIL-Q-9858 as augmented by EOS-4.1.

4.1.1 Responsibility for inspection and test. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified herein. Except as otherwise specified, the supplier may use his own facilities or any commercial laboratory acceptable to the government. The government reserves the right to perform any of the inspections set forth in the specifications where such inspections are deemed necessary to ensure that supplies and services conform to prescribed requirements.

4.2 Quality conformance inspections

4.2.1 Category I tests

4.2.1.1 Development tests. Development tests shall be conducted to verify the module performance parameters, proper interfacing between components of the mocule, and proper interfacing between the module and other spacecraft modules. The development tests shall be conducted at the unit and/or integrated module levels. The type of developmental evaluation in this specification are outlined in Table 3. The tests shall be conducted in accordance with EOS-4.2 and EOS-4.3.

Table 3. Verification Cross Reference Index

VERIFICATION CROSS REFERENCE INDEX										
LEGEND: N/A - Not Applicable; I - Inspection; A - Analysis; S - Similarity; T - Test										
SECTION 3 REQUIREMENTS	Not Applicable	Acceptance			Qualification				SECTION 4 VERIFICATION REQUIREMENTS
		I	A	T	I	A	S	T	
(TBD)									

SAMPLE

4.2.1.2 Qualification tests. Module qualification shall be accomplished by means of qualification tests on the individual units and/or as a normal consequence of having the units installed in the qualification test space vehicle during its qualification testing. The types of qualification evaluation or test to be applied to verify satisfaction of the requirements of this specification are outlined in Table 3. Qualification test verification methods and requirements shall be as defined in EOS-4.2.

4.2.1.2.1 Components. As a minimum, the following types of component qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, are measured before and after the environmental tests.

- Random vibration. Each component will be subjected to a random vibration from 20 to 2000 Hz in all three axes while electrically powered in the maximum normal load condition. Performance will be continuously monitored to detect failures (either hard or trends).

- Thermal vacuum. Thermally cycle the component beyond expected orbital temperatures while at an orbital vacuum condition. The component is operating continuously and detailed performance parameters are measured at each temperature extreme.

- Electromagnetic interference (EMI) and susceptibility EMS). Engineering EMI and EMS data are measured to provide diagnostic information for the module EMI/EMS qualification.

4.2.1.2.2 Module. As a minimum, the following types of module qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, and electrical interface characteristics are measured before and after the environmental tests.

- Acoustics. Acoustics testing will be performed with the module mounted in a receptacle which provides acoustics shielding similar to the Observatory structure. The module will be heavily instrumented to verify design adequacy of interconnections (electrical and mechanical) between the module and other Observatory elements. The test data will be used to confirm estimates of the vibration environment for components mounted on radiators and establish procedures for acceptance testing. The module will be electrically powered and continuously monitored via the data bus to detect failures and out-of-tolerance performance trends.

- Thermal vacuum. The objectives of the module thermal qualification test are to determine: 1) maximum and minimum operating temperatures for the module for worst-case environments and operating duty cycles, 2) temperature levels and temperature variations of the module adjacent to the interface fittings, and 3) heater power requirements for cold-case conditions.

The test will be conducted in a thermal-vacuum chamber with liquid nitrogen cold walls. The module will be mounted on a fixture simulating the spacecraft frame. The properties of this fixture shall permit its temperature to be held constant at any level between 0 and 100°F.

A heating source for the radiators will approximate the absorbed flux of the external environment. This can be done with electrical heaters, infrared lamps, or other techniques where the absorbed heating can be determined accurately.

- Power bus and data bus will be tested in excess of their operational limits to determine design margins and compliance with the interface specification.

- Detailed performance data will be measured to determine module specification values.

- Thermistor/heater control and calibration will be determined.

• EMI and EMS. EMI measurements will determine the levels radiated on each module electrical interface and verify there is adequate design margin to ensure non-interference with other modules. EMS measurements will verify that each module has adequate susceptibility margin to prevent performance degradation resulting from other module allowable radiation levels. The general test methods of MIL-STD-462, as modified or amended by EOS-3.3-5, shall be used during this test program except that, wherever practical, spectrum analyzers or other forms of rapid display and analysis equipment may be used in lieu of equipment requiring manual tuning and data collection.

4.2.1.2.3 Inspection sequence. The inspection sequence shall be governed by the following:

• Examination of the module shall be performed prior to functional testing.

• Functional tests shall be performed prior to, during, where appropriate, and following environmental testing. The functional testing conducted at the conclusion of an environmental test may serve as the functional testing to be performed prior to the next environmental test.

• Environmental testing may be performed in any sequence.

4.2.1.2.4 Failure criteria. The module shall exhibit no failure, malfunction or out-of-tolerance performance degradation as a result of the examinations and test specified herein. Any such failure, malfunction or out-of-tolerance performance degradation shall be cause for rejection.

4.2.2 Test conditions. Unless otherwise specified, the conditions under which all inspections are accomplished shall be specified below:

4.2.2.1 Atmospheric and environmental. All examinations and tests shall be conducted under the local prevailing pressure at the time of test, at a temperature of 75 ±5°F and at a relative humidity of 55 percent or less.

4.2.2.2 Deviations of atmospheric conditions. When tests are performed with atmospheric conditions substantially different from the specified values, proper allowances for changes in instrument readings shall be made to compensate for the deviation from the specified conditions.

4.2.3 Equipment warmup time. The equipment warmup time shall be less than 1 minute.

4.2.4 Temperature stabilization. Temperature stabilization shall have been achieved when temperature of the equipment mounting structure varies not more than 5°F during a period of 15 minutes.

4.2.5 Measurements. Measuring instruments used to determine functional parameter values (such as voltage, frequency, current, flow, pressure, etc.) shall indicate true values with an accuracy determined by the tolerance allowed for the parameter variation itself, such that the measuring instrument shall not introduce an uncertainty greater than 10 percent of the allowable variation of the measured parameter. However, no such measurement accuracy shall be required to exceed 0.5 percent of the required value of the parameter unless otherwise specified.

Except as specifically noted in the inspection methods, tolerances for environmental test conditions shall be defined as follows:

Temperature: ±5°F

Barometric pressure: ±10 percent

Relative humidity: +5, -10 percent

Time: ±5 percent

Vibration amplitude (sine): ±10 percent

Vibration frequency: ±2 percent

4.3 Category II tests. (Not applicable)

4.4 Analyses. The following requirements of Section 3 shall be verified by review of analytical data:

Reliability. To be verified by analysis in accordance with Section 3.2.3 of EOS-4.1.

Safety. To be verified by analysis in accordance with Section 3.3.6 of EOS-4.1.

5. PREPARATION FOR DELIVERY

5.1 General. The AM as specified herein shall be protected from degradation by environments anticipated during shipment, handling, and storage. Standard commercial packaging practices are acceptable provided they fulfill these requirements.

5.2 Preservation and packaging

5.2.1 Cleaning. The AM shall be clean and free of contaminants which might impair its use prior to packaging.

5.2.2 Attaching parts. When attaching parts, such as nuts, bolts, washers, etc., accompanying the AM, they shall be preserved, bagged, appropriately identified, and attached to or adjacent to the fitting for which they are intended.

5.2.3 Electrical connectors. All electrical connectors shall be capped with protected dust caps. Caps used shall be a friction fitting or threaded type which do not require tape or a mechanical defice to secure.

5.2.4 Critical surfaces. External machined surfaces and mounting surfaces of the AM shall be protected with protective pads. Materials used for pads shall not cause item deterioration.

5.2.5 Wrapping. The AM shall be wrapped or bagged using anti-static polyethylene film.

5.2.6 Cushioning. When required for protection, the AM shall be cushioned using a suitable resilient foam cushioning material such as polyethylene, polyurethane, or polystyrene.

5.3 Packing. The AM shall be packed in a suitable shipping container designed for one item only. The shipping container used shall provide protection to the item against corrosion, deterioration, and damage during shipment from the source of supply to the receiving activity. Containers shall comply with applicable tariffs and regulations for particular modes of transportation when so shipped.

5.3.1 Storage conditions. The AM shall not be adversely affected by storage within its container at temperatures between 60°F and 90°F and relative humidities of 60 percent or less.

5.3.2 Shipping conditions. The AM shall be capable of withstanding the following environments:

Temperature:	+160°F in an unsheltered area (125 +35°F due to solar radiation) and -40°F in accordance with restricted air transport criteria
Humidity:	Up to 100 percent of an unsheltered area
Rough handling:	Capable of withstanding and physically protecting the unit from rough handling during shipping by common carrier

5.4 Marking for shipment. Each AM and shipping container shall be marked with the following:

(a) Item nomenclature

(b) Systems part number

(c) Contractor or purchase order number

(d) Manufacturer's name

(e) Manufacturer's part number and serial number (on item container only)

(f) Quantity

(g) Date of manufacture (on item container only)

(h) Fragile — Handle With Care (when applicable)

(i) Space Vehicle Material — Do Not Open in Receiving Or Receiving Inspection (when applicable — shipping container only)

(j) Actual weight.

5.4.1 Documentation. All required reliability and test documentation such as test reports, certifications, shipping invoices, etc., shall be either packed in the AM container or attached to the exterior surface of the shipping container. Attachment shall be such as to preclude loss of these data during handling and shipment by common carrier.

TITLE

PRIME ITEM DEVELOPMENT SPECIFICATION
FOR THE

SOLAR ARRAY AND DRIVE MODULE

DATE 20 SEPT 1974 NO. SP-1116

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

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SPECIFICATION SP-1116

SOLAR ARRAY AND DRIVE

1. SCOPE

This specification establishes the requirements for the design, development, and performance of the solar array and drive module for the EOS multimission spacecraft.

2. APPLICABLE DOCUMENTS

2.1 Government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

SPECIFICATIONS

NASA

SP-11	Observatory Segment Specification
SP-115	EOS Environmental Specification
SP-1111	Structure Assembly Specification
SP-1112	Communication and Data Handling Specification
SP-1113	Electrical Power Specification

Military

MIL-B-5087	Quality Program Requirements Bonding, Electrical, and Lightning Protection, for Aerospace Systems
MIL-Q-9858A	Quality Program Requirements
MIL-E-8983A	Electronic Equipment, Aerospace Extended Space Environment, General Requirements

STANDARDS

Military

MIL-STD-130	Name Plates

MIL-STD-143B	Standard and Specifications, Order of Precedence for Selection
MIL-STD-454C	Standard General Requirement for Electronic Equipment
MIL-STD-462	EMI/EMS Test Methods
MIL-STD-470	Maintainability Program Requirements (For Systems and Equipments)
MIL-STD-749B	Preparation and Submission of Data for Approval of Nonstandard Electronic Parts
MIL-STD-882 15 July 1969	System Safety Program for Systems and Associated Subsystems and Equipment, Requirements for
MIL-STD-1472A	Human Engineering Design Criteria for Military Systems, Equipment, and Facilities

OTHER PUBLICATIONS

NASA

SL-E-0002	Electromagnetic Compatibility Control Plan
NASA STDN101.1 X-560-63-2	STDN User's Guide Aerospace Data Systems Standards
EOS-3.3-1	Program Authorized Parts List
EOS-3.3-2	Program Authorized Materials List
EOS-3.3-3	Program Authorized Processes List
EOS-3.3-4	Electromagnetic Compatibility Control Plan
EOS-3.3-5	EMI/EMS Limits and Test Methods
EOS-3.3-6	Marking of Parts and Assemblies

2.2 Non-government documents. The following documents of the exact issue shown form a part of this specification to the extent specified herein. If no revision or data is shown, the latest released issue of the applicable document shall apply. In the event of conflict between the documents referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

DRAWINGS

10.2	Satellite Primary Power Allocation, ICD
10.3	Satellite Command Allocation, ICD
10.4	Satellite Telemetry Allocation, ICD
30.1	Solar Array and Drive Module Envelope, ICD
30.5	Solar Array and Drive Module Wire List
TBS	Solar Array and Drive Power Module Assembly Drawing

OTHER PUBLICATIONS

EOS-3.3-7	Safety Design Criteria
EOS-3.3-8	Configuration Management Plan
EOS-4.1	System Effectiveness Program Plan
EOS-4.2	Integrated Test Plan

3. REQUIREMENTS

The EOS solar array and drive module shall provide the following major functions:

(a) Power generation

(b) Solar array orientation

(c) Spacecraft structure heater control.

The solar array and drive module shall be capable of generating all power during periods of illumination and maintaining array orientation to sun-normal within ± (TBS) degrees. The solar array portion of the solar array and drive module shall be capable of supporting observatory loads between 0.3 kw and 2.0 kw by addition or removal of modular array sections. Such additional module sections shall be identical in configuration and manufacture so as to be interchangeable with other sections. The specific configuration and equipment complement of the solar array and drive module shall be mission dependent.

3.1 Item definition. The solar array and drive module is required to provide power generation during periods of illumination in support of the observatory power requirements. Primary unregulated power is routed from the solar array and drive module to the electrical power module via the spacecraft harness. Transfer of power across the solar array drive interface shall be accomplished using at least four (4) pairs of slip-rings. Isolation of the solar array power outputs shall be obtained by the use of switches. Solar array drive and drive electronics assemblies shall be provided to orient the solar array towards the sun for maximum power output.

The solar array and drive module shall also provide for control and fault protection of the observatory structure heaters for each mission.

Umbilical interfaces to the solar array and drive module shall be provided to enable control of the solar array disconnect from a remote control panel.

3.1.1 Block diagram. The block and interface diagram of Figure 1 illustrates the components and interfaces necessary to implement the solar array and drive module functions.

3.1.1.1 Solar array drive. This assembly contains the motors, gear reduction, potentiometers, slip-ring subassembly, and the necessary housings, bearings, and connectors to provide for the rotation of the solar arrays for maximum power output. In addition, this assembly provides for power and signal transfer from the solar arrays across the rotating joint to the spacecraft body. The drive motors and position feedback potentiometers shall be redundant and redundant brushes shall be employed on all slip-ring circuits.

3.1.1.2 Solar array drive electronics. This assembly consists of two units with one subassembly for each of the redundant array drive motors. The unused motor drive electronics subassembly is in a standby condition while the other subassembly is in operation. The subassembly produces a low frequency sine/cosine excitation for synchronous operation of the two-phase drive motor. The signal frequency shall be variable by a ground commandable frequency divider.

3.1.1.3 Bus protection assembly. This assembly contains the fuses required to provide fault protection for the primary power bus. Fault isolation shall be incorporated for all functions internal to the solar array and drive module as well as the primary power outputs to the structure heaters.

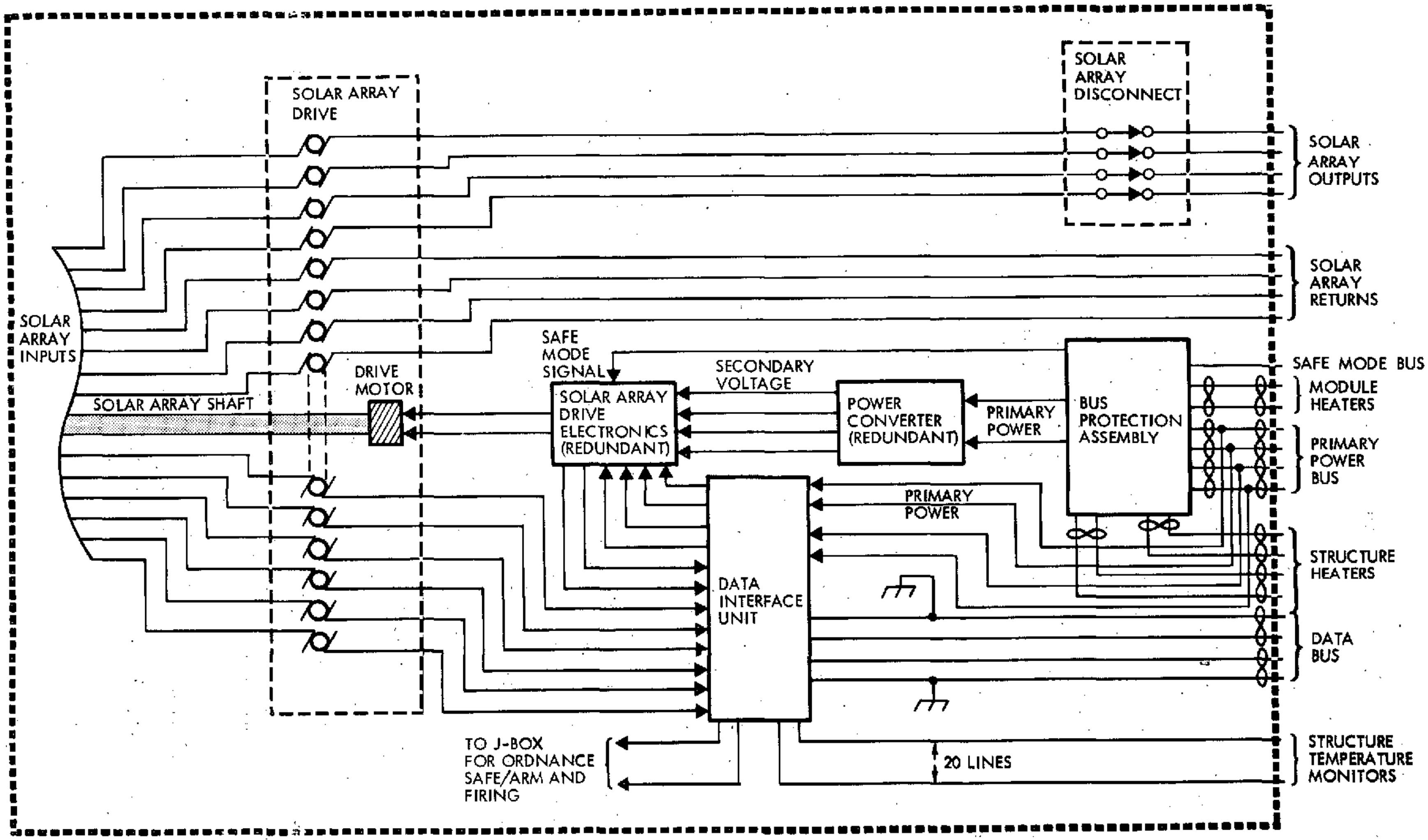

Figure 1. Solar Array and Drive Module Interface Diagram

3.1.1.4 Power converter. This assembly provides all secondary power conversion necessary to the operation of the solar array drive electronics. The power converter shall incorporate redundancy to ensure capability of achieving and sustaining the observatory safe mode.

3.1.1.5 Solar array disconnect assembly. The power disconnect assembly contains the relays and electronic drive circuitry necessary to provide isolation of the solar array power outputs.

3.1.1.6 Data interface unit (DIU). The data interface unit provides the digital-to-bilevel and analog-to-digital conversion necessary for the solar array and drive module interface with the on-board computer. The data interface unit derives power directly from the main bus and is functionally independent of the solar array and drive module.

The data interface unit shall contain all conversion, conditioning, and bus protection equipment necessary for the operation of the data interface components, and shall be a complete assembly delivered ready for installation into the solar array and drive module.

3.1.2 Interface definition.

3.1.2.1 Internal interfaces. The interfaces between assemblies of the solar array and drive module shall be interconnected by means of the harness assembly entirely within the solar array and drive module.

3.1.2.1.1 Solar array/solar array disconnect. The unregulated primary power generated by the solar array shall be routed through the slip rings in the solar array drive assembly to the solar array disconnect. Interconnection of the various array outputs at the slip rings shall be such that at least two electrically independent, isolated array sections are identifiable. This independent interconnection shall be maintained through the solar array disconnect assembly for routing to the electrical power module. The current rating of any one slip ring to which the interconnected array sections are connected shall be at least equal to the current capability of the connected array sections at their maximum power condition.

3.1.2.1.2 Solar array drive/solar array drive electronics. One subassembly of the solar array drive electronics shall be used to drive one motor and process one potentiometer in the solar array drive. The redundant subassembly of the solar array drive electronics shall be in the standby mode and used for the redundant motor and potentiometer in the solar array drive. In the event of a failure in either the operating array drive or drive electronics assemblies, the spacecraft shall be commanded to operate with the redundant array drive and drive electronics assemblies.

3.1.2.1.3 Power converter/solar array drive electronics. The secondary voltages necessary to the operation of the solar array drive electronics shall be derived within the redundant power converter assembly and routed to the solar array drive electronics. Redundancy of the power converter shall be independent of the redundancy within the solar array drive electronics. The outputs of either power converter shall be available for use in either solar array drive electronics.

3.1.2.1.4 Bus protection assembly/power converter. The primary power buses shall be routed through fault isolation devices in the bus protection assembly to the inputs of the redundant power converters.

3.1.2.1.5 Data interface unit/solar array drive electronics. The command inputs and telemetry outputs of the solar array drive electronics shall be routed through the data interface unit for transmission via the data bus.

3.1.2.1.6 Data interface unit/solar array. The data interface unit shall include an interface with the solar array via signal slip rings in the solar array drive for transmission of array telemetry to the data bus.

3.1.2.2 External interfaces.

3.1.2.2.1 Primary unregulated power. Unregulated primary power generated by the solar array shall be routed from the solar array disconnect to the spacecraft J-Box and hence to the electrical power module.

3.1.2.2.2 Secondary power. Secondary power shall not be distributed outside the solar array and drive module.

3.1.2.2.3 Mechanical Interfaces. The solar array and drive module shall meet the requirements of ICD 30.1

3.1.2.2.4 Functional interfaces. The external interfaces of the solar array and drive module exclusive of the solar array outputs shall include data bus, safe mode bus, primary power and module heater buses, structure heater outputs, structure temperature monitor inputs, and ordnance control and firing commands to the transition ring J-Box.

3.1.2.2.4.1 Solar array and drive/communication and data handling module. The solar array and drive module shall include a data interface unit for receipt of commands from and transmission of telemetry to the communication and data handling module via the Observatory data bus.

3.1.2.2.4.2 Solar array and drive module/safe mode bus. The solar array and drive module shall incorporate logic to sense the safe mode bus station and affect the Safe Mode configuration in the solar array drive electronics when the safe mode bus has been enabled.

3.1.2.2.4.3 Primary power and module heater buses. The solar array and drive module shall derive primary power for its operation from the redundant spacecraft primary power buses. Power for the maintenance of thermal control within the solar array and drive module shall be derived from the Observatory module heater bus.

3.1.2.2.4.4 Bus protection assembly/structure heaters. The primary power to the structure heaters shall be routed through individual fault isolation devices in the bus protection assembly. Required control of power to the structural heaters shall be accomplished within the bus protection assembly.

3.1.2.2.4.5 Data interface assembly/structure temperature monitors. The data interface assembly shall include an interface with the spacecraft structure for transmission by telemetry of selected temperature monitor points on the Observatory.

3.1.2.2.4.6 Data interface unit/ordnance control circuits. The data interface shall incorporate a command output interface for control and firing of the Observatory ordnance devices, and a telemetry input interface for monitoring of the safe/arm status of the Observatory ordnance devices.

3.1.3 Government-furnished module equipment. Certain items of module equipment which are identical or nearly identical in function will be fabricated by the integration contractor and provided as GFE to the remaining module contractors. This equipment shall include:

Item	Reference
Data interface unit	3.2.1.3.4 of SP-1112
Power control unit	3.2.1.9 of SP-1113

All module structures and integration of the modules will be provided by the integrating contractor.

3.1.4 Major component list. The major components of the solar array and drive module for the minimum redundancy and nominal redundancy configurations shall consist of the following. (See 6.1 for definition.)

	Minimum Redundancy	Nominal Redundancy
Solar array	1	1
Solar array drive	2	2
Solar array drive electronics	2	2
Power converter (redundant)	2	2
Bus protection assembly	1	1
Data interface unit	1	2
Solar array disconnect	1	1
Solar array and drive module harness	1	1

	Minimum Redundancy	Nominal Redundancy
Solar array and drive thermal control	1	1
Solar array and drive module structure	1	1
Solar array structure	1	1

3.2 Characteristics.

3.2.1 Performance characteristics

3.2.1.1 General. The solar array and drive module shall be capable of generating and transmitting adequate power for steady-state and transient loads to the Observatory during periods of illumination on-orbit. During normal on-orbit operation with illumination, the solar array and drive module shall generate sufficient power to enable all spacecraft and payload modules to operate with the required duty cycle.

The specific equipment complement for the solar array and drive module shall be mission dependent, provided that the capability shall exist without major redesign to direct and maintain orientation of arrays power generating capabilities between 0.5 and 1.5 kw for the baseline mission and up to 3.0 kw for an expanded mission.

As a minimum, the solar array and drive module shall use redundancy to ensure that the Observatory safe mode can be enabled and maintained until servicing or retrieval can be accomplished. No single-point failure mode shall exist that could cause mission failure or prevent enabling and maintaining the safe mode. Current limiting or fuse devices shall be used to protect the primary power buses from short circuits occurring within the components of the solar array and drive module.

3.2.1.2 Solar array. The baseline solar array shall provide a minimum output of 1000 watts at 33 volts after 2 years in space in a 717 km (387 n. mi.), 98.4 degree inclination orbit (1100 a.m. equatorial crossing) at equinox. The array shall be composed of identical flat panels. A panel shall be composed of identical subpanels for ease of manufacture and test. A subpanel shall not exceed one-twentieth of the total baseline array area.

The solar array may be expanded in size to yield a maximum output of 3 kw in the baseline orbit at beginning-of-mission by the addition of solar panels.

3.2.1.2.1 Subpanel. A subpanel shall contain two circuits of solar cells, each connected to the main power bus through redundant diodes connected to the positive (+) side of the circuit. A circuit shall contain a sufficient number of series cells to ensure that the maximum power point voltage of the circuit is equal to or greater than 33 volts after two years in the orbit described in paragraph 3.2.1.2. A circuit shall contain a sufficient number of parallel solar cells, interconnected at the cell level, to preclude the entire loss of the circuit under conditions of partial shadowing or of the loss of one cell in a parallel group due to an open-circuit failure.

The subpanel shall be designed to survive either of the following temperature cycling environments with a loss of electrical power at 33 volts of 2 percent or less.

(a) Low orbit environment. 12,000 temperature cycles from -112 to +176°F with a minimum rate of temperature change on cooling of 20°F/minute.

(b) Geosynchronous orbit environment. 200 temperature cycles from -256 to +140°F with a minimum rate of temperature change on cooling of 100°F/minute.

The solar cells shall be protected against short-circuit failures by an insulating layer between the subpanel substrate and the solar cell adhesive. All wires and jumpers shall be attached to the subpanel with an adhesive that meets the requirements of paragraph 3.3.1.10.

3.2.1.2.2 Solar cells. The solar cell shall be an n/p gridded silicon cell with fully soldered titanium-silver contacts. The cell shall have a silicon monoxide antireflection coating. The cell shall have a maximum size of 2 x 6 x 0.0356 cm. It shall be bonded to the subpanel with a silicone adhesive that meets the requirements of paragraph 3.3.1.10.

3.2.1.2.2 Coverslides. The solar cell coverslides shall be made of 7940 fused silica with a blue reflective coating and a MgF_2 antireflective coating. The coverslide dimensions shall be selected to provide protection against low-energy protons in a geosynchronous orbit application by overhanging the edges of the solar cell and the N-contact strip. The coverslide shall be 0.030 cm (0.012 inch) thick. The coverslide shall be bonded to the solar cell with an optical-grade clear silicone adhesive that meets the requirements of paragraph 3.3.1.10.

3.2.1.3 Solar array drive. The solar array drive shall be capable of orienting the array toward the sun and transferring power from the solar array to the electric power module. The specific requirements on the solar array drive are as follows:

Load inertia:	75 ft-lb-sec^2, minimum
Transverse stiffness:	1 x 10^6 in.-lb/rad, minimum
Torsional stiffness:	1 x 10^5 in.-lb/rad, minimum

Maximum allowable step motion:	0.0002 deg
Position indication:	0.02 deg, maximum
Slip-ring circuits:	4 power circuits at TBD amp each 8 signal circuits at 1 amp each

3.2.1.4 Solar array electronics. The solar array electronics shall accept +15, -15 and +5 volt signals and a clock input and provide the sine/cosine voltages to the two phases of the solar array drive motors for smooth continuous rotation. The drive rate shall be adjustable by ground command by the use of a variable modulo frequency divider on the input high frequency clock signal.

3.2.1.5 Power converter. The power converter assembly of the solar array and drive module shall provide redundant DC/DC and DC/AC conversion necessary to generate the secondary power required by the solar array drive electronics.

3.2.1.6 Bus protection assembly. The bus protection assembly shall provide fault protection for the spacecraft primary power buses from load faults occurring either within the solar array and drive module or on the structure heater power lines supported by the solar array and drive module.

3.2.1.7 Data interface unit. The data interface unit (DIU) shall provide the functions necessary to support the transfer of commands and telemetry between the solar array and drive and communication and data handling modules. Performance of the DIU shall meet the requirements of EOS Specification SP-1112.

3.2.1.8 Solar array disconnect assembly. The solar array disconnect assembly of the solar array and drive module shall provide for the reliable interruption and restoration of the unregulated primary power from the solar array. Protection shall be incorporated to prevent disconnecting the solar array except by command.

3.2.1.9 Harness. The module harness shall provide all electrical interfaces between assemblies within the module and to the module/structure interface and test connectors. The harness shall be of modular design for maximum system flexbility. Installation or removal of the harness should be possible without removing electrical assemblies. It shall be possible to remove electrical assemblies without removing the harness. Cable strain relief or backshell potting shall be employed at all harness terminations.

Wire sizes shall be selected to hold round trip voltage drops between source and load to one percent or less of the supply voltage. The minimum wire size for power and control circuitry shall be AWG No. 20. The minimum wire size for data or test circuitry shall be AWG No. 22. Under worst-case conditions, wire temperature shall not exceed the temperature rating of the wire insulation.

3.2.1.10 Thermal. The module thermal control system design constraints are presented in the following paragraphs.

3.2.1.10.1 Module thermal requirements. The module thermal design shall satisfy the following on-orbit requirements:

- The module shall be capable of operation when the heat sink temperature is ±20°F greater than the most severe predicted operating temperatures, where heat sink is defined as the structure or panel to which the electronic black boxes and other module equipment is mounted. These limits will be termed heat sink qualification temperatures. Less severe temperature limits can be used for components that might be damaged by the qualification temperatures if a waiver is obtained from the contractor.

- The module shall be designed so that the nominal set point temperature of the heat sink is 70°F with electrical heaters turned off.

- Electrical heaters shall be incorporated to maintain the orbit-average temperature at the module attachment locations above 60°F with the heater response approximating a sine pulse over one orbital period (rather than a step-input pulse).

- All module heat dissipation shall be radiated to space from the outboard facing panel.

- The surfaces of the module, except for the panel radiator areas, shall be thermally insulated with multilayer insulation, such that the effective emissivity, $\epsilon \leq 0.01$.

3.2.1.10.2 Module/structure assembly thermal interfaces. The design of the module thermal control system shall consider the following interface constraints:

- The structure assembly/module attach point temperature will be 70 ±10°F.

- Each module attachment fitting on the structure assembly will have a thermal resistance > 5 hrs-°F/Btu.

- The effective emittance, ϵ, of the structure assembly/module insulation barrier will be ≤ 0.02.

3.2.1.10.3 Heater power constraints. Module thermal control system heater power shall not exceed 0 watts under normal operating conditions, and 3 watts under the most severe cold operating conditions that consider predictable variations in duty cycle and heating environment as well as parameter uncertainties in thermal properties, heating environment, insulation heat loss, etc.

3.2.1.11 Module structure. The module structure shall support all equipment listed in paragraph 3.1.4 and shall be capable of supporting additional equipment listed in paragraph 3.2.1.16 for modular expansion or complete redundancy.

No amplification of the vibration or acoustic environment shall be caused by the module structure which may result in degradation of the spacecraft performance.

The module structure, when mounted on the spacecraft structure, shall withstand the launch, ascent, and on-orbit loads as defined in SP-1111.

The structure shall not cause a change in alignment of the spacecraft axes by more than TBD arc seconds.

The factors of safety shall be no less than 1.00 for limit loads and 1.25 for ultimate loads except where loads may be dangerous to personnel, the ultimate loads shall be 1.50.

3.2.1.12 Useful life. The design of the solar array and drive module shall be such that wearout of any item will not occur prior to a useful life of TBD years. Useful life is defined as the operating time of the equipment counted from the time of launch vehicle liftoff.

3.2.1.13 Storage life. The solar array and drive module shall have a minimum storage life of 3 years. Storage life critical components may be refurbished.

3.2.1.14 Telemetry. Telemetry data shall be primarily limited to those functions necessary for control and operation of the power module during flight. This telemetry shall include but not be limited to functions such as bus and battery voltages and currents, the status of bistable or multimode circuits or relays, temperatures, etc. Telemetry indications of equipment status should be as direct an indication as practicable.

3.2.1.15 Expansion capability. A capability shall be provided within the limits of module structure, size, and power available for expanding the baseline configuration to include the addition of redundant units or the addition of new components to perform additional functions.

3.2.2 Physical characteristics.

3.2.2.1 Mechanical.

3.2.2.1.1 Envelope. The module envelope shall be as shown in ICD 30.1.

3.2.2.1.2 Module volume. The module will have a volume of approximately TBD cubic feet and a maximum load carrying capability of TBD pounds of equipment. Components may be mounted to the outboard facing panel, to non-outboard surfaces and the module frame members.

The outboard facing panel may be modified with local cutouts to facilitate assembly or it may be divided into separate equipment heat sink surfaces. Internal stiffness bulkheads and/or mounting panels may be added as required. However, prior to any modifications to the module structure, the subsystem contractor shall perform a structural analysis to ensure structural integrity.

3.2.2.1.3 Module weight. The total weight of the solar array and drive module for the minimum and nominal redundancy configurations shall not exceed the listed weights:

Minimum Redundancy	Nominal Redundancy
212 lb	216 lb

3.2.2.1.4 Attach-points. The attach-points between the module and the spacecraft shall be as shown in ICD 30.1.

3.2.2.1.5 Module/structure interface connector. The module/structure interface connector shall be provided by the system contractor and shall be mounted on the inside face of the module. It is required that the connector position be maintained as specified (ICD 30.1) to ensure interchangeability of modules.

3.2.2.1.6 Equipment expansion. The components shall be arranged in the module such that the expansion capability requirements of 3.2.1.16 can be accommodated with minimum impact on the baseline configuration design.

3.2.2.2 Electrical

3.2.2.2.1 Power. Total power required for the solar array and drive module shall not exceed TBD watts. Allocation of this power is as follows:

Baseline configuration requirement: TBD watts

Redundancy and expansion capability: TBD watts

Power consumption of the solar array and drive module units shall be within the power allocations in ICD 10.2.

3.2.2.2.2 Commands. Commands for controlling the solar array and drive module operation shall be as listed in ICD 10.3.

3.2.2.2.3 Telemetry. The solar array and drive module telemetry measurements shall be as listed in ICD 10.4.

3.2.2.2.4 Signal and power distribution. The solar array and drive module harness shall provide all intramodule electrical connections in conformance to ICD 30.5.

3.2.3 Reliability. The baseline configuration solar array and drive module shall be capable of performing, as specified, for at least TBD years in orbit. This shall include all redundancy incorporated including alternate and backup modes. Demonstration of compliance with these requirements shall be through reliability analysis as called out in EOS-4.1.

3.2.4 Maintainability. The solar array and drive module shall be designed in accordance with the requirements of MIL-STD-1472, paragraph 5.9, as implemented by EOS-4.1.

3.2.5 Environmental conditions. The solar array and drive module shall be designed to withstand or shall be protected against the worst probably combination of environments as specified in SP-11 and as implemented in SP-115.

3.2.6 Transportability. Transportability requirements shall be considered in the design of the solar array and drive module such that it can be transported by all standard modes with a minimum of special packing or precautionary measures, except that transportation of the array panels shall conform to the requirements of paragraph 5.3.2.1.

3.3 Design and construction

3.3.1 Parts, materials, and processes

3.3.1.1 Selection of parts, materials, and processes. Selection of parts, materials, and processes (PMP) shall be to the requirements identified in MIL-STD-143 Group I. Such selections shall be identified as standard PMP. When PMP selection cannot be made within this requirement, the item is nonstandard and justification must be made in accordance with MIL-STD-749. Control of this action shall be effected by the EOS parts, materials, and processes control board (PMPCB), with the PMPCB approving all selections, both standard and nonstandard, in accordance with EOS-4.2.

3.3.1.2 Program authorized parts list. All selections of electronic parts shall be identified and authorized by EOS-3.3-1. This list will reflect all PMPCP electronic part selections.

3.3.1.3 Program authorized materials list. All selections of materials shall be identified and authorized by EOS-3.3-2. This list will reflect all PMPCB material selections.

3.3.1.4 Program authorized processes list. All selection of processes shall be identified and authorized by EOS-3.3-3. This list will reflect all PMPCB process decisions.

3.3.1.5 Dissimilar metals. To avoid electrolytic corrosion, dissimilar metals shall not be used in direct contact unless protection against corrosion has been provided in accordance with MIL-E-8983 and MIL-STD-454, Requirement 16.

3.3.1.6 Magnetic materials. Magnetic materials shall be used only if necessary for equipment operation. Those magnetic materials which are used shall have minimum permanent, induced, and transient magnetic fields.

3.3.1.7 Fungus-inert materials. Materials which are fungus-inert, in accordance with MIL-E-8983, and MIL-STD-454, Requirement 4, shall be used.

3.3.1.8 Flammable toxic and unstable materials. Flammable, toxic, and unstable materials shall not be used.

3.3.1.9 Finish. The surface of each component of the subsystem shall be adequately finished to prevent deterioration from exposure to the specified environments that might jeopardize fulfillment of the specified performance. The finish shall also meet the applicable bonding requirements. Thermal properties of the finishes used on the component shall be compatible with the requirements given in detail in applicable equipment specifications. The requirements for finishes identified in MIL-E-8983 shall be implemented.

3.3.1.10 Outgassing. Low outgassing polymeric materials shall be used where sensitive thermal control and other surfaces are in direct line-of-sight and where temperature differences can exist between such surfaces.

3.3.2 Electromagnetic radiation

3.3.2.1 EMC/EMI requirements. The solar array and drive module, and all internal units, equipment, and/or components comprising a part thereof, shall be designed for compliance with the radiated emission and susceptibility requirements of NASA/JSC Specification SL-E-0002 as modified or amended by EOS-3.3-4 and EOS-3.3-5. The conducted emission and susceptibility requirements of the aforementioned documents are applicable only at the module to spacecraft structure interface. EMI/EMS levels at interfaces within the module shall be controlled to the extent necessary to ensure self-compatibility of the module subsystem with a safety margin of at least six (6) dB.

3.3.2.2 Electrical bonding

3.3.2.2.1 Structural bonds. All metallic members of the basic solar array and drive module radiator panel and support structure shall be electrically bonded to each adjacent member to form an electrically continuous, equi-potential ground reference plane. The maximum allowable DC impedance across any one structural joint shall be 2.5 milliohms with an overall design goal of 10 milliohms, or less, between any two diametrically opposite points on the module. The general methods of MIL-B-5087, Class R, may be used for implementation of these requirements.

3.3.2.2.2 Equipment mounting pads. Surfaces on the module radiator panel and structure which are intended for unit, equipment or component mounting shall be free of paint, anodize, or other nonconductive finishes. The maximum DC impedance between the baseplate of any electrically active unit, equipment or component and the module radiator panel and structure shall be 2.5 milliohms.

3.3.2.2.3 Electrical connectors. All interface electrical connectors, both plug and receptacle, which form a part of the unit and cable RF shielding system within the module shall have electrically conductive body shells, free of nonconductive finishes. They shall also have provisions for terminating (grounding) the shields on the interconnecting electrical harness. The maximum DC impedance between the shield termination point and the baseplate of the parent unit shall be 2.5 milliohms.

3.3.2.2.4 Electrostatic bonds. Electrically passive components or appurtenant structures which are attached to the basic module structure through isolators shall be provided with electrostatic bonds having a DC impedance of 1.0 ohm or less. The general methods of MIL-B-5087, Class S, may be used for implementation of this requirement.

3.3.2.3 Electrical systems grounding

3.3.2.3.1 Primary DC power. Primary power from the solar array is supplied to the electric power module through the structural harness. Primary power is supplied to the solar array and drive module via main bus A and B. The single-point ground for primary power shall be provided in the electric power module. Interconnecting primary power wiring between units and/or assemblies in the solar array and drive module shall be unshielded twisted pairs.

3.3.2.3.2 Secondary DC power. Secondary DC power distribution networks shall, in general, use multiple point grounding within the solar array and drive module with returns through the module radiator panel and/or support structure. These networks shall be initially grounded adjacent to the secondary transformer winding in the power converter and at each user element.

3.3.2.3.3 Secondary AC power. Secondary AC power networks shall use single-point grounding with a two-wire, twisted shielded pair distribution. The ground point may be either at the source or load end, whichever is shown by circuit analysis to be most beneficial to compatible module operation. Structural returns shall not be used for secondary AC power.

3.3.2.3.4 Intramodule signal control circuit (high level). All high-level (≥5 volt logic, bilevel or analog) signal or control circuits which do not exit the solar array and drive module shall be multiple-point grounded at both the source and load end of each circuit branch to unit or component chassis by the shortest most direct path. Circuits sharing space on a common printed circuit board should not share common grounding traces on the board or common hardwire jumpers to chassis ground logs. Preferably, each such board should have a dedicated ground plane layer to which all components requiring ground returns can be directly connected. This ground plane should, in turn, be directly bonded to unit chassis or frame through grounding pads at each hold-down fastener.

3.3.2.3.5 Intermodule signal/control circuits (high level). All high-level signal or control circuitry which exits the solar array and drive module shall be grounded at the final driving element. Two-wire, twisted shielded pair distribution shall be used for each such circuit between the source unit and the module to spacecraft interface connector. The load elements in the external module will be DC isolated from structural grounds. Any signal or control circuitry which enters the solar array and drive module shall be DC isolated from chassis/case/structure ground by a minimum of one (1) megohm resistance. Any such circuits shall also be provided with two-wire, twisted shield pairs between the interface connector and the load unit.

3.3.2.3.6 Analog circuits (low level). Any low-level (<5 volts) analog circuits, which are shown by circuit analysis or test to be sensitive to circulating currents in the module or spacecraft structure, shall be single-point grounded either at the source or load element, whichever is most appropriate for the circuit under consideration. Wherever possible, balanced differential circuitry should be used. In the case of low-level circuits which enter or exit the solar array and drive module, the location of the circuit ground point shall be coordinated with the Systems Integration Contractor.

3.3.2.3.7 Data bus. The command and telemetry data bus system shall be differentially driven and balanced to structural ground in the CDH module. This system shall be transformer-coupled at each remote terminal. Each individual data bus wire entering the solar array and drive module shall be DC isolated from chassis/case/structure by a minimum of one (1) megohm resistance.

3.3.2.3.8 Wire shields. External shields shall be provided for all interconnecting wires between units, equipments, or components in the solar array and drive module and between each input/output connector and the module to spacecraft interface connectors except for input and output primary DC power lines. In general, these shields shall be multipoint grounded at each end and at each intermediate interface. An exception to this rule will be allowed for low-level analog circuitry where single-point shield grounding may be necessary. If possible, such circuitry should be provided with two mutually isolated shields, the inner shield being single-point grounded and the outer shield multipoint grounded.

3.3.2.3.9 EMI filter components. High performance EMI filters shall be required at the primary DC power input and return terminals of each power converter unit or subunit in the solar array and drive module in order to ensure compliance with the electromagnetic interference requirements of the applicable specifications. These filters should have two stages: (1) an AF ripple filter stage balanced line to line; and (2) a pair of RF feedthrough filters bulkhead-mounted behind the input connector. The combined AF/RF filter circuit should be designed for the minimum capacitance from either line to chassis necessary to achieve compliance with the specifications. Excessive line to ground capacitance will tend to negate the beneficial effects of the twisted pair wiring used in the primary DC power harness. Similar filtering may be required at the input and output terminals of the pulse width switching regulator assembly; however, because of fault isolation requirements, line-to-case feed-through RF filters shall not be used.

3.3.3 Nameplates and product marking. Each unit shall be identified in accordance with MIL-STD-130 as implemented by EOS-3.3-6. Where practical, a minimum character height of 0.09 inch shall be used. Marking shall include the manufacturer's name or initials, assembly part number and revision status, assembly serial number, contract number, and others as delineated in the component equipment specifications.

3.3.4 Workmanship. The module and its component equipment shall be constructed, finished, and assembled in accordance with MIL-STD-454, Requirement 9, and the specifications and drawings specified herein.

3.3.5 Interchangeability and replaceability. The solar array and drive module shall be designed to permit removal and replacement of components with a minimum of disturbance of associated or adjacent equipment. Design for service and access shall conform to the principles and requirements of MIL-STD-470.

3.3.6 Safety. The solar array and drive module shall be designed to meet or exceed the requirements of EOS-3.3-7. The design criteria shall include but are not limited to those set forth in MIL-STD-882.

3.3.7 Human performance/human engineering. MIL-STD-1472A shall be used for the design of man/machine interfaces.

3.4 Documentation. Documentation shall be prepared as defined in EOS 3.3-8.

4. QUALITY ASSURANCE PROVISIONS

4.1 General. The quality assurance program controls shall be in accordance with the requirements of MIL-Q-9858 as augmented by EOS-4.1.

4.1.1 Responsibility for inspection and test. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified herein. Except as otherwise specified, the supplier may use his own facilities or any commercial laboratory acceptable to the government. The government reserves the right to perform any of the inspections set forth in the specifications where such inspections are deemed necessary to ensure that supplies and services conform to prescribed requirements.

4.2 Quality conformance inspections

4.2.1 Category I tests

4.2.1.1 Development tests. Development tests shall be conducted to verify the module performance parameters, proper interfacing between components of the module, and proper interfacing between the module and other spacecraft modules. The development tests shall be conducted at the unit and/or integrated module levels. The type of developmental evaluation or test to be applied to verify satisfaction of the requirements defined in ths specification are outlined in Table 1. The tests shall be conducted in accordance with EOS-4.2.

4.2.1.2 Qualification tests. Module qualification shall be accomplished by means of qualification tests on the individual units and/or as a normal consequence of having the units intalled in the qualification test space vehicle during its qualification testing. The types of qualification evaluation or test to be applied to verify satisfaction of the requirements of this specification are outlined in Table 1. Qualification test verification methods and requirements shall be as defined in the EOS-4.2.

4.2.1.2.1 Components. As a mininum, the following types of component qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, are measured before and after the environmental tests.

- Random vibration. Each component will be subjected to a random vibration from 20 to 2000 Hz in all three axes while electrically powered in the maximum normal load condition. Performance will be continuously monitored to detect failures (either hard or trends).

- Thermal vacuum. Thermally cycle the component beyond expected orbital temperatures while at an orbital vacuum condition. The component is operating continuously and detailed performance parameters are measured at each temperature extreme.

- Electromagnetic interference (EMI) and susceptibility (EMS). Engineering EMI and EMS data are measured to provide diagnostic information for the module EMI/ EMS qualification.

4.2.1.2.2 Module. As a minimum, the following types of module qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, and electrical interface characteristics are measured before and after the environmental tests.

Table 1. Verification Cross Reference Index

VERIFICATION CROSS REFERENCE INDEX

LEGEND: N/A - Not Applicable, I - Inspection, A - Analysis, S - Similarity, T - Test

SECTION 3 REQUIREMENTS	Not Applicable	Acceptance			Qualification				SECTION 4 VERIFICATION REQUIREMENTS
		I	A	T	I	A	S	T	
(TBD)									

SAMPLE

- Acoustics. Acoustics testing will be performed with the module mounted in a receptacle which provides acoustics shielding similar to the Observatory structure. The module will be heavily instrumented to verify design adequacy of interconnections (electrical and mechanical) between the module and other Observatory elements. The test data will be used to confirm estimates of the vibration environment for components mounted on radiators and establish procedures for acceptance testing. The module will be electrically powered and continuously monitored via the data bus to detect failures and out-of-tolerance performance trends. The solar array will be excluded from this test.

- Thermal vacuum. The objectives of the module thermal qualification test are to determine: 1) maximum and minimum operating temperatures for the module for worst-case environments and operating duty cycles, 2) temperature levels and temperature variations of the module adjacent to the interface fittings, and 3) heater power requirements for cold-case conditions.

  The test will be conducted in a thermal-vacuum chamber with liquid nitrogen cold walls. The module will be mounted on a fixture simulating the spacecraft frame. The properties of this fixture shall permit its temperature to be held constant at any level between 0 and 100°F.

- Qualification thermal/vacuum testing. The module thermal control system design shall be verified by testing in a simulated space environment. Test conditions representing the hot and cold extremes shall be included. Representative structure assembly module attach-points shall be used to support the module and a simulated structure/module insulation barrier provided. Solar and earth heat inputs shall be represented in the test.

- Temperature extreme test. The module shall be functionally tested in a vacuum environment with heat sink temperatures ±20°F greater than the most severe predicted operating temperatures. This test can be performed in conjunction with the qualification thermal/vacuum test.

- EMI and EMS. EMI measurements will determine the levels radiated on each module electrical interface and verify there is adequate design margin to ensure non-interference with other modules. EMS measurements will verify that each module has adequate susceptibility margin to prevent performance degradation resulting from other module allowable radiation levels. The general test methods of MIL-STD-462, as modified or amended by EOS-3.3-5, shall be used during this test program except that, wherever practical, spectrum analyzers or other forms of rapid display and analysis equipment may be used in lieu of equipment requiring manual tuning and data collection.

4.2.1.2.3 Inspection sequence. The inspection sequence shall be governed by the following:

- Examination of the module shall be performed prior to functional testing.
- Functional tests shall be performed prior to, during, where appropriate, and following environmental testing. The functional testing conducted at the conclusion of an environmental test may serve as the functional testing to be performed prior to the next environmental test.
- Environmental testing may be performed in any sequence.

4.2.1.2.4 Failure criteria. The module shall exhibit no failure, malfunction or out-of-tolerance performance degradation as a result of the examinations and test specified herein. Any such failure, malfunction or out-of-tolerance performance degradation shall be cause for rejection.

4.2.2 Test conditions. Unless otherwise specified, the conditions under which all inspections are accomplished shall be specified below:

4.2.2.1 Atmospheric and environmental. All examinations and tests shall be conducted under the local prevailing pressure at the time of test, at a temperature of 75 $\pm 5^{\circ}$F and at a relative humidity of 55 percent or less.

4.2.2.2 Deviations of atmospheric conditions. When tests are performed with atmospheric conditions substantially different from the specified values, proper allowances for changes in instrument readings shall be made to compensate for the deviation from the specified conditions.

4.2.3 Equipment warmup time. The equipment warmup time shall be less than 1 minute.

4.2.4 Temperature stabilization. Temperature stabilization shall have been achieved when temperature of the equipment mounting structure varies not more than 5°F during a period of 15 minutes.

4.2.5 Measurements. Measuring instruments used to determine functional parameter values (such as voltage, frequency, current, flow, pressure, etc.) shall indicate true values with an accuracy determined by the tolerance allowed for the parameter variation itself, such that the measuring instrument shall not introduce an uncertainty greater than 10 percent of the allowable variation of the measured parameter. However, no such measurement accuracy shall be required to exceed 0.5 percent of the required value of the parameter unless otherwise specified.

Except as specifically noted in the inspection methods, tolerances for environmental test conditions shall be defined as follows:

Temperature: $\pm 5^{\circ}F$

Barometric pressure: ±10 percent

Relative humidity: +5, -10 percent

Time: ±5 percent

Vibration amplitude (sine): ±10 percent

Vibration frequency: ±2 percent

4.3 Category II tests. (Not applicable)

4.4 Analyses. The following requirements of Section 3 shall be verified by review of analytical data:

3.2.3 Reliability: To be verified by analysis in accordance with Section TBD of EOS-4.1.

3.3.7 Safety: To be verified by analysis in accordance with Section TBD of EOS-4.1.

5. PREPARATION FOR DELIVERY

5.1 General. The solar array and drive module as specified herein shall be protected from degradation by environments anticipated during shipment, handling, and storage. Standard commercial packaging practices are acceptable provided they fulfill these requirements.

5.2 Preservation and packaging

5.2.1 Cleaning. The solar array and drive module shall be clean and free of contaminants which might impair its use prior to packaging.

5.2.2 Attaching parts. When attaching parts, such as nuts, bolts, washers, etc., accompanying the solar array and drive module, they shall be preserved, bagged, appropriately identified, and attached to or adjacent to the fitting for which they are intended.

5.2.3 Electrical connectors. All electrical connectors shall be capped with protected dust caps. Caps used shall be a friction fitting or threaded type which do not require tape or a mechanical device to secure.

5.2.4 Critical surfaces. External machined surfaces and mounting surfaces of the solar array and drive module shall be protected with protive pads. Materials used for pads shall not cause item deterioration.

5.2.5 Wrapping. The solar array and drive module shall be wrapped or bagged using anti-static polyethylene film.

5.2.6 Cushioning. When required for protection, the solar array and drive module shall be cushioned using a suitable resilient foam cushioning material such as polyethylene, polyurethane, or polystyrene.

5.3 Packing. The solar array and drive module, excluding the solar array assembly, shall be packed in a suitable shipping container designed for one item only. The shipping container used shall provide protection to the item against corrosion, deterioration, and damage during shipment from the source of supply to the receiving activity. Containers shall comply with applicable tariffs and regulations for particular modes of transportation when so shipped.

The solar array shipping container shall be TBS.

5.3.1 Storage conditions. The solar array and drive module, excluding the solar array assembly, shall not be adversely affected by storage within its container at temperatures between 60°F and 90°F and relative humidities of 60 percent or less.

5.3.1.1 Storage conditions, solar array assembly. TBS.

5.3.2 Shipping conditions. The solar array and drive module shall be capable of withstanding the following environments:

Temperature:	+160°F in an unsheltered area (125°F + 35°F due to solar radiation) and -40°F in accordance with restricted air transport criteria
Humidity:	Up to 100 percent in an unsheltered area
Rough Handling:	Capable of withstanding and physically protecting the unit from rough handling during shipping by common carrier.

The shipping containers and/or constraints shall be equipped with suitable monitors to enable the detection and lagging of excessive shock and acceleration stress.

5.4 Marking for shipment. Each solar array and drive module and shipping container shall be marked with the following:

(a) Item nomenclature

(b) Systems part number

(c) Contract or purchase order number

(d) Manufacturer's name

(e) Manufacturer's part number and serial number (on item container only)

(f) Quantity

(g) Date of manufacture (on item container only)

(h) FRAGILE — HANDLE WITH CARE (when applicable)

(i) SPACE VEHICLE MATERIAL — DO NOT OPEN IN RECEIVING OR RECEIVING INSPECTION (when applicable — shipping container only)

(j) Actual weight.

5.4.1 Documentation. All required reliability and test documentation such as test reports, certifications, shipping invoices, etc., shall be either packed in the solar array and drive module container or attached to the exterior surface of the shipping container. Attachment shall be such a manner as to preclude loss of this data during handling and shipment by common carrier.

6. NOTES

6.1 Definition of spacecraft configuration.

6.1.1 Minimum redundancy configuration. The minimum redundancy configuration is defined as the spacecraft configuration which contains the minimum redundancy of units necessary to ensure that no plausible single-point failure will prevent Observatory retrieval by the Space Shuttle System. For purposes of this specification this configuration is identified as the baseline spacecraft configuration.

6.1.2 Nominal redundancy configuration. The nominal redundancy configuration is defined as the spacecraft configuration which includes standby redundant units for most of the electronic assemblies to provide a "typical" redundancy level for long-life spacecrafts.

TITLE

PRIME ITEM DEVELOPMENT SPECIFICATION
FOR THE

WIDEBAND COMMUNICATIONS MODULE

DATE 20 SEPT 1974 NO. SP-1124

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

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SPECIFICATION SP-1124

WIDEBAND COMMUNICATIONS MODULE

1. SCOPE

This document establishes the requirements for the performance, design, test, and qualification of the wideband communications module for the Earth Observatory Satellite, herein referred to as the WBC module configured item number TBD. The WBC module provides the communications capability required to transmit satellite experiment wideband data to ground-based users.

2. APPLICABLE DOCUMENTS

2.1 Government documents. The following documents of the exact issue shown by revision or data form a part of this specification to the extent specified herein. If no revision or date is shown, the latest released issue of the applicable document shall apply.

STANDARDS

NASA

NASA/JSC	Electromagnetic Compatibility Control Plan
NASA STDN 101.1	STDN User's Guide
———	Electromagnetic Compatibility Requirements for Space Systems

Military

MIL-STD-143B	Standard and Specifications, Order of Precedence and Selection
MIL-STD-454C	Standard General Requirement for Electronic Equipment
MIL-STD-749B	Preparation and Submissions of Data for Approval of Nonstandard Electronic Parts
MIL-STD-1472A	Human Engineering Design Criteria for Military Systems, Equipment, and Facilities
MIL-STD-882 15 Jul 1969	System Safety Program for Systems and Associated Subsystems and Equipment, Requirements for

MIL-STD-462	EMI/EMS Test Methods
MIL-STD-130D	Identification Marking of U.S. Military Property

SPECIFICATIONS

NASA

SP-11	Observatory Segment Specification
SP-115	Observatory Environmental Criteria Specification
SP-1112	Communication and Data Handling Module Specification
SP-1113	Power Module Specification

Military

MIL-E-8983A	Electronic Equipment, Aerospace Extended Space Environment
MIL-Q-9858A	Quality Program Requirements

EOS Program Documents

EOS-3.3-1	Program Authorized Parts List
EOS-3.3-2	Program Authorized Materials List
EOS-3.3-3	Program Authorized Processes List
EOS-3.3-4	Electromagnetic Compatibility Control Plan
EOS-3.3-5	EMI/EMS Limits and Test Methods
EOS-3.3-6	Marking of Parts and Assemblies
EOS-3.3-7	Safety Design Criteria
EOS-3.3-8	Configuration Management Plan
EOS-4.1	System Effectiveness Program Plan
EOS-4.2	Integrated Test Plan

3. REQUIREMENTS

3.1 General

3.1.1 Function. The wideband communication module contains the digital and RF equipment required to telemeter selected thematic mapper (TM) data, at 20 Mbit/sec to low-cost ground stations (LCGS) and full frame TM and high resolution pointing imager (HRPI) data, at 256 Mbit/sec to the wideband data collection stations. The 20 Mbit/sec selected data is transmitted in a PCM, biphase-PSK format on an X-band carrier in the vicinity of 8 GHz. The 256 Mbit/sec data is transmitted in a quadriphase-PSK format, also in the same frequency region.

3.1.2 Operation. The WBC module (Figure 1) will be composed of the digital and RF equipment required to process and transmit selected TM data to LCGS and full frame TM and HRPI data to the Central Data Processing Facility (CDPF) via selected NASA STDN stations.

The wideband communications system is the major component of the wideband communications module. This system functions to modulate, amplify, and transmit the data received from the LCGS speed buffer and the HRPI and TM multiplexers (these units make up the major components of the data handling module). Two X-band links relay the sensor data to NASA STDN stations and to any selected one of a set of low-cost ground stations. The band allotted to these links is 8.025 to 8.400 GHz (total 395 MHz bandwidth).

The data rates of the TM processor and the HRPI multiplexer are 128 Mbits/sec. The LCGS speed buffer selects a subset of TM data which is fed into the LCGS modulator at 20 Mbits/sec. The combined TM and HRPI multiplexer outputs (each at 128 Mbits/sec) are converted to quadriphase modulation at 256 Mbits/sec.

3.1.3 Government-furnished module equipment. Certain items of module equipment which are identical or nearly identical in function will be fabricated by the integration contractor and provided as GFE to the remaining module contractors. This equipment shall include:

Item	
Data interface unit	3.2.1.3.4 of SP-1112
Power control unit	3.2.1.9 of SP-1113

All module structures and integration of the modules will be provided by the integrating contractor.

3.1.4 Equipment list. The equipment contained within the WBC module shall be as follows:

Component	Quantity Per Module
RF Equipment Group	
X-band antennas	2
Gimbal biaxial drive assembly	2
Gimbal drive electronics	1
Biphase modulator	1
Quadriphase modulator	1
X-band power amplifier	2

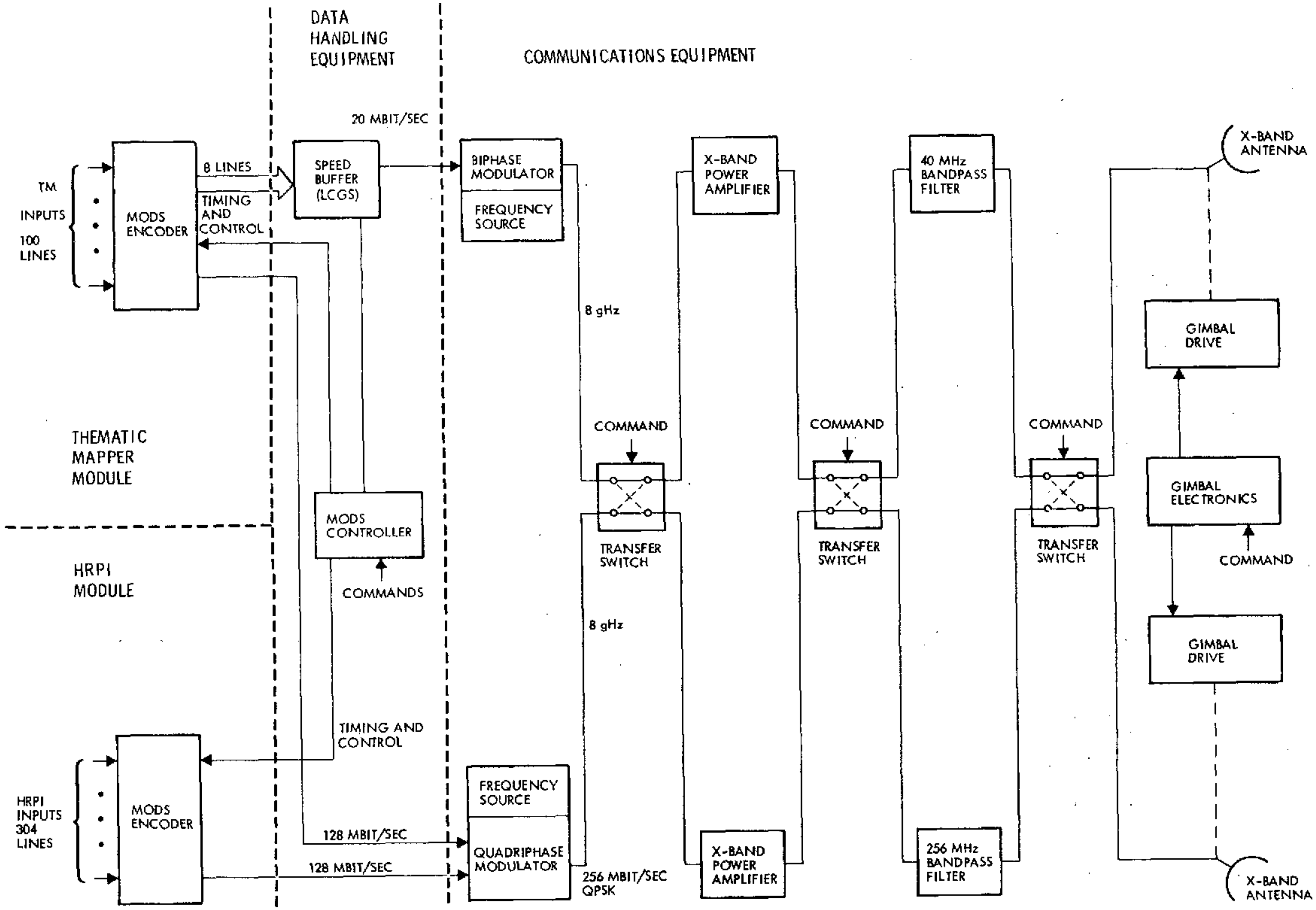

Figure 1. Typical Wideband Communications Module Block Diagram

Component	Quantity Per Module
Narrowband pretransmission filter	1
Wideband pretransmission filter	1
RF transfer switches	3
Flexible waveguide	1
Data Handling Equipment Group	
LCGS speed buffer	1
Multi-megabit operational data systems (MODS) controller	1
Data interface unit	1
Power Conditioning	
Power conditioning unit	1

3.2 Characteristics

3.2.1 Performance

3.2.1.1 Electrical interface parameters. The module electrical interface lines are illustrated in Figure 2.

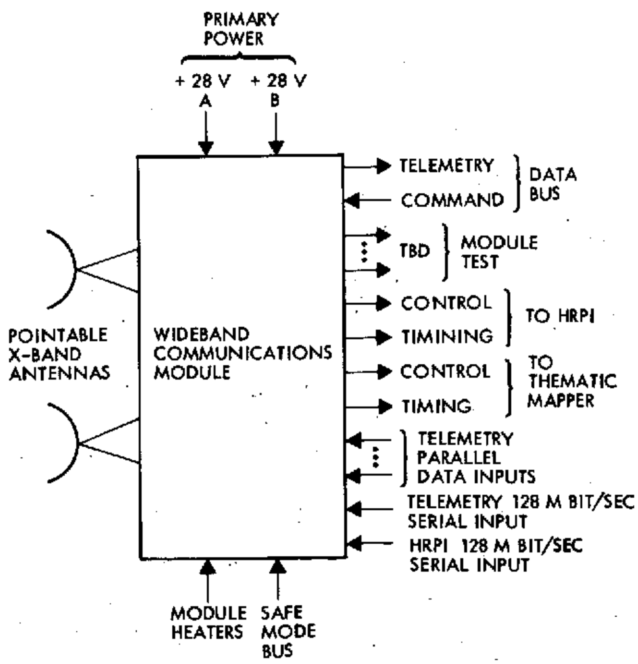

Figure 2

Wideband Communications Module — Electrical Interfaces

3.2.1.1.1 <u>RF interface</u>. Two RF LINKS shall be provided between the spacecraft and the ground-based stations. One link shall transmit data at 256 Mbit/sec in quadriphase form to the NASA Central Data Processing Facility via STDN while the other shall transmit data at 20 Mbit/sec in biphase form to the LCGS. Both RF carriers shall lie in the X-band frequency region.

3.2.1.1.1.1 <u>256 Mbit/sec wideband RF link</u>. The wideband (256 Mbit/sec) link to the CDPF shall be configured with the following parameters:

Transmit carrier range:	8.025 to 8.400 GHz
Transmit frequency:	TBD GHz
RF signal polarization:	RHCP
Transmit EIRP:	25.0 dBW
Wideband data rate:	256 Mbit/sec
Carrier modulation:	Quadriphase/PSK

3.2.1.1.1.2 <u>20 Mbit/sec RF link</u>. The 20 Mbit/sec link to the LCGS shall be configured with the following parameters:

Transmit carrier range:	8.025 to 8.400 GHz
Transmit frequency:	TBD GHz
RF signal polarization:	RHCP
Transmit EIRP:	25.0 dBW
Data rate:	20 Mbit/sec
Carrier modulation:	Biphase/PSK

3.2.1.1.2 <u>Data handling interface</u>. The WBC module shall accept thematic mapper data provided in two forms. The first shall be parallel input data provided at 128 Mbit/sec on eight parallel lines at 16 Mbit/sec per line. This information shall be processed within the LCGS speed buffer unit contained in the module to derive lower rate TM data to be transmitted at 20 Mbit/sec. The second data input provided to the WBC module from the TM shall be in serial form at 128 Mbit/sec. This data shall be combined with data provided from the HRPI to make up the wideband data transmitted at 256 Mbit/sec. The HRPI shall provide a serial, 128 Mbit/sec data input to the WBC module. The HRPI data and the serial TM data shall quadriphase modulate the downlink X-band carrier at a 256 Mbit/sec rate.

The WBC module shall contain a central data controller and timing unit which will provide control and timing signals to the TM and HRPI multi-megabit operational data system (MODS) encoders.

3.2.1.1.2.1 TM data interface. The WBC module shall be configured with the following TM data interface parameters:

(a) LCGS Speed Buffer Input:

(1) Data rate: 128 Mbit/sec parallel on 8 lines (16 Mbit/sec/line).

(2) Form: parallel (on eight lines)

(3) Level: TBD

(b) Serial Input:

(1) Data rate: 16 Mbit/sec on each of 8 lines

(2) Form: Serial data on 8 parallel links

(3) Format: See Figure 3.

(4) Level: TBD

(c) LCGS Speed Buffer Output. Selected TM data to be transmitted to the LCGS shall be formatted as follows:

(1) One spectral band, full swath

(2) Two spectral bands, one-half swath

(3) Four spectral bands, one-quarter swath

(4) Seven spectral bands, 90 meter resolution

(5) Output data rate: 20 Mbit/sec

(6) Data Reformatting: see Figure 4

3.2.1.1.2.2 HRPI data interface. The WBC module shall be configured with the following data interface parameters:

Data Rate: 128 Mbit/sec

Form: Serial

Format: See Figure 5

Level: TBD

3.2.1.1.2.3 MODS controller interface.

Major frame timing: 13 bits (16, 384)

Minor frame timing: 7 bits (128)

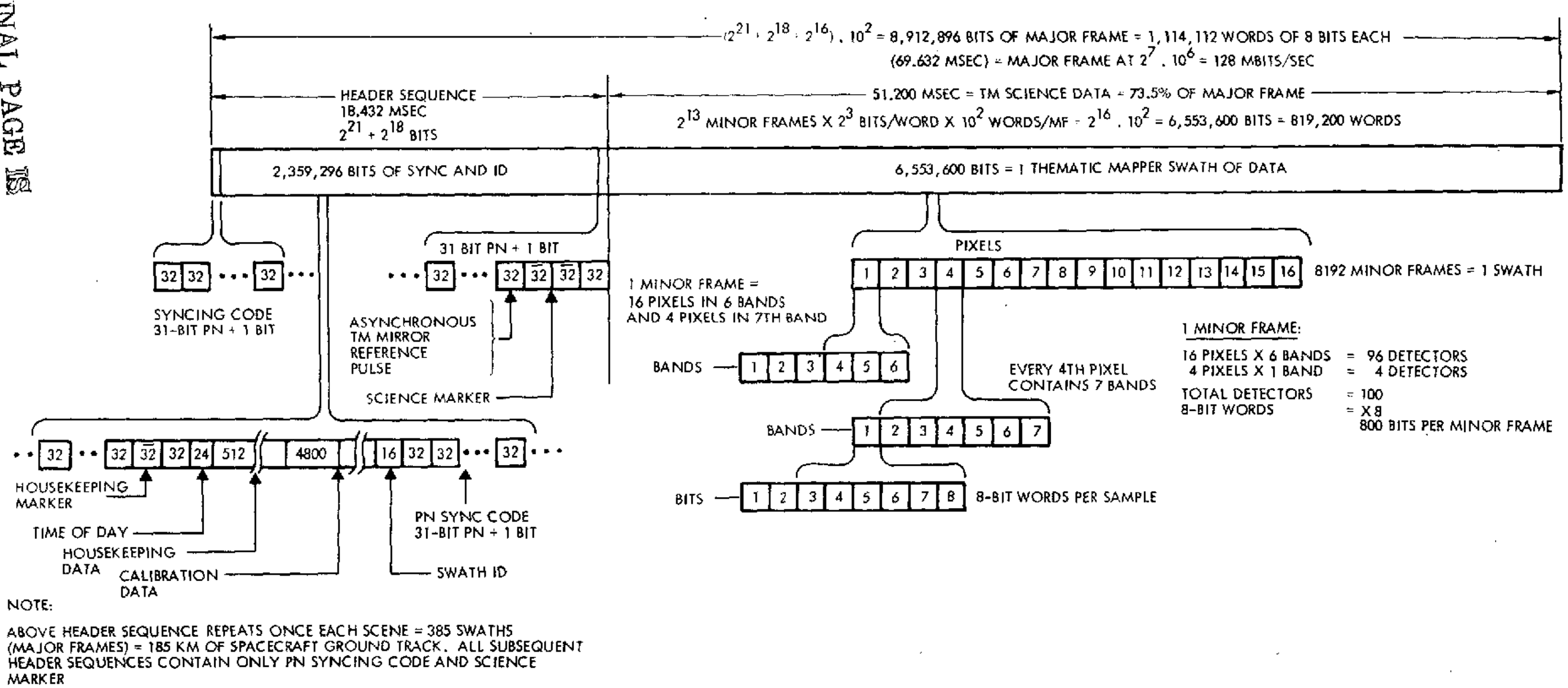

Figure 3. Wideband Data System CDPF Data Format – Thematic Mapper

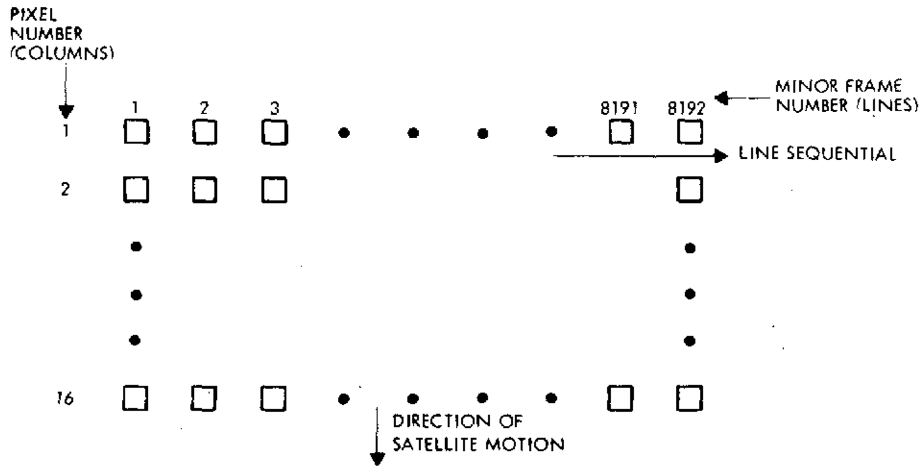

Figure 4. Speed Buffer Data Format (One Swath)

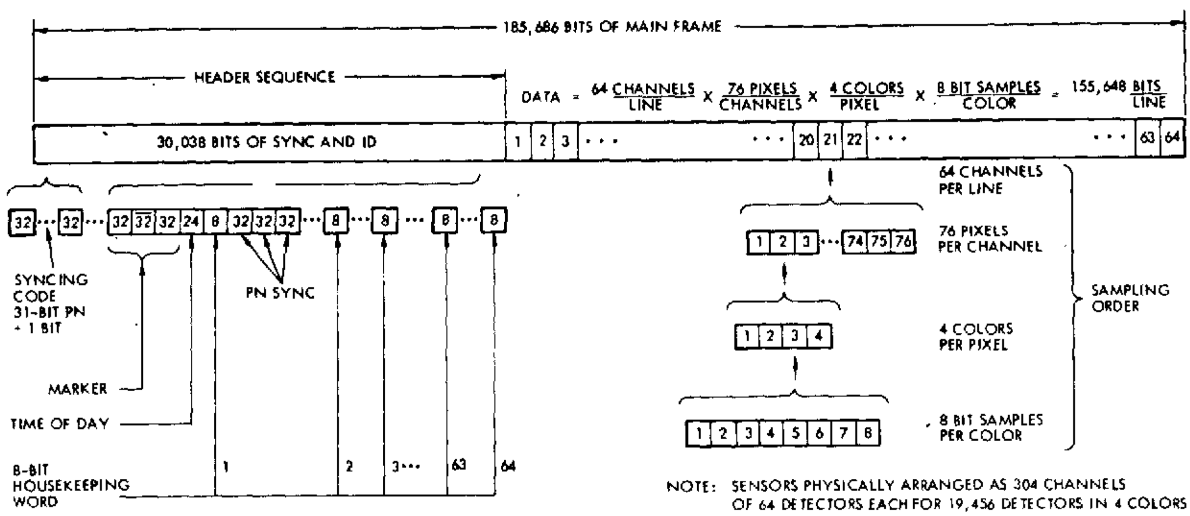

Figure 5. Wideband Data System CDPF — HRPI

Word timing:	Selectable word lengths of 4, 8, 16, and 32 bits
Long-term stability:	Less than 1 percent frequency change for the life of the mission
Medium-term stability:	Less than 1 x 10^{-9} frequency change averaged over any 30-sec period
Short-term stability:	Less than 6 nsec peak jitter at each sample. Samples occur every 62.5 nsec.

3.2.1.1.3 Primary power. The WBC module shall receive power on redundant lines as follows:

Voltage: +28V ±7 (including ripple)

Current: 3 amp maximum

3.2.1.1.4 Module heaters. A connection to heater power shall be provided. The line shall operate at +28 volts nominal with TBD amperes maximum current.

3.2.1.1.5 Safe mode bus. A connection to the safe mode bus shall be provided. Normal mode operation shall be initiated by a +5 volt level (10 ma maximum module sink current). Safe mode operation shall be initiated by a 0 volt level.

3.2.1.1.6 Launch vehicle shuttle umbilical. Buffered lines shall be provided for the umbilical as follows:

(a) LCGS speed buffer output

(b) 128 Mbit/sec TM data input to quadriphase modulator

(c) 128 Mbit/sec HRPI data input to quadriphase modulator

3.2.1.1.7 Module test connector. A test connector shall be provided for module test of the following signals: TBD.

3.2.1.2 RF equipment group. The RF equipment group (Figure 1) shall contain the following equipment:

(a) X-band antennas

(b) Gimbal biaxial drive assembly

(c) Gimbal drive electronics

(d) Biphase modulator

(e) Quadriphase modulator

(f) X-band power amplifier

(g) Narrowband pretransmission filter

(h) Wideband pretransmission filter

(i) RF transfer switches

(j) Flexible waveguide

3.2.1.2.1 X-band antennas. Two identical X-band high gain antennas shall be employed to transmit both high and low data rate RF signals to the LCGS and CDPF/STDN. Each antenna shall be mounted on its own two-axis pedestal for accurate pointing.

3.2.1.2.1.1 Functional characteristics. Each high gain antenna shall operate over the 8.0 to 8.4 GHz frequency range. The antennas shall be parabolic reflector excited by a focal point feed using crossed dipoles. The effective diameter of the antennas shall be 2.0 feet.

3.2.1.2.1.1.1 Peak gain. The high gain antennas shall have a peak gain of 31.5 ±0.5 dB.

3.2.1.2.1.1.2 Half power beamwidth. The high gain antennas shall have a half power beamwidth of 4.3 ±0.2 degrees.

3.2.1.2.1.1.3 Polarization. The high gain antenna shall be right-hand circularly polarized. The rotating electric field vector shall be clockwise for an observer looking in the direction of propagation.

3.2.1.2.1.1.4 Axial ratio. The axial ratio of the high gain antenna shall not exceed 0.5 dB over the half power beamwidth.

3.2.1.2.1.1.5 VSWR. The input impedance of the high gain antenna shall present a voltage standing wave ratio (VSWR) of 1.5 to 1.0 or less when referenced to 50 ohms resistive over the specified frequency band.

3.2.1.2.1.2 Physical characteristics

3.2.1.2.1.2.1 Dimensions. The maximum diameter of the high gain antenna shall be 26 inches. The maximum height of the antenna shall be 14 inches.

3.2.1.2.1.2.2 Weight. The weight of the high gain antenna shall not exceed 2.2 pounds.

3.2.1.2.2 Antenna gimbal biaxial drive assembly. Two, identical, high gain antenna gimbal biaxial drive assemblies (BDA) shall be provided as part of the WBC module equipment for the purpose of mounting and pointing the antennas.

The BDA shall consist of two stepper motors and two gear reducers, position transducers, latches and angular stops. One motor and gear reducer shall produce rotation about one axis and the other about an orthogonal axis. This component, along with the appropriate bearings and electrical connectors, will be combined in a single unit which will support a load and position it in azimuth and elevation relative to the spacecraft body upon which it is mounted. Each axis drive shall be interchangeable with the other axis drive.

3.2.1.2.2.1 Functional characteristics

3.2.1.2.2.1.1 Direction of rotation. The BDA shall be capable of rotating the load either clockwise or counterclockwise in both the elevation and azimuth axes.

3.2.1.2.2.1.2 Angular travel

(a) Total travel. The BDA shall have a minimum angular excursion capability of ±100 deg in each axis.

(b) Nominal travel. The BDA shall be capable of moving the load ±10 deg minimum in each axis about the null position. This is the normal operating range of the drive.

(c) Stops. The BDA shall have at least one hard stop in each axis located at a known position such that excursion to the stops at the extreme of the total travel shall be calibratable to within ±0.063 deg. Frequency of stop contact shall be in accordance with paragraph 3.1.2.3.4.

(d) Latching. The BDA shall have mechanical latching mechanisms in each axis such that motion is permitted past the latch when traveling from the extreme position toward the null position but motion is prohibited past the latch when traveling the opposite direction. Hand operated latch releases shall be provided for test purposes to allow motion past the latches when traveling from the null position toward the extreme positions.

3.2.1.2.2.1.3 Speed

(a) Nominal. At a nominal pulse rate of 1.0 pulse/sec, the BDA shall be capable of rotating the load at a minimum speed of 0.03 deg/sec.

(b) Slow speed. At a slow pulse rate of 4.0 pulses/sec, the BDA shall be capable of rotating the load at a minimum speed of 0.12 deg/sec.

3.2.1.2.2.1.4 Stopping capability. The BDA shall be capable of sustaining being driven against a hard stop in both axes.

3.2.1.2.2.1.5 Alignment. The alignment of the azimuth axis, elevation axis, spacecraft mounting flange, and load mounting flanges shall be as follows:

(a) Spacecraft flange-azimuth. The azimuth axis shall be perpendicular to the spacecraft mounting flange within 0.10 deg. For a 200 in-lb moment, the maximum movement from null in this axis shall be 0.05 deg.

(c) Elevation-load flange. The elevation axis shall be perpendicular to the load mounting flange within 0.10 deg. For a 200 in-lb moment, the maximum movement from null in this axis shall be 0.05 deg.

3.2.1.2.2.1.6 Backlash. The backlash in each axis shall be less than 0.0167 deg (one minute of arc) as measured at each output. This backlash does not include compliance of the drive train.

3.2.1.2.2.1.7 Step size. The nominal step size shall be 0.0315 deg. The cumulative step error shall not exceed 0.035 deg over 1 deg of travel and 0.05 deg over the 20 deg nominal travel including backlash. (See Figure 6.)

3.2.1.2.2.1.8 Position indication. The BDA shall incorporate a 36-speed resolver in each axis to provide angular position information.

3.2.1.2.2.1.9 Temperature sensing. One thermistor on each axis drive with a range of 0 to 150°F shall be provided adjacent to the motor for temperature telemetry.

3.2.1.2.2.1.10 Electrical isolation. The insulation resistance between electrically isolated circuits, and between electrically isolated circuits and the housing, shall be greater than 20 Mohms at 50 VDC.

3.2.1.2.2.1.11 Power. The peak power consumption shall be 28 watts per motor at 33 VDC. DC power for the BDA shall be provided from the central power converter unit contained within the WBC module.

3.2.1.2.2.2 Physical characteristics

3.2.1.2.2.2.1 Weight. The maximum weight of each BDA shall be 15.2 pounds.

3.2.1.2.2.2.2 Dimensions. The maximum envelope dimensions of each BDA shall not exceed 6.4 x 7.1 x 13.6 inches.

3.2.1.2.3 Antenna gimbal drive electronics. An antenna gimbal drive assembly (GDA) shall be provided as part of the WBC module equipment for the purpose of providing the drive signals required to operate the biaxial drive assembly BDA specified in paragraph 3.2.1.2.2.

The GDA shall receive primary and secondary power from the central power converter unit and inputs from the data interface unit contained within the WBC module. The assembly shall also provide telemetry conditioning for four angular position transducers on the BDA's.

3.2.1.2.3.1 Functional characterististics

3.2.1.2.3.1.1 Electrical inputs

Signal Inputs. The assembly shall accept the following signal inputs.

(a) X36 sine resolver signals. There shall be four X36 sine resolver input signals designated: -Y azimuth, -Y elevation, +Y azimuth, +Y elevation.

(b) X36 cosine resolver signals. There shall be four X36 cosine resolver input signals designated: -Y azimuth, -Y elevation, +Y azimuth, +Y elevation.

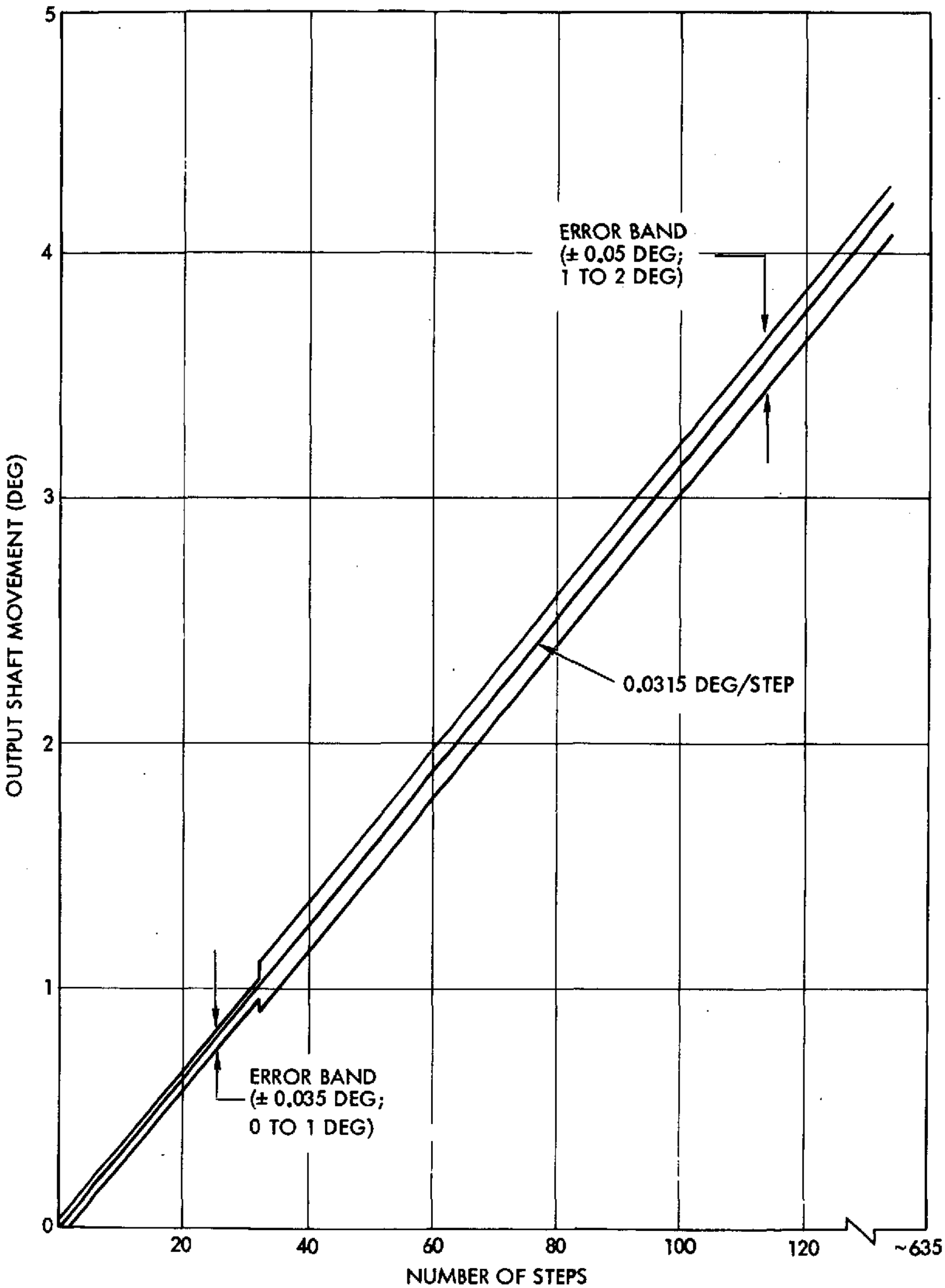

Figure 6. Maximum Cumulative Error Band

(c) X1 sine resolver signals. There shall be four X1 sine resolver input signals designated: -Y azimuth, -Y elevation, +Y azimuth, +Y elevation.

(d) RE clock 1. There shall be one digital reference clock signal designated RE clock 1.

(e) RE clock 2. There shall be one digital reference clock signal designated RE clock 2.

(f) Gimbal address. There shall be a two-line motor address selecting the gimbal motor to be stepped. The identification shall be coded as follows:

Antenna	Axis	Address Line A	Address Line B
-Y	Elevation	0	1
-Y	Azimuth	0	0
+Y	Elevation	1	1
+Y	Azimuth	1	0

(g) Gimbal direction. There shall be a sign bit identifying the direction of motor rotation. The direction of motor rotations as viewed from the shaft end shall be determined as follows:

Antenna	Axis	Sign "1"	Sign "0"
-Y	Elevation	CW	CCW
-Y	Azimuth	CCW	CW
+Y	Elevation	CCW	CW
+Y	Azimuth	CCW	CW

(h) Gimbal step pulse. There shall be a step pulse for each motor step of the addressed motor.

(i) Command inputs. The assembly shall accept the following command inputs:

Resolver advance. The receipt of this command shall cause the assembly to process the next successive set of resolver signal inputs (X36 sine, X36 cosine and X1 sine) with reference to the resolver outputs being processed at the time the command is received. Resolver selection shall be advanced in the following order: -Y azimuth, -Y elevation, +Y azimuth, +Y elevation, -Y azimuth, etc.

(j) Power inputs. The assembly shall accept the following power inputs: +5 VDC, -15 VDC and primary spacecraft power for controlling the biaxial drive assemblies and +15 VDC and -15 VDC for operating the telemetry conditioning circuitry.

3.2.1.2.3.1.2 Signal outputs

(a) -Y antenna azimuth motor drive. The assembly shall control the -Y azimuth motor upon receipt of a "00" gimbal address. The motor shall be commanded to stop by providing a current sink path in the following sequence for a logical "0" gimbal direction command.

	Winding			
Step	A	B	C	D
1	0	+	+	0
2	0	+	0	+
3	+	0	0	+
4	+	0	+	0
5	0	0	+	0

Note: + indicates deenergized coil
0 indicates energized coil

The assembly shall sequence the motor coils in the reverse of the above sequence for a logical "1" gimbal direction command. One step in the above sequence shall be commanded for each step pulse input. The motor coils shall be energized for 62 ±12 msec and the assembly shall be capable of sinking 840 milliamp during the time the motor is energized. The assembly shall be capable of stepping through the above sequence at a maximum rate of eight steps per second.

(b) +Y antenna azimuth motor drive. The assembly shall control the +Y antenna azimuth motor upon receipt of a "10" gimbal address. The control characteristics shall be identical to those specified in paragraph 3.1.1.2.1.1.

(c) +Y antenna elevation motor drive. The assembly shall control the +Y antenna elevation motor upon receipt of a "11" gimbal address. The control characteristics shall be identical to those specified in paragraph 3.1.1.2.1.1.

(d) -Y antenna elevation motor drive. The assembly shall control the -Y antenna elevation motor upon receipt of a "01" gimbal address. The motor shall be commanded to stop by providing a current sink path for the windings in the sequence specified in paragraph 3.1.1.2.1.1 for a logical "1" gimbal direction command. The assembly shall sequence the motor coils in the reverse of the specified sequence for a logical "0" gimbal direction command.

3.2.1.2.2.3.1.3 Power outputs. The unit shall provide the following power outputs.

(a) Resolver excitation. There shall be four identical resolver excitation outputs designated -Y azimuth, -Y elevation, +Y azimuth, +Y elevation. These outputs shall be sinusoidal with an amplitude of 8 ±5 percent volts peak and a frequency of 1.027 ±1 percent kHz, and a maximum harmonic content of:

0.6%	Third harmonic
0.5%	Fifth harmonic
0.4%	Seventh harmonic
0.35%	Ninth harmonic

(b) Plus 28 VDC. There shall be four identical +28 VDC outputs designated -Y azimuth, -Y elevation, +Y azimuth, +Y elevation. The voltage range of these outputs shall be +19 to +33 VDC. Current limiting at 3 amperes shall be provided for all four outputs.

3.2.1.2.3.1.4 Power

(a) Inputs. The assembly shall accept primary spacecraft power and switched secondary power inputs of +5 and -15 VDC as exclusive "OR" sets for functional operation of the control channels. The assembly shall also accept a switched input power set of +15 and -15 VDC for on/off control of the telemetry conditioning circuitry. The power requirements are as follows.

(1) Primary spacecraft power. The voltage shall be between 28 ±7 VDC. The current consumption from this voltage shall be 1 ±0.3 amp for 62 ±12 msec at a commanded maximum repetition rate of 8 pulses per second. The current consumption from this voltage shall not exceed 20 micro-amp when no input stop command is present. The assembly shall tolerate the application of up to 35 volts.

(2) +5 VDC (control channel 1). The nominal voltage shall be +5 VDC. The composite deviation of voltage shall be 3 percent or less. The current consumption from this voltage shall be 90 ±20 milliamp.

(3) +5 VDC (control channel 2). The characteristics are identical to those specified in paragraph 3.2.1.4.1.2.

(4) -15 VDC (control channel 1). The nominal voltage shall be -15 VDC. The composite deviation of voltage shall be 3 percent or less. The current consumption from this voltage shall be 1.25 ±0.75 milliamp.

(5) -15 VDC (control channel 2). The characteristics are identical to those specified in paragraph 3.2.1.4.1.4.

(6) +15 VDC (resolver electronics (on/off). The nominal voltage shall be +15 VDC. The composite deviation of voltage shall be 3 percent or less. The current consumption from this voltage shall be 113 ±26 milliamp.

(7) -15 VDC (resolver electronics on/off). The nominal voltage shall be -15 VDC. The composite deviation of voltage shall be 3 percent or less. The current consumption from this voltage shall be 66 ±11 milliamp.

(b) Outputs

(1) Resolver excitation. The resolver excitation signal shall be sinusoidal with an amplitude of 8 ±5 percent volts peak and a frequency

of 1.027 ±1 percent kHz. The source impedance shall be less than 50 ohms at 1.027 kHz. The load shall be 0.2 watt maximum at a power factor of 0.55 ±0.05.

(2) +28 VDC. All +28 VDC outputs shall have the same characteristics as those specified in paragraph 3.2.1.4.1.1 (plus 28 VDC input) with the exception that the current consumption from any one of these outputs shall not exceed 840 milliamp for 62 ±12 msec at a maximum repetition rate of 8 pulses per second. Only one of these outputs shall supply current at any given time.

3.2.1.2.3.2 Physical characteristics

3.2.1.2.3.2.1 Weight. The maximum weight of the GDA shall be 8.0 pounds.

3.2.1.2.3.2.2 Dimensions. The maximum envelope dimension of the GDA shall be 9.6 x 5.0 x 7.7 inches.

3.2.1.2.4 Biphase modulator. An X-band biphase modulator and frequency source shall be employed to provide the 20 Mbit/sec PSK wide-band communication signal.

3.2.1.2.4.1 Functional characteristics — biphase modulator

3.2.1.2.4.1.1 Frequency. The biphase modulator shall operate at a center frequency of 8.050 GHz.

3.2.1.2.4.1.2 Data rate. The biphase modulator shall operate at 20 Mbit/sec data rate.

3.2.1.2.4.1.3 Data format. The biphase modulator shall operate with NRZ data.

3.2.1.2.4.1.4 Carrier input power. The biphase modulator shall be driven by the X-band carrier reference signal at a power level of +8 dBm.

3.2.1.2.4.1.5 Phase orthogonality. The phase balance at the output of the biphase modulator shall be less than or equal to ±3.0 degrees.

3.2.1.2.4.1.6 Amplitude imbalance. The amplitude imbalance at the output of the biphase modulator shall be less than or equal to ±0.3 dB.

3.2.1.2.4.1.7 Switching speed. The switching speed of the biphase modulator output signal phase shall be less than or equal to 500 pulses per second.

3.2.1.2.4.1.8 Bandwidth. The (1 dB) bandwidth of the biphase modulator shall be ±300 MHz.

3.2.1.2.4.1.9 Phase linearity. The phase linearity of the biphase modulator shall be less than or equal to 4 degrees peak to peak over a ±150 MHz frequency range.

3.2.1.2.4.1.10 AM/PM conversion. The AM/PM conversion for the biphase modulator shall be less than or equal to 1.0 deg/dB.

3.2.1.2.4.2 Functional characteristics — frequency source

3.2.1.2.4.2.1 Frequency. The frequency source shall provide a continuous wave output signal at 8.050 GHz.

3.2.1.2.4.2.2 Frequency stability. The frequency stability of the frequency source shall be ±3 parts per million over the operating temperature range (not including aging).

3.2.1.2.4.2.3 Aging. The frequency drift due to aging over a 5-year lifetime shall not exceed ±25 parts per million.

3.2.1.2.4.2.4 Phase noise. The phase noise of the frequency source shall not exceed 0.30 deg RMS as measured in a phase lock loop with an 8 kHz two-sided noise bandwidth.

3.2.1.2.4.2.5 Spurious outputs. The spurious responses within a ±300 MHz bandwidth of the output frequency shall be equal to or less than -75 dBc.

3.2.1.2.4.3 Physical characteristics

3.2.1.2.4.3.1 Dimensions. The maximum envelope dimensions of the biphase modulator and frequency source shall be 6.0 x 6.0 x 1.5 inches.

3.2.1.2.4.3.2 Weight. The weight of the biphase modulator and frequency source shall not exceed 2.4 pounds.

3.2.1.2.4.3.3 DC power. The DC power input to the biphase modulator and frequency source shall not exceed 4.7 watts and shall be provided from a central power converter unit contained within the WBC module.

3.2.1.2.4.3.4 Operating temperature. The biphase modulator and frequency source shall operate over a temperature range of 0 to 120°F.

3.2.1.2.5 Quadriphase modulator. An X-band quadriphase modulator and frequency source shall be employed to provide the 256 Mbit/sec QPSK wideband communication signal.

3.2.1.2.5.1 Functional characteristics — quadriphase modulator

3.2.1.2.5.1.1 Frequency. The quadriphase modulator shall operate at a center frequency of 8.250 GHz.

3.2.1.2.5.1.2 Data rate. The quadriphase modulator shall operate at a 256 Mbit/sec data rate.

3.2.1.2.5.1.3 Data format. The quadriphase modulator shall operate with NRZ data.

3.2.1.2.5.1.4 Carrier input power. The quadriphase modulator shall be driven by the X-band carrier reference signal at a power level of +8 dBm.

3.2.1.2.5.1.5 Phase orthogonality. The phase balance at the output of the quadriphase modulator shall be less than or equal to ±3.0 degrees.

3.2.1.2.5.1.6 Amplitude imbalance. The amplitude imbalance at the output of the quadriphase modulator shall be less than or equal to ±0.3 dB.

3.2.1.2.5.1.7 Switching speed. The switching speed of the quadriphase modulator output signal phase shall be less than or equal to 500 pulses per second.

3.2.1.2.5.1.8 Bandwidth. The (1 dB) bandwidth of the quadriphase modulator shall be ±300 MHz.

3.2.1.2.5.1.9 Phase linearity. The phase linearity of the quadriphase modulator shall be less than or equal to 4 degrees peak-peak over a ±150 MHz frequency range.

3.2.1.2.5.1.10 AM/PM conversion. The AM/PM conversion for the quadriphase modulator shall be less than or equal to 1.0 deg/dB.

3.2.1.2.5.2 Functional characteristics — frequency source

3.2.1.2.5.2.1 Frequency. The frequency source shall provide a continuous wave output signal at 8.250 GHz.

3.2.1.2.5.2.2 Frequency stability. The frequency stability of the frequency source shall be ±3 parts per million over the operating temperature range (not including aging).

3.2.1.2.5.2.3 Aging. The frequency drift due to aging over a 5-year lifetime shall not exceed ±25 parts per million.

3.2.1.2.5.2.4 Phase noise. The phase noise of the frequency source shall not exceed 0.30 deg RMS as measured in a phase-lock loop with an 8 kHz two-sided noise bandwidth.

3.2.1.2.5.2.5 Spurious outputs. The spurious response within a ±300 MHz bandwidth of the output frequency shall be equal to or less than -75 dBc.

3.2.1.2.5.3 Physical characteristics

3.2.1.2.5.3.1 Dimensions. The maximum envelope dimensions of the quadriphase modulator and frequency source shall be 6.0 x 6.0 x 1.5 inches.

3.2.1.2.5.3.2 Weight. The weight of the quadriphase modulator and frequency source shall not exceed 2.4 pounds.

3.2.1.2.5.3.3 DC power. The DC power input to the quadriphase modulator and frequency source shall not exceed 4.7 watts and shall be provided from a central power converter unit contained within the WBC module.

3.2.1.2.5.3.4 Operating temperature. The quadriphase modulator and frequency source shall operate over a temperature range of 0 to 120°F.

3.2.1.2.6 X-band power amplifier. Two X-band solid-state amplifiers shall be provided to drive the wideband communications high gain antennas. Both amplifiers shall be capable of operating with either the 20 Mbit/sec biphase modulated signal or the 256 Mbit/sec quadriphase modulated signal.

3.2.1.2.6.1 Functional characteristics

3.2.1.2.6.1.1 Output power. The power amplifier shall provide a nominal output power level of -27 dBm, ±1 dB.

3.2.1.2.6.1.2 Frequency. The power amplifier shall operate at a center frequency of 8.225 GHz.

3.2.1.2.6.1.3 Bandwidth. The power amplifier 1 dB bandwidth shall be 450 MHz.

3.2.1.2.6.1.4 Output impedance. The power amplifier output impedance shall be 50 ohms.

3.2.1.2.6.1.5 Input impedance. The power amplifier input impedance shall be 50 ohms.

3.2.1.2.6.1.6 Output load VSWR. The maximum VSWR presented to the power amplifier by the output load shall be 1.6:1.

3.2.1.2.6.1.7 Gain. The power amplifier shall have a nominal gain of 27 dB.

3.2.1.2.6.1.8 Phase linearity. The phase linearity of the power amplifier shall be less than or equal to 3 degrees peak to peak over a ±225 MHz frequency range.

3.2.1.2.6.2 Physical characteristics

3.2.1.2.6.2.1 Dimensions. The maximum envelope dimensions of the X-band power amplifier shall be 7.0 x 6.0 x 2.0 inches.

3.2.1.2.6.2.2 DC power. The DC power input to the power amplifier shall not exceed 12 watts and shall be provided from a central power converter unit contained within the WBC module.

3.2.1.2.6.2.3 Weight. The weight of the power amplifier shall not exceed 2 pounds.

3.2.1.2.6.2.4 Operating temperature. The power amplifier shall operate over a temperature range of 0 to 120°F.

3.2.1.2.7 Narrowband pretransmission filter. An X-band pretransmission filter shall be employed to restrict the power spectrum of the 20 Mbit/sec biphase modulated signal before it is applied to the high gain antenna.

3.2.1.2.7.1 Functional characteristics

3.2.1.2.7.1.1 Frequency. The filter shall operate at a center frequency of 8.045 GHz.

3.2.1.2.7.1.2 Bandwidth. The filter 1 dB bandwidth shall be a minimum of 40 MHz.

3.2.1.2.7.1.3 Rejection. The filter rejection shall be greater than 17.6 dB ±45 MHz from the center frequency.

3.2.1.2.7.1.4 Amplitude ripple. The filter amplitude ripple shall be less than 1 dB across the 40 MHz bandwidth.

3.2.1.2.7.1.5 Insertion loss. The filter insertion loss shall be less than 1.5 dB.

3.2.1.2.7.1.6 Phase linearity. The phase linearity of the filter shall be less than 10 degrees peak to peak over a ±20 MHz frequency range.

3.2.1.2.7.2 Physical characteristics

3.2.1.2.7.2.1 Dimensions. The filter maximum envelope dimensions shall not exceed 5 x 1.25 x 0.625 inches.

3.2.1.2.7.2.2 Weight. The filter weight shall not exceed 0.6 pound.

3.2.1.2.8 Wideband pretransmission filter. An X-band pretransmission filter shall be employed to restrict the power spectrum of the 256 Mbit/sec quadriphase modulated signal before it is applied to the high gain antenna.

3.2.1.2.8.1 Functional characteristics

3.2.1.2.8.1.1 Frequency. The filter shall operate at a center frequency of 8.250 GHz.

3.2.1.2.8.1.2 Bandwidth. The filter 1 dB bandwidth shall be a minimum of 256 MHz.

3.2.1.2.8.1.3 Rejection. The filter rejection shall be greater than 13.2 dB ±18 MHz from the center frequency.

3.2.1.2.8.1.4 Amplitude ripple. The filter amplitude ripple shall be less than 1 dB across the 256 MHz bandwidth.

3.2.1.2.8.1.5 Insertion loss. The filter insertion loss shall be less than 1.5 dB.

3.2.1.2.8.1.6 Phase linearity. The phase linearity of the filter shall be less than 10 degrees peak to peak over a ±128 MHz frequency range.

3.2.1.2.8.2 Physical characteristics

3.2.1.2.8.2.1 Dimensions. The filter maximum envelope dimensions shall not exceed 5 x 1.25 x 0.625 inches.

3.2.1.2.8.2.2 Weight. The filter weight shall not exceed 0.6 pound.

3.2.1.2.9 RF transfer switch. Three identical, four-port waveguide transfer switches shall be provided to allow selection and coupling of: the biphase and quadriphase modulators to the X-band power amplifiers; the X-band amplifiers to the narrowband and wideband pretransmission filters; and the pretransmission filters to the high gain antennas (see Figure 1).

3.2.1.2.9.1 Functional characteristics. The transfer switches shall be of the four-port latching rotor type using curved waveguide to opposite ports. The transfer switch shall operate over the 8.0 to 8.4 GHz frequency range.

3.2.1.2.9.1.1 Insertion loss. The insertion loss between the input port and the output port of the transfer switch shall not exceed 0.2 dB.

3.2.1.2.9.1.2 VSWR. The voltage standing wave ratio of the transfer switch shall not exceed 1.2 to 1.0 at each terminal when all other terminals are terminated in a matched load.

3.2.1.2.9.1.3 Isolation. The isolation between the isolated terminals of the transfer switch shall not be less than 60 dB when all other terminals are terminated in a matched load.

3.2.1.2.9.2 Physical characteristics

3.2.1.2.9.2.1 Dimensions. The maximum envelope dimensions of the transfer switch shall not exceed 3.0 x 3.0 x 2.0 inches.

3.2.1.2.9.2.2 Weight. The weight of the transfer switch shall not exceed 1.0 pound.

3.2.1.2.10 Flexible waveguide. A flexible waveguide shall be provided to couple RF signals to the gimbal-mounted high gain antennas and eliminate the need for an RF rotary joint. The flexible waveguide shall be capable of repeated bending of ±90 degrees in the waveguide E-plane.

3.2.1.2.10.1 Functional characteristics. The flexible waveguide shall be constructed from seamless, nontwistable, beryllium copper and shall have a maximum length of 24 inches. The flexible waveguide shall be WR-112 for operation over the 8.0 to 8.4 GHz frequency band.

3.2.1.2.10.1.1 VSWR. The voltage standing wave ratio of the flexible waveguide shall not exceed 1.1 to 1.0 when terminated in a matched load over ±90 degrees of bending.

3.2.1.2.10.1.2 Insertion loss. The insertion loss of the flexible waveguide shall not exceed 0.1 dB per foot over ±90 degrees of bending.

3.2.1.2.10.2 Physical characteristics

3.2.1.2.10.2.1 Dimensions. The maximum envelope dimensions of the flexible waveguide shall be 1.5 x 0.8 x 24 inches.

3.2.1.2.10.2.2 Weight. The weight of the flexible waveguide shall not exceed 0.6 pound for the 24 inch length.

3.2.1.3 Data handling group. The data handling group of the WBC module shall contain the following components:

(a) LCGS speed buffer unit

(b) MODS controller unit

(c) Data interface unit

3.2.1.3.1 LCGS speed buffer. The LCGS speed buffer (hereafter called the buffer) accepts 128 Mbit/sec data from the thematic mapper, then edits and buffers the data for two purposes:

(a) To reduce the data rate to the LCGS to 20 Mbit/sec

(b) To reformat the data to line-sequential rather than column sequential, as in the data format when leaving the MODS A/D converter (see Figure 7).

3.2.1.3.1.1 Interface signals. The buffer shall conform to the module interface requirements specified in paragraph 3.2.1.1.2.1 of this document. Additional module internal interface signals shall be as shown in Table 1 and Figure 7.

Table 1. LCGS Speed Buffer Interface Requirements

Interface Line	Bit Rate			Signal Level	Comments
	Maximum	Nominal	Minimum		
Input Data	18 Mbit/sec	16 Mw/sec	1 Kbit/sec	TBD	8-bit parallel at WC rate
Word clock (WC)	18 Mbit/sec	16Mbit/sec	1 Kbit/sec	TBD	
Minor frame clock (FC)	$\frac{\text{Max. WC}}{100}$	16 Mbit/sec	$\frac{\text{Min. WC}}{100}$	TBD	
Major frame clock	$\frac{\text{Max. FC}}{8390}$	19 bit/sec	$\frac{\text{Min. FC}}{8390}$	TBD	
Output data	$\frac{\text{Max. WC}}{0.8}$	20 Mbit/sec	$\frac{\text{Min. WC}}{0.8}$	TBD	
DIU instruction	1/sec	NR	NR	T^2L	32-bit word
Secondary power	NA	NA	NA	TBD	

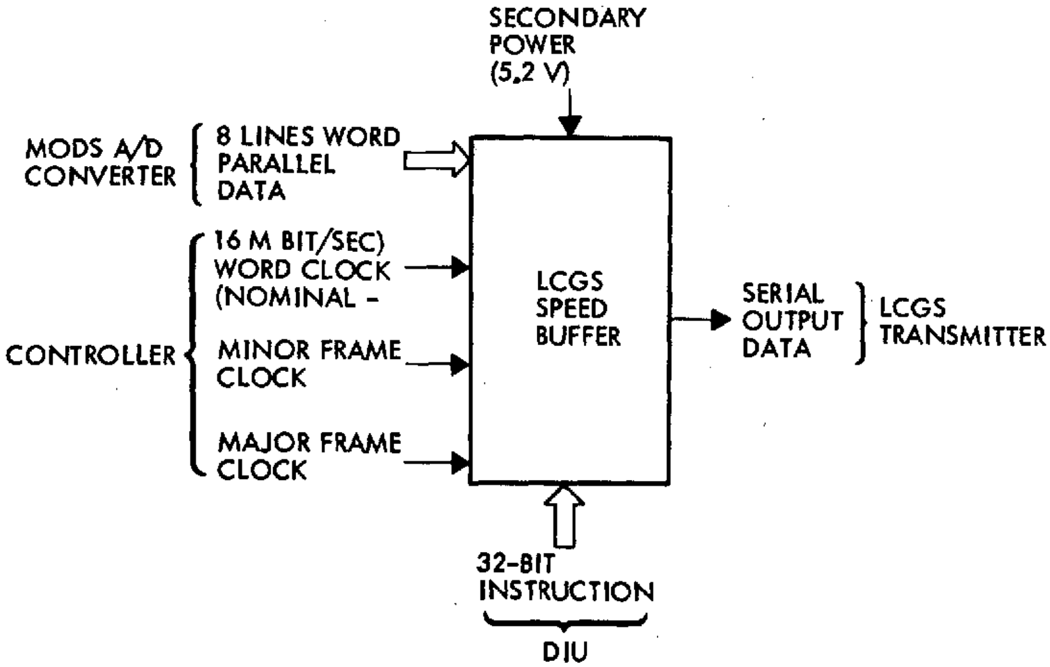

Figure 7. LCGS Speed Buffer Interface Block Diagram

3.2.1.3.1.2 Functional requirements

3.2.1.3.1.2.1 Data editing. The buffer shall perform in good command one of the following four data editing functions:

(a) All data from any one of seven possible bands, full swath

(b) All data from any two bands for one-half swath

(c) All data from any four bands for one-quarter swath

(d) All data from all seven bands, full swath, reduced resolution.

3.2.1.3.1.2.2 Swath selection. Any portion of the swath may be selected in paragraph 3.2.1.3.1.2.1 except that in cases (b) and (c) the bands must be from the same portion of the swath. The swath may start at any one of 16 points.

3.2.1.3.1.2.3 Reduced resolution. The center pixel of each 9-pixel square is selected and transmitted, reducing the resolution by 9X.

3.2.1.3.1.2.4 Data reformatting. The data is reformatted from column sequential to line sequential as shown in Figure 5.

3.2.1.3.1.3 Physical characteristics

3.2.1.3.1.3.1 Weight. The maximum weight of the LCGS speed buffer shall be (TBD) pounds.

3.2.1.3.1.3.2 Dimensions. The maximum envelope dimensions of the LCGS speed buffer shall be (TBD) inches.

3.2.1.3.2 MODS controller unit. The MODS controller controls the formatting of the MODS multipliers which are located in the instrument modules. In addition to this it provides timing signals to the LCGS speed buffer.

3.2.1.3.2.1 Interface signals. The MODS controller interface signals and basic oscillator stability are listed in paragraph 3.2.1.1.2.3 of this specification.

3.2.1.3.2.2 Frame size and organization. Major and minor frame size and organization shall be firmware controlled. The following control bits are listed as a minimum:

(a) Major frame timing: 13 bits (8192)
The major frame may be quantized in as many as 8192 minor frame groups.

(b) Minor frame timing: 7 bits (128)
The minor frame may be quantized in as many as 128 word groups.

(c) Work timing: Word lengths of 4, 8, 16 and 32 are selectable (hardwire is acceptable).

3.2.1.3.2.3 Frame synchronizator. Frame synchronization is programmable with firmware. Synchronizator codes may be any length at the major frame, minor frame or both.

3.2.1.3.2.4 Housekeeping data. Housekeeping data is programmed in the same manner as experiment data.

3.2.1.3.2.5 Data rate. The controller shall be capable of operating at any data rate from 1 to 128 Kbit/sec. Data rate is controlled by the internal clock rate.

3.2.1.3.2.6 Clock stability. Clock stability is specified in paragraph 3.2.1.1.2.3 of this document.

3.2.1.3.2.7 Physical characteristics

3.2.1.3.2.7.1 Weight. The maximum weight of the MODS controller shall be (TBD) pounds.

3.2.1.3.2.7.2 Dimensions. The maximum envelope dimensions of the MODS controller shall be (TBD) inches.

3.2.1.3.3 Data interface unit. The data interface unit shall contain a remote multiplexer for data acquisition and a remote decoder for command distribution. Power for the data interface unit shall be supplied by an independent power converter which receives power directly from the primary power bus.

3.2.1.3.3.1 Functional characteristics

3.2.1.3.3.1.1 Remote multiplexer

(a) Multiplexer configuration. Each multiplexer shall have a minimum of 64 inputs that can be used for analog, bilevel, and serial digital signals. The signal handling capability shall allow a user to use any input for analogs, any input for bilevel (in groups of 8), and any of 32 inputs for serial digital signals.

(b) Input signal levels. All inputs of the multiplexer shall have an input impedance of 5 Mohms minimum in the normal mode and 10 Kohms minimum during sampling. The multiplexer shall be capable of surviving a short circuit to ±10 VDC maximum on any one input for an indefinite time.

(1) Analog inputs (digitized to 8 bits)

Range	0 to +5 VDC
Z source	2 Kohms maximum
Accuracy	±30 MV

(2) Bilevel digital inputs

Logical "1"	+3.0 to +5.5 VDC
Logical "0"	0 to +0.8
Fault tolerance	±10 VDC
Z source	500 ohms minimum; 10 Kohms maximum

(3) Serial digital inputs (8 bits/word — telemetry, 16 bits/word — computer data)

Clock rate*	1.024 Mbps
Gate width*	Envelopes 8 or 16 clock pulses
Input data	
Logical "1"	+3.0 to 5.5 volts
Logical "0"	0 to +3.0 volts
Z Source	500 ohms minimum

3.2.1.3.3.1.2 Remote decoder

(a) Characteristics. Each remote decoder shall have a minimum of 32 pulse command outputs and 7 serial magnitude command outputs. Pulse commands may serve as relay driver inputs.

*These signals are multiplexer outputs with the same voltage and impedance characteristics as those shown for pulse commands in the following paragraph.

(1) Pulse commands

Pulse duration	7.8 msec minimum
Logical "1"	+3.0 to +5.5 volts
Logical "0"	0 to +0.8 volt
R source at "0"	500 ohms minimum

(2) Magnitude commands

Clock rate*	1.024 Mbit/sec
Gate width*	Envelopes 16 clock pulses
Command word*	16 bits serial

(b) Remote unit expansion. For using modules needing more commands, the addition of expanders shall augment the capability in increments of 7 and 32 per expander to up to 56 serial digital commands and 256 pulse commands expanders. For using modules with requirements for more than 64 data channels, the addition of expanders shall augment the capability of each DIU in increments of 64 channels up to 512 channels.

3.2.1.3.3.2 Physical characteristics

3.2.1.3.3.2.1 Weight. The weight of the data interface unit shall not exceed 1.2 pounds.

3.2.1.3.3.2.2 Dimensions. The maximum envelope dimensions of the data interface unit shall not exceed 6.0 x 8.0 x 1.0 inches.

3.2.1.4 Power conditioning. Power conditioning for the module shall be accomplished with the following components:

(a) Bus protection assembly

(b) Secondary power converter

A block diagram of the power conditioning equipment is shown in Figure 8.

3.2.1.4.1 Bus protection assembly. The bus protection assembly shall provide the following functions:

(a) Fusing for the +28 volt module primary power

(b) Fusing for the +28 volt heater power

(c) Safe mode bus logic.

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*These signal outputs have the same voltage and impedance characteristics as those shown for pulse command.

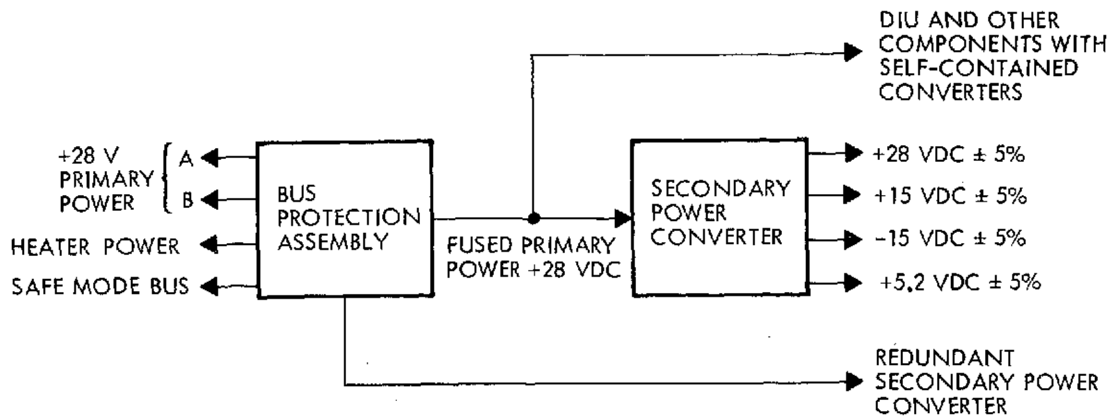

Figure 8. Power Conditioning Equipment

3.2.1.4.1.1 Module primary power fusing. Redundant fusing shall be provided for each secondary power converter as follows:

Converter No. 1: 3.7 ±0.5 amp

Converter No. 2: 2.7 ±0.5 amp

3.2.1.4.1.2 Heater power fusing. Redundant fusing shall be provided for each heater line as follows:

Heater No. 1: TBD amp

Heater No. 2: TBD amp

3.2.1.4.1.3 Safe mode bus logic. Logic shall be provided to turn off the following module components when the safe mode line voltage is below 0.4 volt: all module components (heater power remains on).

3.2.1.5 Electrical grounding

3.2.1.5.1 Primary DC power. All primary DC power returns shall be isolated from chassis/case/structure by a minimum of 1 megohm resistance. These returns shall be routed, along with the DC power input lead, via unshielded twisted-pair wiring to the module-to-spacecraft interface connector for eventual grounding in the power module. Power conversion/DC isolator units or subunits shall be provided at each primary power user terminal to maintain the required DC isolation between the primary and secondary power distribution systems.

3.2.1.5.2 Secondary DC power. Secondary DC power distribution networks shall, in general, use multiple point grounding within the wideband communications module with returns through the module radiator panel and/or support structure. These networks shall be initially grounded adjacent to the secondary transformer winding in the power converter and at each user element.

3.2.1.6 Instrumentation. As a minimum the following instrumentation (telemetry) shall be provided within the module.

Signal Definition	Signal Type
(TBD)	(TBD)

3.2.1.7 Redundancy and functional expansion. Provision shall be made for redundant equipment with appropriate signal cross-strapping. The exact configuration shall be TBD for the mission.

3.2.1.8 Harness. The module harness shall provide all electrical interfaces between subsystem assemblies within the module and to the module/structure interface and test connectors. The harness shall be of modular design for maximum system flexibility. Installation or removal of the harness should be possible without removing electrical assemblies. It shall be possible to remove electrical assemblies without removing the harness. Cable strain relief or backshell potting shall be employed at all harness terminations.

Wire sizes shall be selected to hold round-trip voltage drops between source and load to 1 percent or less of the supply voltage. The minimum wire size for power and control circuitry shall be AWG No. 20. The minimum wire size for data or test circuitry shall be AWG No. 22. Under worst case conditions, wire temperature shall not exceed the temperature rating of the wire insulation.

3.2.1.9 Thermal. The module thermal control system design constraints are presented in the following paragraphs.

3.2.1.9.1 Module thermal requirements. The module thermal design shall satisfy the following on-orbit requirements:

- The module shall be capable of operation when the heat sink temperature is $\pm 20^{\circ}$F greater than the most severe predicted operating temperatures, where heat sink is defined as the structure or panel to which the electronic black boxes and other module equipment is mounted. These limits will be termed heat sink qualification temperatures. Less severe temperature limits can be used for components that might be damaged by the qualification temperatures if a waiver is obtained from the contractor.

- The module shall be designed so that the nominal set point temperature of the heat sink is 70°F with electrical heaters turned off.

- Electrical heaters shall be incorporated to maintain the orbit-everage temperature at the module attachment locations above 60°F with the heater response approximating a sine pulse over one orbital period (rather than a step-input pulse).

- All module heat dissipation shall be radiated to space from the outboard facing panel.

- The surfaces of the module, except for the panel radiator areas, shall be thermally insulated with multilayer insulation, such that the effective emissivity, $\epsilon \leq 0.01$.

3.2.1.9.2 Module/structure assembly thermal interfaces. The design of the module thermal control system shall consider the following interface constraints:

- The structure assembly/module attach point temperature will be 70 ±10°F.

- Each module attachment fitting on the structure assembly will have a thermal resistance greater than 5 hr-°F/BTU.

- The effective emittance, ϵ, of the structure assembly/module insulation barrier will be ≤ 0.02.

3.2.1.9.3 Heater power constraints. Module thermal control system heater power shall not exceed 0 watts under normal operating conditions, and 16 watts under the most severe cold operating conditions that consider predictable variations in duty cycle and heating environment as well as parameter uncertainties in thermal properties, heating environment, insulation heat loss, etc.

3.2.2 Physical characteristics

3.2.2.1 Mechanical

3.2.2.1.1 Envelope. The module envelope shall be as shown in ICD 40.1.

3.2.2.1.2 Module volume. The module will have a volume of approximately TBD cubic feet and a maximum load carrying capability of 600 pounds of equipment. Components may be mounted to the outboard facing panel, to non-outboard surfaces, and the module frame members.

3.2.2.1.3 Module weight. The total weight of the WBC module for the minimum and nominal configurations shall not exceed the listed weights.

Baseline Module (lb)	Nominal Module (lb)
TBD	TBD

3.2.2.1.4 Module center of gravity. The center of gravity of the module shall be located within TBD.

3.2.2.1.5 Attach-points. The attach-points between the module and the spacecraft shall be as shown in ICD 40.1.

3.2.2.1.6 Module/structure interface connector. The module/structure interface connector shall be provided by the system contractor and shall be mounted on the side face of the module. It is required that the connector position be maintained as specified to ensure interchangeability of modules.

3.2.2.1.7 Equipment expansion volume. The components shall be arranged in the module such that a minimum of TBD feet squared of the outboard facing panel is left vacant in the center of the module for the addition of up to TBD pounds of mission-peculiar WBC hardware.

3.2.2.2 Electrical

3.2.2.2.1 Power. The total power required for the WBC module shall not exceed TBD watts. Allocation of this power is as follows:

Baseline configuration requirement: TBD watts
Redundancy and expansion capability: TBD watts

Power consumption of the WBC units shall be within the power allocation in ICD 10.2.

3.2.2.2.2 Commands. The WBC module shall distribute commands to other modules and/or users in accordance with ICD 10.3.

3.2.2.2.3 Telemetry. The WBC module shall accept telemetry inputs as listed in ICD 10.4.

3.2.2.2.4 Signal and power distribution. The WBC module harness shall provide all intramodule electrical connections in conformance to ICD 40.5.

3.2.2.2.5 Functional expansion and redundancy provisions

3.2.2.2.5.1 Functional expansion. The module shall be capable of accommodating the following additional components without structure modification.

Component	Maximum Size	Maximum Weight
Tape recorder	TBD	TBD
TBD		

3.2.3 Reliability. Compliance with reliability requirements shall be taken by prediction techniques in conformance with EOS-4.1. The allocated reliability for the WBC module baseline configuration operating under conditions specified herein for a TBD period is TBD.

Where mission objectives require changes to the baseline configuration, the reliability allocation shall be as specified in the mission specifications.

3.2.4 Maintainability. The WBC module shall be designed to emphasize accessibility and interchangeability. Field maintenance will be limited to checkout, removal, and replacement of equipment at the integral unit level. An integral unit is defined as a physical package containing factory-assembled parts.

3.2.5 Environmental conditions. The WBC module shall be designed to withstand or shall be protected against the worst probable combination of environments as specified in SP-11 and as implemented in SP-115.

3.2.6 Transportability. Transportability requirements shall be considered in the design of the WBC module such that it can be transported by all standard modes with a minimum of protection. Special packaging may be used, as required, to ensure that common carrier transportation does not impose design restrictions.

3.3 Design and construction

3.3.1 Parts, materials, and processes

3.3.1.1 Selection of parts, materials, and processes. Selection of parts, materials, and processes (PMP) shall be to the requirements identified in MIL-STD-143 Group I. Such selections shall be identified as standard PMP. When PMP selection cannot be made within this requirement, the item is nonstandard and justification must be made in accordance with MIL-STD-749. Control of this action shall be effected by the EOS Parts, Materials and Processes Control Board (PMPCB), with the PMPCB approving all selections, both standard and nonstandard, in accordance with EOS Document EOS-TBD, the System Effectiveness Program Plan.

3.3.1.2 Program authorized parts list. All selections of electronic parts shall be identified and authorized by the EOS program authorized parts list (PAPL) EOS-3.3-1. This list will reflect all PMPCP electronic part selections.

3.3.1.3 Program authorized materials list. All selections of materials shall be identified and authorized by the EOS program authorized materials list (PAML) EOS-3.3-2. This list will reflect all PMPCB material selections.

3.3.1.4 Program authorized processes list. All selections of processes shall be identified and authorized by the EOS program authorized processes list (PAPRL) EOS-3.3-3. This list will reflect all PMPCB process decisions.

3.3.1.5 Dissimilar metals. To avoid electrolytic corrosion, dissimilar metals shall not be used in direct contact unless protection against corrosion has been provided in accordance with MIL-E-8983 and MIL-STD-454, Requirement 16.

3.3.1.6 Magnetic materials. Magnetic materials shall be used only if necessary for equipment operation. Those magnetic materials which are used shall have minimum permanent, induced, and transient magnetic fields.

3.3.1.7 Fungus-inert materials. Materials which are fungus-inert, in accordance with MIL-E-8983, and MIL-STD-454, Requirement 4, shall be used.

3.3.1.8 Flammable toxic and unstable materials. Flammable, toxic, and unstable materials shall not be used.

3.3.1.9 Finish. The surface of each component of the subsystem shall be adequately finished to prevent deterioration from exposure to the specified environments that might jeopardize fulfillment of the specified performance. The finish shall also meet the applicable bonding requirements. Thermal properties of the finishes used on the component shall be compatible with the requirements given in detail in applicable equipment specifications. The requirements for finishes identified in MIL-E-8983 shall be implemented.

3.3.1.10 Outgassing. Low outgassing polymeric materials shall be used where sensitive thermal control and other surfaces are in direct line-of-sight and where temperature differences can exist between such surfaces.

3.3.2 Electromagnetic radiation

3.3.2.1 EMC/EMI requirements. The wideband communications module, and all internal units, equipments and/or components comprising a part thereof, shall be designed for compliance with the radiated emission and susceptibility requirements of NASA/JSC Specification SL-E-0002 as modified or amended by 3.3-4, "Electromagnetic Compatibility Control Plan, EOS Program," and 3.3-5, "EMI/EMS Limits and Test Methods, EOS Program." The conducted emission and susceptibility requirements of the aforementioned documents are applicable only at the module-to-spacecraft structure interface. EMI/EMS levels at interfaces within the module shall be controlled to the extent necessary to ensure self-compatibility of the module subsystem with a safety margin of at least 6 dB.

3.3.2.2 EMI tests. Design qualification tests shall be performed on the fully assembled WBC module (qualification model) to verify compliance with the applicable electromagnetic interference and susceptibility control requirements. The general test methods of MIL-STD-462, as modified or amended by 3.3-5, "EMI/EMS Limits and Test Methods, EOS Program," shall be used during this test program except that, wherever practical, spectrum analyzers or other forms of rapid display and analysis equipment may be used in lieu of equipment requiring manual tuning and data collection.

3.3.2.3 Electrical bonding. All metallic members of the basic WBC module radiator panel and support structure shall be electrically bonded to each adjacent member to form an electrically continuous, equipotential ground reference plane. The maximum allowable DC impedance across any one structural joint shall be 2.5 milliohms with an overall design goal of 10 milliohms, or less, between any two diametrically opposite points on the module. The general methods of MIL-B-5087, Class R, may be used for implementation of these requirements.

3.3.2.4 Equipment mounting pads. Surfaces on the module radiator panel and structure which are intended for unit, equipment or component mounting shall be free of paint, anodize, or other nonconductive finishes. The maximum DC impedance between the baseplate of any electrically active unit, equipment or component and the module radiator panel and structure shall be 2.5 milliohms.

3.3.2.5 Electrical connectors. All interface electrical connectors, both plug and receptical, which form a part of the unit and cable RF shielding system within the module shall have electrically conductive body shells, free of nonconductive finishes. They shall also have provisions for terminating (grounding) the shields on the interconnecting electrical harness. The maximum DC impedance between the shield termination point and the baseplate of the parent unit shall be 2.5 milliohms.

3.3.2.6 Electrostatic bonds. Electrically passive components or appurtenant structures which are attached to the basic module structure through thermal isolators shall be provided with electrostatic bonds having a DC impedance of 1.0 ohm, or less. The general methods of MIL-B-5087, Class S, may be used for implementation of this requirement.

3.3.3 Nameplates and product marking. Each subsystem shall be identified in accordance with MIL-STD-130 as implemented by EOS Specification 3.3-6, "Marking of Parts and Assemblies." Where practical, a minimum character height of 0.09 inch shall be used. Marking shall include the manufacturer's name or initials, assembly part number and revision status, assembly serial number, contract number, and others as delineated in the component equipment specifications.

3.3.4 Storage. The wideband communications module and component equipment shall be capable of being stored for a minimum of five years with subsequent satisfactory performance to specification.

3.3.5 Workmanship. The subsystem and its component equipment shall be constructed, finished, and assembled in accordance with MIL-STD-454, Requirement 9, and the specifications and drawings specified herein.

3.3.6 Interchangeability and replaceability. The WBC module shall be designed to permit removal and replacement of components with a minimum of disturbance of associated or adjacent equipment. Design for service and access shall conform to the principles and requirements of MIL-STD-470.

3.3.7 Safety. The subsystem shall be designed to meet or exceed the requirements of EOS-3.3-7 as implemented by EOS-4.1. The design critiera include but are not limited to those set forth in MIL-STD-882.

3.3.8 Human performance/human engineering. MIL-STD-1472A, "Human Engineering Design Criteria for Military Systems," shall be used for the design of man/machine interfaces.

3.4 Documentation. Documentation shall be prepared in accordance with EOS Document 3.3-8, "Configuration Management Plan."

4. QUALITY ASSURANCE PROVISIONS

4.1 General. The quality assurance program controls shall be in accordance with the requirements of MIL-Q-9858 as augmented by EOS-4.1.

4.1.1 Responsibility for inspection and test. Unless otherwise specified in the contract or purchase order, the supplier is responsible for the performance of all inspection and test requirements as specified herein. Except as otherwise specified, the supplier may use his own facilities or any commercial laboratory acceptable to the government. The government reserves the right to perform any of the inspections set forth in the specifications where such inspections are deemed necessary to ensure that supplies and services conform to prescribed requirements.

4.2 Quality conformance inspections

4.2.1 Category I tests

4.2.1.1 Development tests. Development tests shall be conducted to verify the module performance parameters, proper interfacing between components of the module, and proper interfacing between the module and other spacecraft modules. The development tests shall be conducted at the unit and/or integrated module levels. The type of developmental evaluation or test to be applied to verify satisfaction of the requirements defined in this specification are outlined in Table 2. The tests shall be conducted in accordance with EOS-4.2 and EOS-4.3.

4.2.1.2 Qualification tests. Module qualification shall be accomplished by means of qualification tests on the individual units and/or as a normal consequence of having the units installed in the qualification test space vehicle during its qualification testing. The types of qualification evaluation or test to be applied to verify satisfaction of the requirements of this specification are outlined in Table 2. Qualification test verification methods and requirements shall be as defined in the EOS-4.2.

4.2.1.2.1 Components. As a minimum, the following types of component qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, are measured before and after the environmental tests.

Table 2. Verification Cross Reference Index

VERIFICATION CROSS REFERENCE INDEX

LEGEND: N/A - Not Applicable
I - Inspection
A - Analysis
S - Similarity
T - Test

SECTION 3 REQUIREMENTS	Not Applicable	Acceptance			Qualification				SECTION 4 VERIFICATION REQUIREMENTS
		I	A	T	I	A	S	T	
(TBD)									

SAMPLE

- Random vibration. Each component will be subjected to a random vibration from 20 to 2000 Hz in all three axes while electrically powered in the maximum normal load condition. Performance will be continuously monitored to detect failures (either hard or trends).

- Thermal vacuum. Thermally cycle the component beyond expected orbital temperatures while at an orbital vacuum condition. The component is operating continuously and detailed performance parameters are measured at each temperature extreme.

- Electromagnetic interference (EMI) and susceptibility (EMS). Engineering EMI and EMS data are measured to provide diagnostic information for the module EMI/EMS qualification.

4.2.1.2.2 Module. As a minimum, the following types of module qualification tests shall be included:

- Functional. Performance parameters, electrical and/or mechanical, and electrical interface characteristics are measured before and after the environmental tests.

- Acoustics. Acoustics testing will be performed with the module mounted in a receptacle which provides acoustics shielding similar to the Observatory structure. The module will be heavily instrumented to verify design adequacy of interconnections (electrical and mechanical) between the module and other Observatory elements. The test data will be used to confirm estimates of the vibration environment for components mounted on radiators and establish procedures for acceptance testing. The module will be electrically powered and continuously monitored via the data bus to detect failures and out-of-tolerance performance trends.

- Thermal vacuum. The objectives of the module thermal qualification test are to determine: 1) maximum and minimum operating temperatures for the module for worst-case environments and operating duty cycles, 2) temperature levels and temperature variations of the module adjacent to the interface fittings, and 3) heater power requirements for cold-case conditions.

  The test will be conducted in a thermal-vacuum chamber with liquid nitrogen cold walls. The module will be mounted on a fixture simulating the spacecraft frame. The properties of this fixture shall permit its temperature to be held constant at any level between 0 and 100°F.

  A heating source for the radiators will approximate the absorbe flux of the external environment. This can be done with electrical heaters, infrared lamps, or other techniques where the absorbed heating can be determined accurately.

- Power bus and data bus will be tested in excess of their operational limits to determine design margins and compliance with the interface specification.

- Detailed performance data will be measured to determine module specification values.

- Thermistor/heater control and calibration will be determined.

- EMI and EMS. EMI measurements will determine the levels radiated on each module electrical interface and verify there is adequate design margin to ensure non-interference with other modules. EMS measurements will verify that each module has adequate susceptibility margin to prevent performance degradation resulting from other module allowable radiation levels. The general test methods of MIL-STD-462, as modified or amended by EOS-3.3-5, shall be used during this test program except that, wherever practical, spectrum analyzers or other forms of rapid display and analysis equipment may be used in lieu of equipment requiring manual tuning and data collection.

4.2.1.2.3 Inspection sequence. The inspection sequence shall be governed by the following:

- Examination of the module shall be performed prior to functional testing.

- Functional tests shall be performed prior to, during, where appropriate, and following environmental testing. The functional testing conducted at the conclusion of an environmental test may serve as the functional testing to be performed prior to the next environmental test.

- Environmental testing may be performed in any sequence.

4.2.1.2.4 Failure criteria. The module shall exhibit no failure, malfunction or out-of-tolerance performance degradation as a result of the examinations and test specified herein. Any such failure, malfunction or out-of-tolerance performance degradation shall be cause for rejection.

4.2.2 Test conditions. Unless otherwise specified, the conditions under which all inspections are accomplished shall be specified below:

4.2.2.1 Atmospheric and environmental. All examinations and tests shall be conducted under the local prevailing pressure at the time of test, at a temperature of 75 ±5°F and at a relative humidity of 55 percent or less.

4.2.2.2 Deviations of atmospheric conditions. When tests are performed with atmospheric conditions substantially different from the specified values, proper allowances for changes in instrument readings shall be made to compensate for the deviation from the specified conditions.

4.2.3 Equipment warmup time. The equipment warmup time shall be less than 1 minute.

4.2.4 Temperature stabilization. Temperature stabilization shall have been achieved when temperature of the equipment mounting structure varies not more than 5°F during a period of 15 minutes.

4.2.5 Measurements. Measuring instruments used to determine functional parameter values (such as voltage, frequency, current, flow, pressure, etc.) shall indicate true values with an accuracy determined by the tolerance allowed for the parameter variation itself, such that the measuring instrument shall not introduce an uncertainty greater than 10 percent of the allowable variation of the measured parameter. However, no such measurement accuracy shall be required to exceed 0.5 percent of the required value of the parameter unless otherwise specified.

Except as specifically noted in the inspection methods, tolerances for environmental test conditions shall be defined as follows:

Temperature: ±5°F

Barometric pressure: ±10 percent

Relative humidity: +5, -10 percent

Time: ±5 percent

Vibration amplitude (sine): ±10 percent

Vibration frequency: ±2 percent

4.3 Category II tests. (Not applicable)

4.4 Analyses. The following requirements of Section 3 shall be verified by review of analytical data:

3.2.3 Reliability. To be verified by analysis in accordance with Section TBD of EOS-4.1.

3.3.6 Safety. To be verified by analysis in accordance with Section TBD of EOS-4.1.

5. PREPARATION FOR DELIVERY

5.1 General. The wideband communications module as specified herein shall be protected from degradation by environments anticipated during shipment, handling, and storage. Standard commercial packaging practices are acceptable provided they fulfill these requirements.

5.2 Preservation and packaging

5.2.1 Cleaning. The WBC module shall be clean and free of contaminants which might impair its use prior to packaging.

5.2.2 Attaching parts. When attaching parts, such as nuts, bolts, washers, etc., accompany the WBC module they shall be preserved, bagged, appropriately identified, and attached to, or adjacent to, the fitting for which they are intended.

5.2.3 Electrical connectors. All electrical connectors shall be capped with protected dust caps. Caps used shall be a friction fitting or threaded type which do not require tape or a mechanical device to secure.

5.2.4 Critical surfaces. External machined surfaces and mounting surfaces of the WBC module shall be protected with protective pads. Materials used for pads shall not cause item deterioration.

5.2.5 Wrapping. The WBC module shall be wrapped or bagged using anti-static polyethylene film.

5.2.6 Cushioning. When required for protection, the WBC module shall be cushioned using a suitable resilient foam cushioning material such as polyethylene, polyurethane, or polystyrene.

5.3 Packing. The WBC module shall be packed in a suitable shipping container designed for one item only. The shipping container used shall provide protection to the item against corrosion, deterioration, and damage during shipment from the source of supply to the receiving activity. Containers shall comply with applicable tariffs and regulations for particular modes of transportation when so shipped.

5.3.1 Storage conditions. The WBC module shall not be adversely affected by storage within its container for a period of three years from date of final acceptance at TRW Systems, at temperatures between 60 and 90°F and relative humidities of 60 percent or less.

5.3.2 Shipping conditions. The WBC module shall be capable of withstanding the following environments:

Temperature:	+160°F in an unsheltered area (125 +35°F due to solar radiation) and -40°F in accordance with restricted air transport criteria.
Humitidy:	Up to 100% in an unsheltered area.
Rough handling:	Capable of withstanding and physically protecting the unit from rough handling during shipping by common carrier.

5.4 Marking for shipment. Each WBC module and shipping container shall be marked with the following:

(a) Item nomenclature

(b) Part number

(c) Contract or purchase order number

(d) Manufacturer's name

(e) Manufacturer's part number and serial number (on item container only)

(f) Quantity

(g) Date of manufacture (on item container only)

(h) Fragile – Handle With Care (when applicable)

(i) Space Vehicle Material – Do Not Open In Receiving Or Receiving Inspection (when applicable – shipping container only)

(j) Actual weight

5.4.1 Documentation. All required reliability and test documentation such as test reports, certifications, shipping invoices, etc., shall be either packed in the WBC module container or attached to the exterior surface of the shipping container. Attachment shall be such a manner as to preclude loss of this data during handling and shipment by common carrier.

6. NOTES

6.1 Definition of spacecraft configuration.

6.1.1 Minimum redundancy configuration. The minimum redundancy configuration is defined as the spacecraft configuration which contains the minimum redundancy of units necessary to ensure that no plausible single-point failure will prevent Observatory retrieval by the Space Shuttle system. For purposes of this specification this configuration is identified as the baseline spacecraft configuration.

6.1.2 Nominal redundancy configuration. The nominal redundancy configuration is defined as the spacecraft configuration which includes standby redundant units for most of the electronic assemblies to provide a "typical" redundancy level for long-life spacecraft.