NASA CR-143671

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

REPORT NO. 5: SYSTEM DESIGN AND SPECIFICATIONS

• Part 1: Observatory System Element Specifications

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GRUMMAN

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

REPORT NO. 5: SYSTEM DESIGN AND SPECIFICATIONS

• Part 1: Observatory System Element Specifications

Prepared For

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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ABBREVIATIONS

ACS	Attitude Control Subsystems
A.H.	Ampere hour
Bat.	Battery
Biø	Bi phase
BPS	Bits per second
CCIR	Consultative Committee on International Radio
C & DH	Communication and Data Handling
CMD	Command
dB	Decibels
DNTD	Descending mode time of day
EMC	Electromagnetic compatibility
E/No	Energy-to-noise spectral density
EOS	Earth Observatory Satellite
EPS	Electrical Power Subsystem
FEC	Forward error correction
FMEA	Failure Modes and Effects Analysis
GN_2	Gaseous nitrogen
GSE	Ground Support Equipment
LOS	Line of sight
MFR	Multifunction Receiver
MHZ	Megahertz
MPT	Maximum power tracking (of Solar Array)
MUX	Multiplexer
N_2H_4	Anhydrous hydrazine
Ni-Cd	Nickel-Cadmium
NRZ	Non-return to zero
OA	Orbit adjust
OA/RCS	Orbit Adjust/Reaction Control Subsystem
P.M.	Power Module
PCM	Pulse code modulation
PN	Pseudo noise
PSK	Phase shift keyed
RCS	Reaction Control Subsystem
RF	Radio frequency

ABBREVIATIONS (Cont)

RHCP	Right hand circularly polarized
S/C	Spacecraft
SE	Support Equipment
STDN	Spaceflight Tracking and Data Network
TDRS	Tracking and Data Relay Satellite
TM	Telemetry
VSWR	Voltage standing wave ratio

I

SCOPE

1 - SCOPE

This specification establishes the performance, design, and quality assurance requirements for the Earth Observatory Satellite (EOS) Observatory and Ground System program elements required to perform the Land Resources Management (LRM) "A" mission. The specification is divided into two parts. Part 1 contains requirements for the Observatory element with the exception of the Instruments Specifications which are contained in Report No. 2 of the EOS System Definition Study. Part 2 contains the Ground System requirements.

The EOS Basic Spacecraft shall be designed to accommodate follow-on EOS missions in addition to the LRM "A" mission requirements. Part 1 presents the EOS System Definition and the specifications for the LRM "A" mission, including the Basic Spacecraft Interfaces and Instrument Mission Peculiars, and it identifies the follow-on missions and design driver requirements to accommodate the follow-on missions. Since the overall program approach is a "Design to Cost Approach", cost targets for each of the EOS program elements are also identified in Part 1.

APPLICABLE DOCUMENTS

2 APPLICABLE DOCUMENTS

2.1 GOVERNMENT DOCUMENTS

2.1.1 SPECIFICATIONS

2.1.1.1 FEDERAL

None

2.1.1.2 MILITARY

- MIL-P-26536 C Propellant, Hydrazine
- MIL-P-27401 B Propellant, Nitrogen Pressurizing Agent
- MIL-W-5088
- MIL-C-17

2.1.1.3 NASA

- GSFC, EOS-410-04 Performance Specification for Spacecraft Subsystems, Sept. 1973.
- NASA S-320-G-1 General Environmental Test Specification for Spacecraft and Components.

2.1.1.4 OTHER GOVERNMENT SPECIFICATIONS (anticipated end item specifications)

- EOS-SY-120 System Specification for the Basic Spacecraft
- EOS-SY-130 System Specification for the Observatory System, Land Resources Management Mission A.
- EOS-SY-131 System Specification for the Observatory System, Land Resources Management Mission B.
- EOS-SY-132 System Specification for the Observatory System, Land Resources Management Mission C.
- EOS-SY-140 System Specification for the Observatory System, SEASAT Mission.
- EOS-SY-150 System Specification for the Observatory System, Solar Maximum Mission
- EOS-SY-160 System Specification for the Observatory System, Synchronous Earth Observatory Satellite.

- EOS-SY-170 System Specification for the Observatory System, TIROS-N Mission
- EOS-SY-180 Support Equipment Specification
- EOS-SY-190 Software Specification
- EOS-SS-200 Specification for the Communication and Data Handling Subsystem Module
- EOS-SS-210 Specification for the Electrical Power Subsystem Module
- EOS-SS-220 Specification for the Attitude and Control Subsystem Module
- EOS-SS-230 Specification for the Structure Subsystem
- EOS-SS-240 Specification for the Thermal Subsystem
- EOS-SS-250 Specification for the Orbit Adjust/Reaction Control Subsystem/Orbit Transfer Subsystem Module
- EOS-SS-260 Specification for the Electrical Integration Subsystem.

2.1.2 STANDARDS

2.1.2.1 FEDERAL

- Fed-Std-209A (10 August 1966) Clean Room and Work Station Requirements Controlled Environment.

2.1.2.2 MILITARY

- MIL-STD-810 B (15 June 67) Notice 1 to 4 (21 Sept. 70) Environmental test Methods
- MIL-STD-1246 A (18 August 67) Product Cleanliness Levels and Contamination Control Program.

2.1.2.3 NASA

- NASA SP-8005 - "Solar Electromagnetic Radiation"
- NASA SP-8067 - "Earth Albedo and Emitted Radiation"
- NASA SP-3024, Volume III through VII "Models of Trapped Radiation Environment"
- NASA, NSSDC 72-06, "Model of the Outer Radiation Zone Electron Environment"
- NASA, NSSDC 72-10, "The Inner Zone Electron Model AE-5"

- NASA, NSSDC 72-13 "A Model Environment For Outer Zone Electrons"

2.1.3 DRAWINGS

2.1.3.1 INTERFACE CONTROL DRAWINGS (to be prepared)

- 314-ICD-001 Basic Spacecraft/Mission Peculiar Equipment
- 314-ICD-002 Instrument Mission Peculiar/Equipment/Instruments
- 314-ICD-003 Observatory Communication/STDN/TDRSS
- 314-ICD-004 Wide Band Instrument Data/Primary Ground Station/TDRSS
- 314-ICD-005 Medium Band Instrument Data/Low Cost Ground Station/TDRSS
- 314-ICD-006 Observatory/Launch Vehicle
- 314-ICD-007 Observatory/Launch Support Systems.

2.1.4 TECHNICAL MEMORANDUM

- NASA TMX-64589 (10 May 71) — Terrestrial Environment (climate) Criteria Guidelines for use in Space Vehicle Development.
- NASA TMS-64627 (15 Nov. 71) — Space and Planetary Environment Criteria Guidelines for use in Space Vehicle Development

2.2 NON-GOVERNMENT DOCUMENTS

2.2.1 SPECIFICATIONS

- GAC-SP-1001 (TBD) — Mass Properties Control Requirements for Sellers, Earth Observatory Satellite, General Specification for
- GAC-SP-1002 (TBD) — Basic Mass Properties Control Requirements for Sellers, Earth Observatory Satellite, General Specification for.

REQUIREMENTS

3 REQUIREMENTS

3.1 SYSTEMS DEFINITION

The general design objective of the Earth Observatory Satellite Program is to provide a flexible, cost effective facility for conducting a broad range of earth remote sensing missions. The facility will consist of a general purpose, or standard spacecraft capable of accommodating a wide variety of instruments, and all ground data acquisition and processing systems necessary to provide data directly to the users.

3.1.1 PROGRAM DRIVER REQUIREMENTS

The top level program driver requirements are as follows:

(a) The EOS System shall provide a basic capability to perform Land Resources Management (LRM) missions and shall be adaptable, with minimum modification, to support the following mission categories:

- Earth Observation
  - o Marine and Water Resources and Pollution
  - o Ocean Dynamics and Sea Ice
  - o Weather and Climate
- Solar Observation
- Stellar Observation
- Inertial Pointing (EGRET)

A typical mission model comprising these missions is shown in Fig. 3-1.

(b) The EOS System design for LRM shall accommodate combined operational and R&D functions. Consideration shall be given to the integration of hardware and coordination of operations for this dual program relationship.

(c) The Basic Spacecraft shall be modular and standardized for the range of missions described in Paragraph 3. 1. 1a. Specialized Mission Peculiar equipment shall be provided above the interface of the Basic Spacecraft. The Basic Spacecraft contains three basic subsystem modules: (1) Altitude Control Subsystem (ACS) module, (2) Communications and Data Handling (C & DH) modules, and (3) Electrical Power Subsystem (EPS) module. A fourth module, Orbit Adjust/Reaction Control Subsystem (OA/RCS) module, completes the Basic Spacecraft. Mission Peculiar equipment includes the instruments, wide band data processing and communications hardware and orbit transfer modules.

(d) The EOS Observatory shall be designed to utilize Shuttle for economic and operational benefits. Capability for incorporating Shuttle retrieval and in-orbit resupply provisions shall be provided.

(e) Earth scanning revisit cycle for LRM missions shall be a maximum of 17 days. A design goal to 6 to 9 days revisit cycle should be considered.

(f) Data turn-around time for LRM missions shall be 24 to 48 hours. Turn-around time is defined as from time of data receipt at the earth-based receiving station to time of transmittal to the user.

(g) Basic LRM central processing facility data processing (output products) shall be digital and secondary output products shall be photographic. Data products shall include Computer Compatible Tapes and High Density Digital Tapes, black and white, and color images. Output products are required for up to 100 generic users. Central Processing throughput rate shall be capable of handling a minimum of 10^{10} bits per day and shall be expandable to as much as 10^{12} bits per day.

(h) LRM imaging and data acquisition shall be primarily of continental United States (CONUS). Capability shall also provide for International Data Acquisition (IDA) via TDRSS. Provisions shall also be made for optional on-board tape recorders to accomplish IDA.

(i) EOS payload data shall be radiometrically and geometrically corrected (with and without GCP's) before delivery to the data users.

(j) The EOS system for LRM, and the Basic Spacecraft shall each be designed to a target cost.

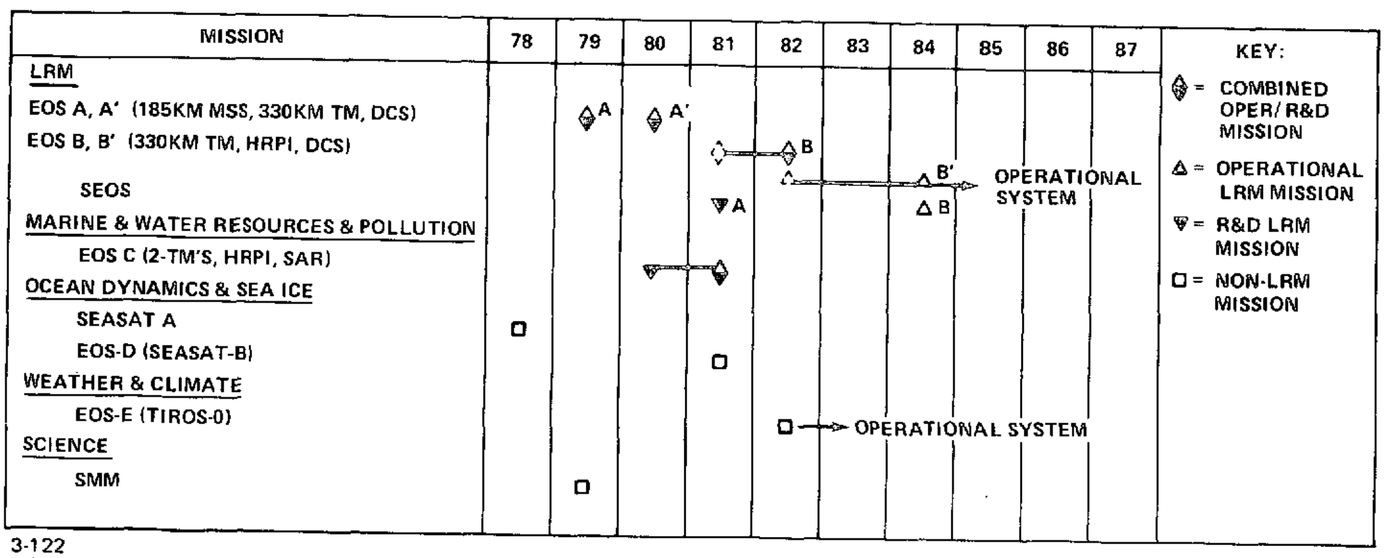

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Fig. 3-1 EOS Mission Model

3.1.2 GENERAL DESCRIPTION

3.1.2.1 PROGRAM ELEMENTS

The Observatory is one element of the Earth Observatory Satellite Program. The total program is illustrated in Fig. 3-2 and consists of the Launch Vehicle, Observatory, STDN Ground Station, TDRSS, Low Cost Ground Station, Project Control Center, Central Data Processing Facility and the Support Equipment.

3.1.2.2 OBSERVATORY SYSTEM ELEMENTS

The Observatory System Elements comprise the Observatory, Support Equipment and interfaces with the launch vehicles, communication nets and subordinate functional areas defined in Fig. 3-3.

3.1.2.2.1 Observatory

The Observatory shall consist of a Basic Spacecraft, instruments, mission peculiar equipment and software.

The Basic Spacecraft is illustrated in Fig. 3-4 and shall consist of a triangular structure to support the three common modules, ACS, EPS and Communication and Data Handling within the confines of the launch vehicle shroud. The Orbit Adjust/RCS module shall attach to the end of the lower bulkhead completing the Basic Spacecraft.

The instruments and mission peculiar equipment shall be mounted to the Basic Spacecraft upper bulkhead. Provisions shall be made to accommodate a variety of instruments and supporting equipment to support the missions defined in Paragraph 3.1.1.

The Observatory software consists of software modules linked to form a software package tailored for the specific spacecraft. The software modules are derived from three sources:

- Basic software, taken from the EOS software library without modification
- Adaptable basic software, taken from the EOS software library and adapted to the specific spacecraft
- Mission peculiar software, prepared especially for the specific spacecraft and its instruments.

3.1.2.2.2 Support Equipment

The Support Equipment shall support the Observatory during flight equipment development, manufacturing, assembly, acceptance testing, transportation, pre-launch and launch checkout operations.

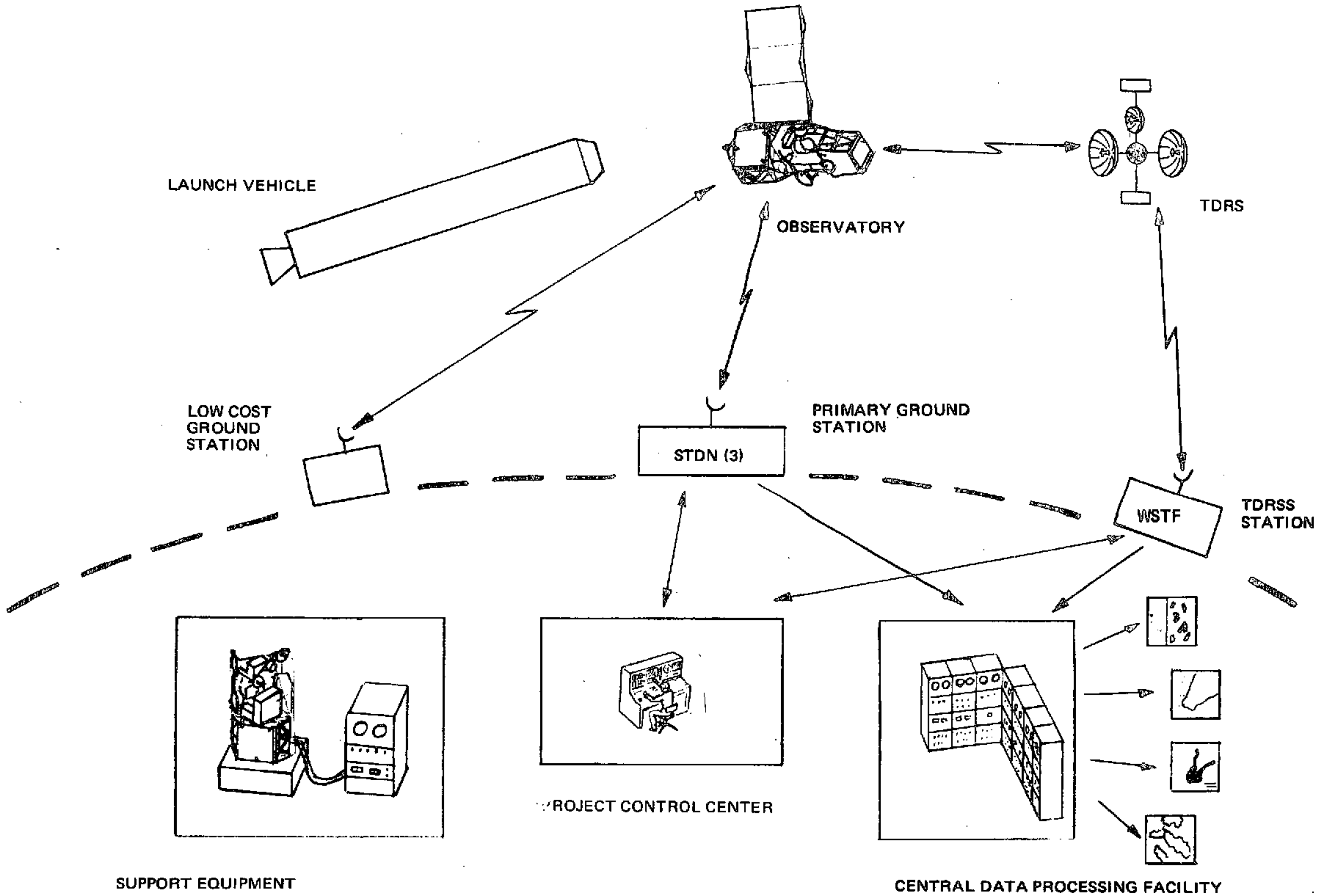

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Fig. 3-2 Earth Observatory Satellite System Program Segments

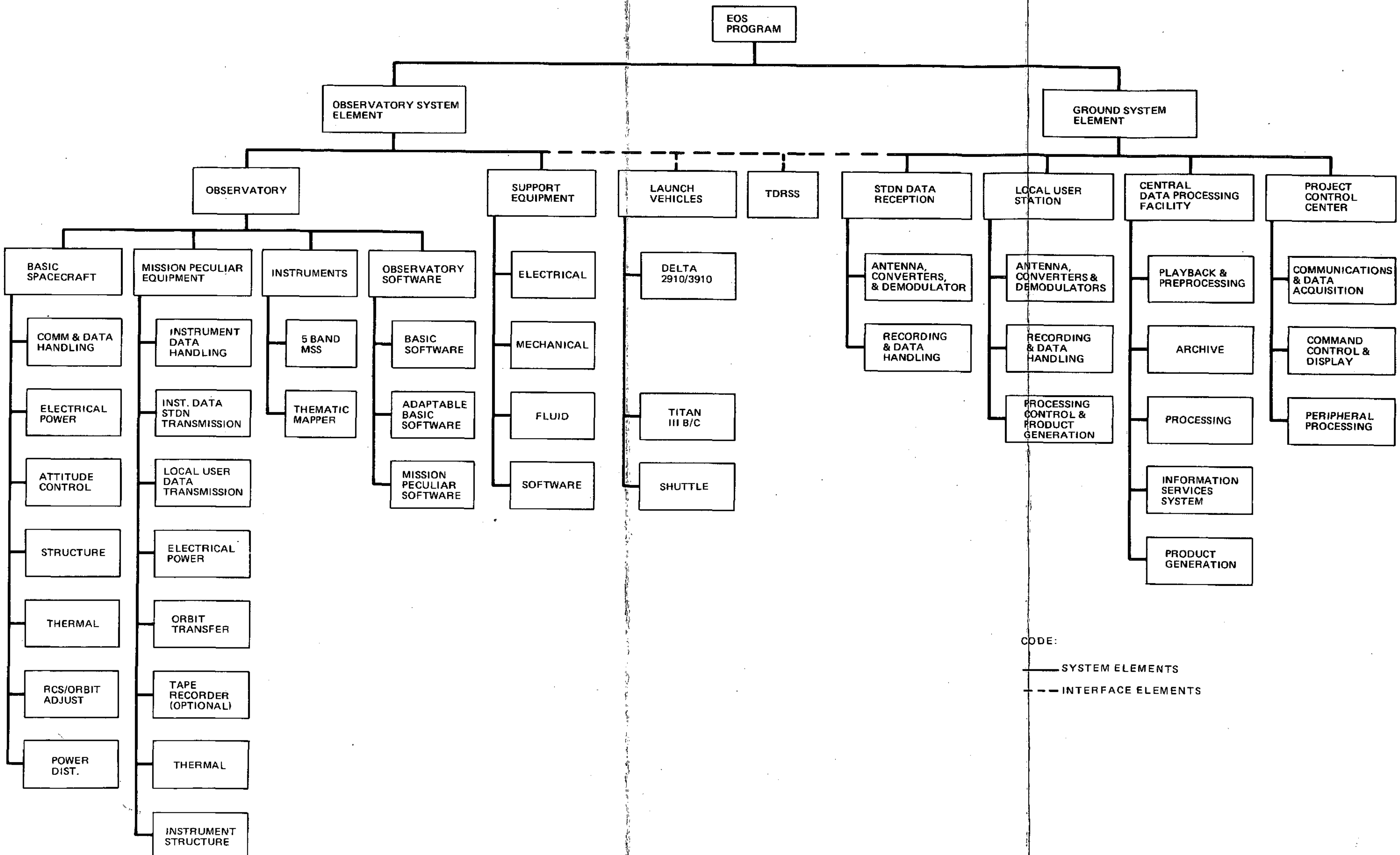

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Fig. 3-3 Program Functional Elements

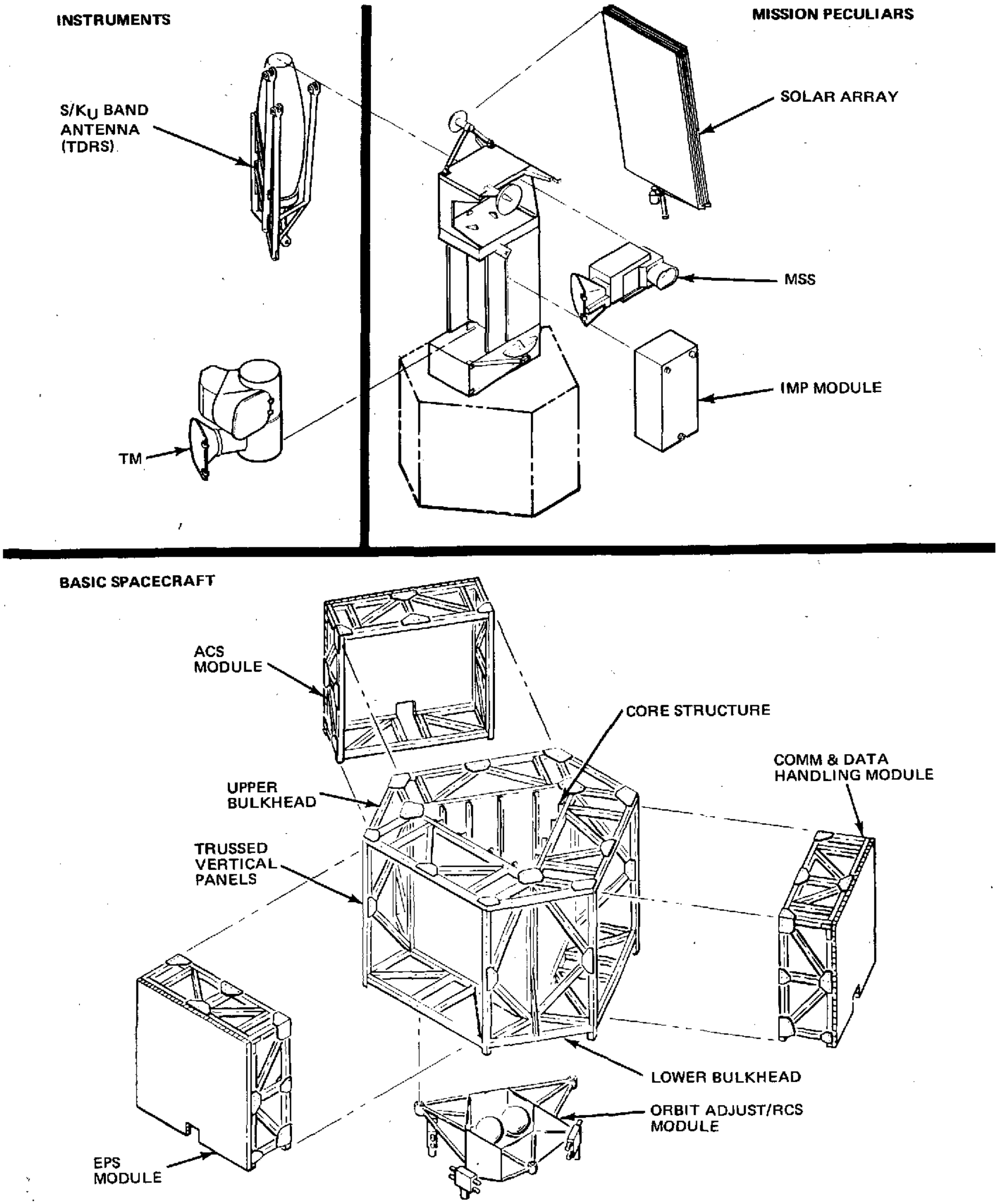

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Fig. 3-4 Observatory

3.1.2.2.2.1 Electrical

Electrical support equipment in conjunction with standard test equipment shall provide the capability to verify the integrity of all critical elements of the EOS, provide power and control functions, monitor performance, and provide failures indication of the EOS at the system, subsystem and replaceable component level. Failures within the electrical support equipment shall not induce failures within the S/C elements.

3.1.2.2.2.2 Mechanical

Mechanical support equipment shall provide the capability to: handle; transport; provide access to; support; install; align; weigh; balance, and mechanically maintain the EOS at the system, subsystem and replaceable component level.

3.1.2.2.2.3 Fluid (Liquid & Gaseous)

Fluid support equipment shall be capable of providing appropriate fluids at the required cleanliness, pressure, temperature and flow rate to the EOS fluid interfaces. In addition this equipment shall provide for the necessary measurements to insure the integrity of all EOS fluid lines. Fluid support equipment subject to degradation of performance by contaminants shall be provided with devices to maintain contamination at acceptable levels.

3.1.2.2.3 Interfaces

3.1.2.2.3.1 Launch Vehicles

The Observatory shall be launched by the Delta 2910 (LRM Mission A), and be compatible with the Delta 3910, Titan III C-7, and the Titan III B/NUS. The Observatory must also be compatible with the Space Shuttle for retrieval. With simple adaptation the Observatory shall permit launch and servicing by the Shuttle.

3.1.2.2.3.2 Communication Elements

The Observatory shall be capable of interfacing with the TDRS for relay of Instrument Data and Observatory status telemetry to a control/receiving station. It shall also interface to the Project Control Center via direct transmission to STDN ground stations. The Observatory Instrument Data shall also be transmitted to Low Cost Ground Stations for local users.

3.1.3 PROGRAM COSTS

Costs shall be considered as a major design requirement for the EOS program. Cost targets shall be established for the total EOS program, and for each EOS program element.

Individual performance requirements defined in this specification, lower level performance requirements and management requirements may be traded, within over all EOS System performance requirements, to achieve the specified element cost targets.

The specific cost targets for the Observatory and Central Data Processing Elements of the EOS "A" system defined in this specification are as follows:

EOS ELEMENT	BASIC SPACECRAFT		LRM "A" MISSION	
	NON REC	REC	NON REC	REC
Basic Spacecraft	17.5 M	5.5 M	20.5 M	7.2 M
Spacecraft LRM "A" Mission Peculiars. (R&D Miss.)	----	----	3.8 M	1.2 M
Spacecraft LRM "A" Mission Peculiars. (Operational Miss.)	----	----	3.0 M	2.1 M
TOTAL EOS "A" Observatory Cost	----	----	27.3 M	10.5 M
Central Data Processing (R&d Miss.)	----	----	10.0 M	----
Central Data Processing (Operational Miss.)	----	----	10.2 M	----

Note: Launch costs, etc. are not included.

3.1.4 MISSIONS

3.1.4.1 LAND RESOURCES MANAGEMENT (LRM) MISSION A

The Observatory and Ground system shall support the LRM mission.

3.1.4.1.1 Mission Objectives

Develop instruments, data processing and other spacecraft systems to acquire spectral measurements and images suitable for generating thematic maps of the earth's surface.

Operate these systems to generate a data base from which land use information such as crop or timber acreages or volumes, courses and amounts of actual or potential water run-off and the nature and extent of stresses on the environment will be extracted.

Demonstrate the application of this extracted information to the management of resources such as food and water, the assessment and prediction of hazards such as floods, and planning and regulation of land use such as strip mining and urbanization.

3.1.4.1.2 Mission Description

The basic requirement of the LRM instruments is repeating earth coverage under nearly constant observation conditions. This requires a circular sun synchronous orbit with an integral number of orbits and days per repeat ground trace pattern. A solar orbit of 98° inclination with an orbital altitude of 365 to 385 n mi and descending node time of day ranging from 9:30 a.m. to 11:30 a.m., meets these requirements.

3.1.4.1.3 Instruments

Instrument data shall be transmitted from the observatory via RF links to ground stations and forwarded to the data processing facility for processing. Data products, both photographic and computer compatible will be produced for transmittal to the User community.

The following instruments are planned for the LRM mission A, 5-Band Multi Spectral Scanner and Thematic Mapper.

3.1.4.2 FOLLOW-ON MISSIONS

The Observatory System shall be capable of accommodating follow-on missions.

3.1.4.2.1 Land Resources Mission B

These satellites are the operational version of LRM A. The instruments to be installed are the Thematic Mapper (TM) and the High Resolution Pointable Imager (HRPI).

3.1.4.2.2 Land Resources Mission C

This Observatory is another operational version of LRM A. It will provide data for the evaluation of marine and water resources and pollution by utilizing two TM's, a HRPI and Synthetic Aperture Radar (SAR).

3.1.4.2.3 SEASAT A

3.1.4.2.3.1 Mission Objectives

The SEASAT-A mission is designed for development and demonstration of space techniques for forecasting and monitoring sea state currents, circulation, pileup, storm surges, tsunamis, air/sea interactions, surface winds, and ice formations.

3.1.4.2.3.2 Mission Description

Nominal circular orbital altitude of 391 n mi (725 km) at an inclination of 82°.

3.1.4.2.3.3 Instruments

Instruments planned for this mission are: active and passive microwave facilities and an infrared/visible imager.

3.1.4.2.4 Solar Maximum Mission (SMM)

3.1.4.2.4.1 Mission Objectives

The SMM is a low earth orbit solar pointing satellite designed for solar observations during the period of maximum solar activity. Its general mission objective is to make solar observations in all areas of the spectrum from IR to gamma rays and obtain data to supplement data acquired during the SKYLAB/ATM mission. The SMM will serve specific applications in the fields of: solar flares, flare associated X and gamma radiation as well as high energy particles, solar interior to corona energy transfer, solar and stellar evolution.

3.1.4.2.4.2 Mission Description

Initial launch is scheduled on a Delta launch vehicle. Subsequent retrieval and re-deployment is planned for Shuttle. Minimum orbital life is one year. The nominal orbit is 275-300 n mi circular at an inclination of $28\text{-}33^{\circ}$.

3.1.4.2.4.3 Instruments

The instrument payload of SMM is made up of X-ray and UV Spectrometers, Spectroheliographs (images), Spectrographs, and a Coronagraph.

3.1.4.2.5 Synchronous Earth Observatory Satellite (SEOS) Mission

3.1.4.2.5.1 Mission Objectives

The SEOS mission is intended to investigate remote sensing techniques for measuring transient environmental phenomena from a geosynchronous orbit.

3.1.4.2.5.2 Mission Description

Mission altitude will be 19,323 n mi circular at an inclination of 0°. Nominal orbit positioning will be 96° West longitude and mission duration is to be 2 years. Recovery and/or on-orbit servicing is not planned.

3.1.4.2.5.3 Instruments

Prime instrument for this mission is the Large Earth Survey Telescope (LEST). Other instruments being considered are: Advanced Atmosphere Sounder and Imaging Radiometer (AASIR), Microwave Sounder, Data Collection System and Framing Camera.

3.1.4.2.6 TIROS O Mission

3.1.4.2.6.1 Mission Objectives

The TIROS O vehicle is intended to verify for operational use an advanced environmental operation payload. This spacecraft will have implemented operational versions of remote sensing techniques proven in Nimbus and LRM flight experiments as well as improvements in those sensors carried by the previous N/ITOS vehicles. The TIROS O satellite will be designed so that in-orbit refurbishment of the payload can be effected and evaluated.

3.1.4.2.6.2 Mission Description

Nominal altitude of 910 n mi at an inclination of 103°.

3.1.4.2.6.3 Instruments

The instruments planned for this mission are: High Resolution Radiometer, Advanced Tiros Operational Vertical Sounder, Scanning Multi-Channel Microwave Radiometer, Microwave Radiometer/Scatterometer, Cloud Physics Radiometer, Space Environmental Monitor and Date Collection System.

3.1.4.2.7 Explorer Gamma Ray Experiment Telescope (EGRET) Mission

3.1.4.2.7.1 Mission Objective

The purpose of the EGRET mission is to reveal the dynamic, high energy (i.e., non-thermal) process in our galaxy and in the universe.

3.1.4.2.7.2 Mission Description

The spacecraft will be launched into a 250 n mi circular orbit with an inclination of 28°.

3.1.4.2.7.3 Instrument

The only instrument presently planned for the mission is the Explorer Gamma Ray Experiment Telescope.

3.1.5 SYSTEMS DIAGRAMS

3.1.5.1 OBSERVATORY SYSTEMS DIAGRAM

Figure 3-5 contains the Observatory Systems Diagram.

3.1.5.2 SUPPORT EQUIPMENT SYSTEM DIAGRAM

A System Level Functional Flow Diagram for Support Equipment is shown in Fig. 3-6.

3.1.6 OBSERVATORY INTERFACE DEFINITION

Detailed interface requirements shall be as documented in the following ICD's (anticipated documents):

- 314-ICD-001 Basic Spacecraft/Mission Peculiar
- 314-ICD-002 Instrument Mission Peculiar Equipment/Instruments
- 314-ICD-003 Observatory Communication/STDN/TDRSS
- 314-ICD-004 Wide Band Instrument Data/Primary Ground Station/TDRSS
- 314-ICD-005 Medium Band Instrument Data/Low Cost Ground Station/TDRSS
- 314-ICD-006 Observatory/Launch Vehicle
- 314-ICD-007 Observatory/Support Systems

3.1.6.1 INTERFACES WITHIN THE OBSERVATORY

The functional interfaces between the Basic Spacecraft and the instruments and mission peculiar equipment consisting of power, commands, telemetry, and timing shall be as shown in Fig. 3-5. The quantitative definition of these interfaces shall be as described in 314-ICD-001 and 002.

3.1.6.2 INTERFACES WITH OTHER SEGMENTS

3.1.6.2.1 Launch Vehicle and Fairing

The EOS mission A shall be launched by a Delta 2910 Launch Vehicle protected by a standard 96-inch outside diameter MDAC Payload Fairing. The S/C shall be designed to accommodate follow-on mission launches by the Weight Constrained Titan, the Titan III B/NUS, Titan III C7 and the Space Shuttle, by utilization of specifically designed conversion kits. The LMSC P-123 Type Payload Fairing shall be used when the spacecraft is Tital Launched. LRM mission A shall use the Space Shuttle for retrieval operations. Figures 3-7 and 3-8 illustrate the Delta and Titan payload fairing envelopes.

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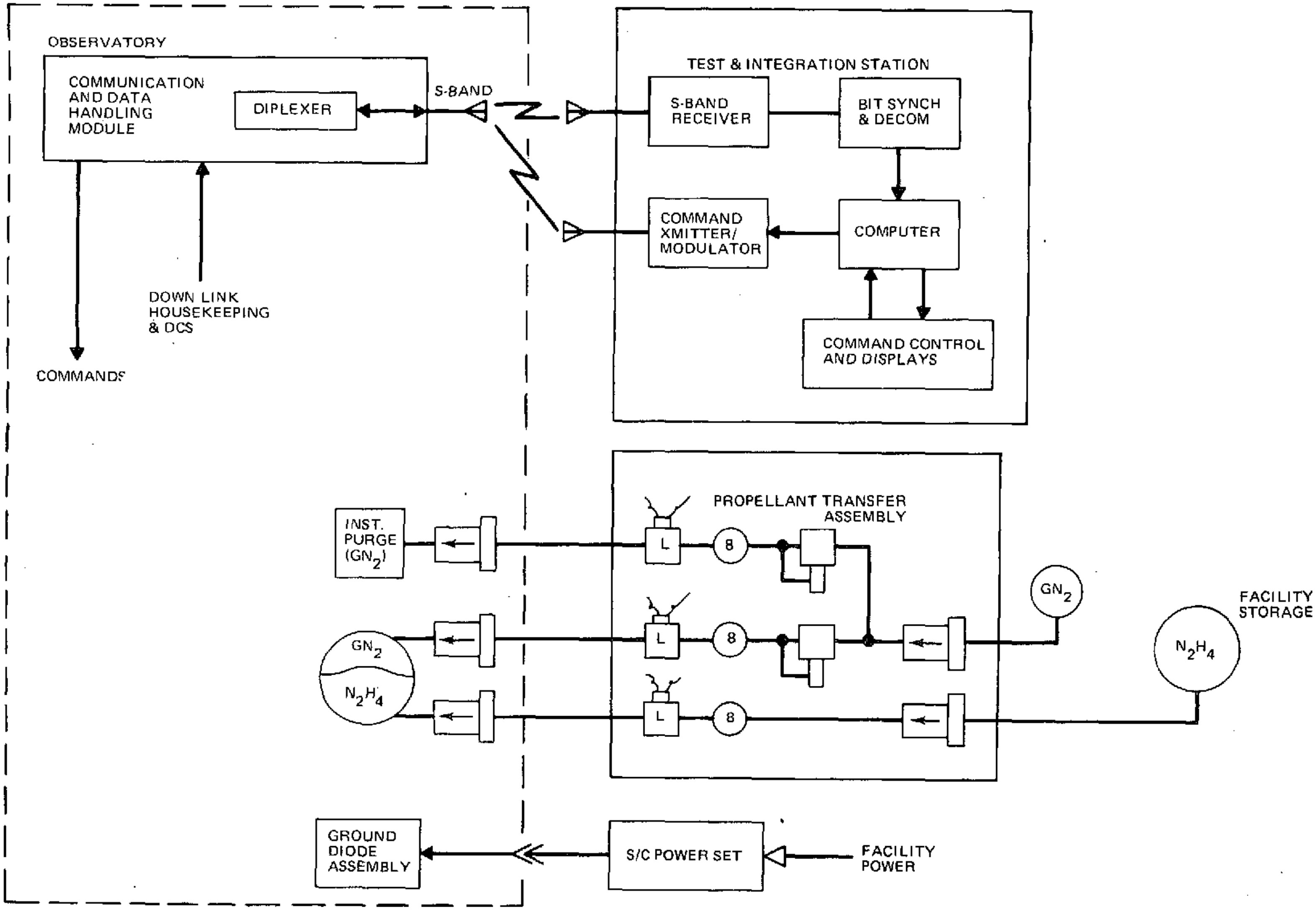

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Fig. 3-6 Support System Functional Level Diagram

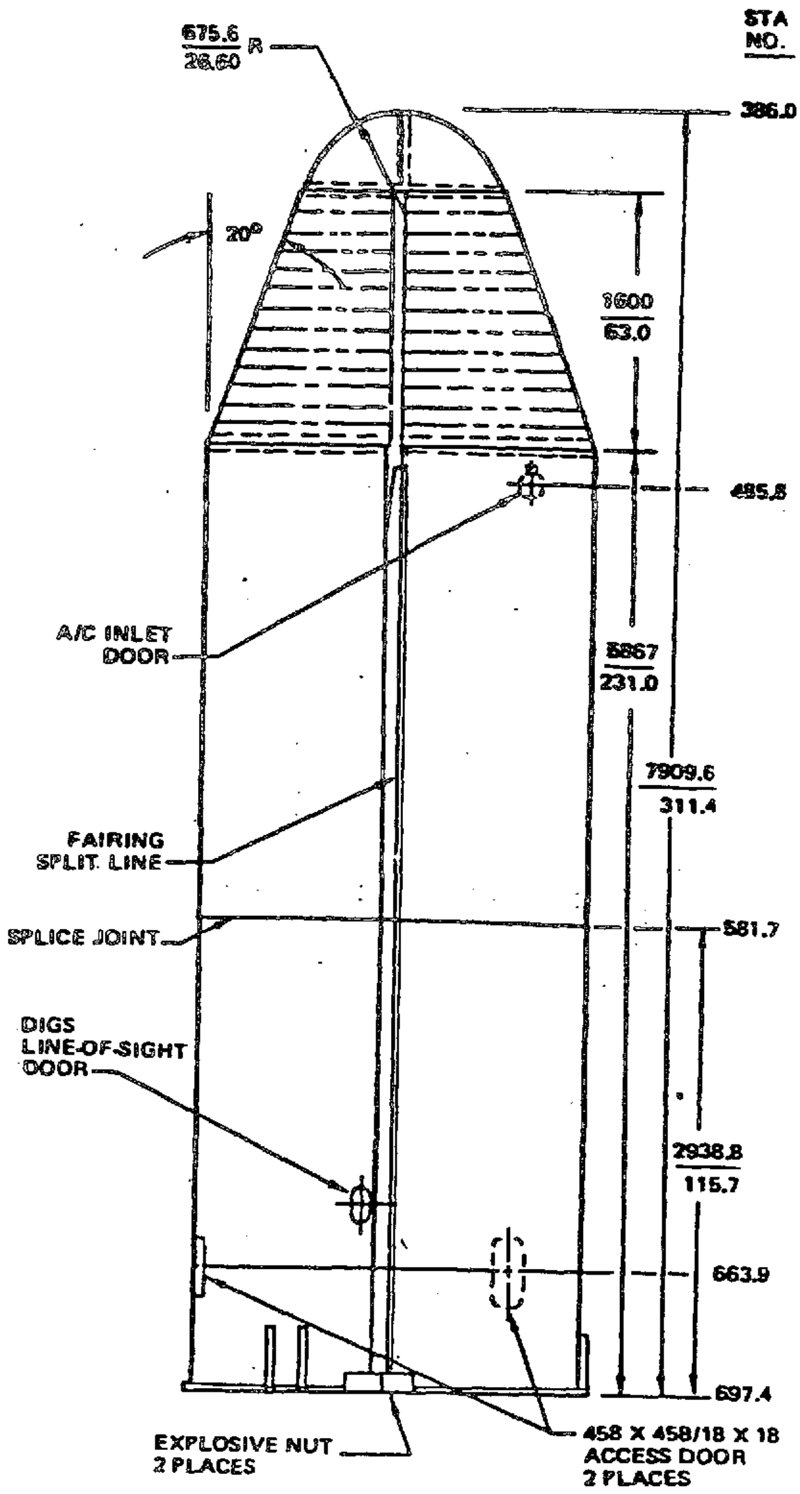

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Fig. 3-7 Delta 96-Inch Fairing Profile

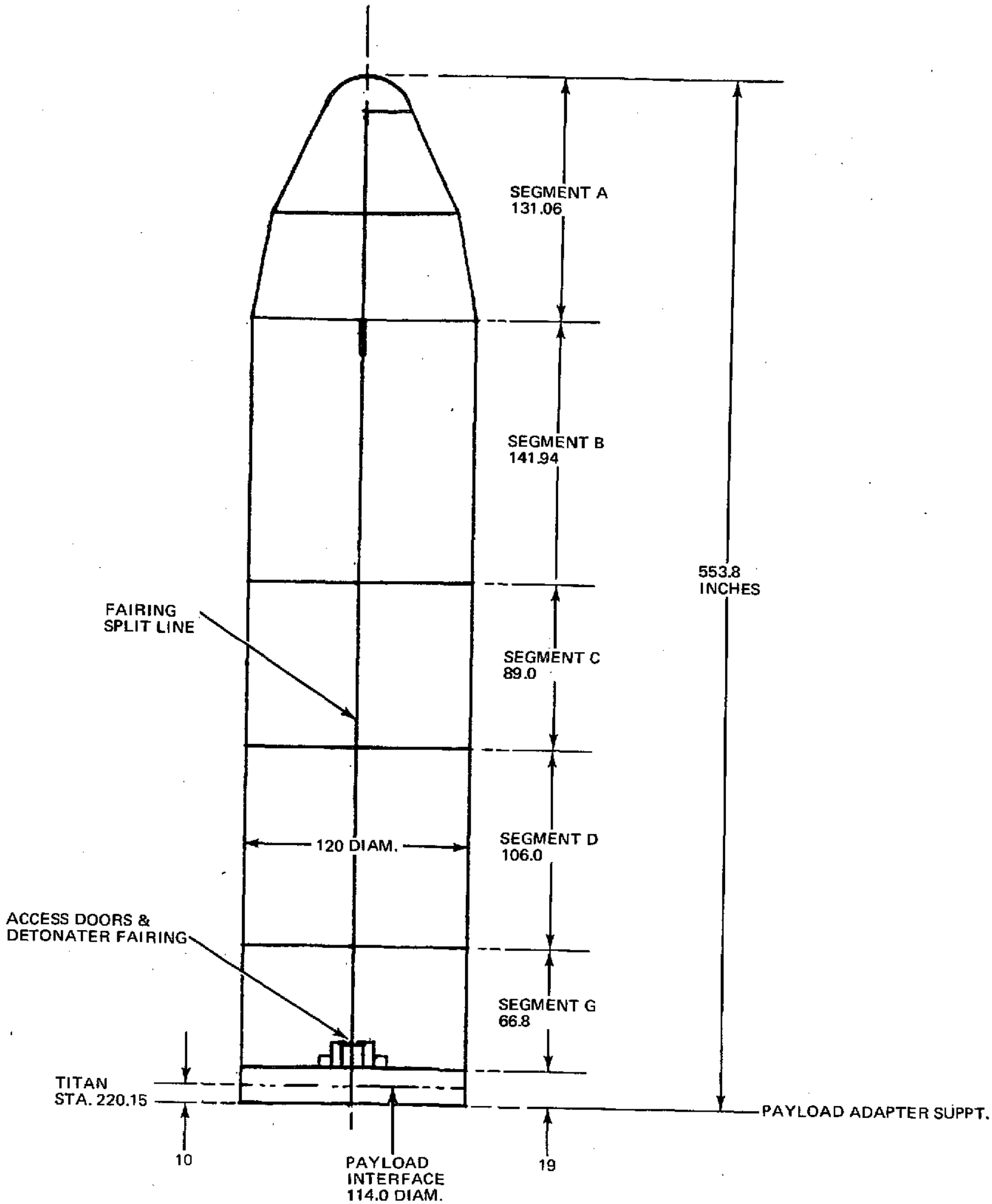

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Fig. 3-8 Titan P.123 Fairing Profile

3.1.6.2.2 Launch Support Systems

The Launch Support Systems shall be designed for use at the Western Test Range (WTR) facilities to receive, stack, checkout, service and launch the Observatory. These facilities include a launch umbilical and service structure which will require support equipment to operate at ground and EOS levels during final checkout and servicing. (An initial list of equipment required at the launch site as part of the Launch Support System is provided in Table 3-17 under prelaunch operations and launch.)

The interface between the Launch Support System and the Observatory shall be defined by an Interface Control Document which will contain the final list of equipment.

Figure 3-6 defines the system level functional test configuration for both pre- and pad-launch checkout. Note that this configuration is identical to the functional test performed prior to Observatory delivery to the launch site, thus assuring compatibility with, and verification of, previous tests.

3.1.6.2.3 STDN Tracking, Command and Telemetry

(a) TDRSS

The S-band capability of the TDRSS is capable of performing the tracking telemetry and command functions for the observatory. Interfaces (via the NASCOM network) between the Project Control Center (PCC) and the TDRS ground station act effectively as a STDN site. Command and telemetry data can thus be relayed between the PCC and the observatory. Tracking of the observatory can be accomplished from the TDRS ground station and this data will then be routed to the GSFC orbit determination facility for observatory ephemeris generation.

(b) STDN

The first link between the Observatory and the PCC is the STDN site supporting the real time operations. There will be a total of three STDN sites used by the Observatory. These are:

Goldstone, California	(GDS)
Engineering Test Center, Greenbelt, Md.	(ETC)
Fairbanks, Alaska	(ULA)

The operational details of these sites are defined in the STDN User's Guide Baseline Document. STDN No. 101.1. May 1974. Revision B.

The second link between the Observatory and the PCC is the NASCOM network. These support requirements will be within the capabilities defined in the Data System Development Plan. NASCOM Network. (Revision 9). F7 74-1.

3.1.7 GOVERNMENT FURNISHED PROPERTY LIST

No government property has been identified in support of the EOS program.

3.1.8 OPERATIONAL & ORGANIZATIONAL CONCEPTS

3.1.8.1 OPERATIONAL CONCEPT

The EOS Program is comprised of the Observatory, the Project Control Center, the STDN and TDRS Communication links, the Low Cost Ground Stations, the Central Data Processing Facility, and the Launch Vehicle.

The Observatory will be launched from the WTR and inserted into a circular polar orbit by the launch vehicle. The Observatory will be stabilized and configured for survivability during activation of subsystems. Next, the subsystems and instruments will be checked out and verified operational. The Observatory will maintain earth pointing attitude in a sun synchronous orbit. The instruments will record data and transmit it directly to the ground or via TDRS, or store it for later transmission. After complete system verification, MSS data shall be given full operational status. TM data may be used to enhance or back-up MSS data in addition to providing R&D data. During normal mission operations the Observatory orbit will be adjusted to compensate for orbital decay. The Observatory will be compatible with the Shuttle for possible later retrieval.

3.1.8.2 ORGANIZATIONAL CONCEPT

3.1.8.2.1 Observatory Element

The Observatory Element shall provide a suitable RF environment for the Ground Element to control, interrogate, and to receive data from the Observatory.

3.1.8.2.2 Ground Element

3.1.8.2.2.1 Control System Element

The Control System Element will track the Observatory and determine ephemeris. It will determine when the Observatory orbit must be adjusted and command the delta velocity required. The Control System Element shall program the area of earth to be scanned by the instruments and command data dumps as required.

3.1.8.2.2.2 Central Data Processing Facility

The CDPF will be implemented over a period of time using the phased approach. The initial facility will be a limited capability system with flexibility to permit changes to be incorporated as NASA and the user community gain experience with the application of digital imaging. The CDPF will be a full capability system; however, the flexibility will be limited for the purpose of obtaining a minimum cost system. It is anticipated that the initial facility will principally be implemented using a configuration of general purpose mini-computers and that the full facility will be implemented using principally special-purpose hardware.

3.1.8.2.2.3 Local User Systems/Low Cost Ground Stations

The users that operate Low Cost Ground Stations will receive data directly from the Observatory and process their own data.

3.2 CHARACTERISTICS

3.2.1 PERFORMANCE

3.2.1.1 OBSERVATORY PERFORMANCE CHARACTERISTICS

3.2.1.1.1 Mission Orbit

The Observatory shall be placed in a sun-synchronous orbit at an altitude in the range of 365 n mi to 385 n mi. The initial right ascension of the line of nodes shall be selected to yield an orbit time of day in the range of 9:30 a.m. to 11:30 a.m.

The errors at insertion shall not exceed the three sigma values presented below:

Injection Velocity Deviation:	± 22.5 fps
Flight Path	
- Pitch:	± 0.04 deg
- Yaw:	± 0.04 deg
Altitude:	± 14.0 n mi
Inclination:	± 0.04 deg

The initial orbit shall be trimmed to meet the requirements of Paragraph 3.2.1.1.4.

3.2.1.1.2 Mission Duration

The Observatory shall be capable of operating as defined herein for a minimum of two years following injection into the nominal orbit described in Paragraph 3.2.1.1.1.

Spacecraft survival life shall be five years and survival consumables shall be sized for five years.

3.2.1.1.3 Mapping Coverage

The Observatory shall periodically observe the same geographic locations under the same or very similar lighting conditions established by the final orbit selected from the range specified in Paragraph 3.2.1.1.1. The swath width overlap at the equator shall be between 10 n mi and 20 n mi to facilitate the matching of adjacent imagery.

The orbit repeat cycle (N), the number of orbits per repeat cycle (n), and the number of days delay (p) until the closest overlapping swath is generated shall be selected to conform with the selection of swath width and orbit altitude. The parametric relationship of repeat cycle, swath width, and orbit altitude may be found in Fig. 3-9 and 3-10.

3.2.1.1.4 Mission Orbit Tolerances

Orbit corrections shall be made to the Observatory to maintain swath overlap at the equator to 20 n mi or less.

3.2.1.1.5 Positional Accuracy

The Observatory shall be capable of earth observation with a Thematic Mapper and Multispectral Scanner within the one sigma coordinate accuracies noted below, and allocated in Table 3-1:

- ± 450 meters - assuming data correction by the central processing facility for correction of earth-rotation, line length adjustment, earth curvature correction and two day emphemeris prediction.
- ± 170 meters - assuming data correction by the central processing facility noted above plus the use of best fit ephemeris measured data.
- ± 15 meters - assuming the use of ground control points and the data corrections noted above (TM only).

Table 3-1
System Ground Position Accuracy Allocation* (Meters-1 σ)

System Tolerance		Observatory	Ephemeris Measured	Ephemeris Predicted	Ground Processing	Ground Reference
Required	Budget					
±450	393	146	76	357	10	—
±170	165	146	76	—	10	—
± 15	15	5	—	—	10	10

*Note: Initial allocation. Allocation are subject to system tradeoff.

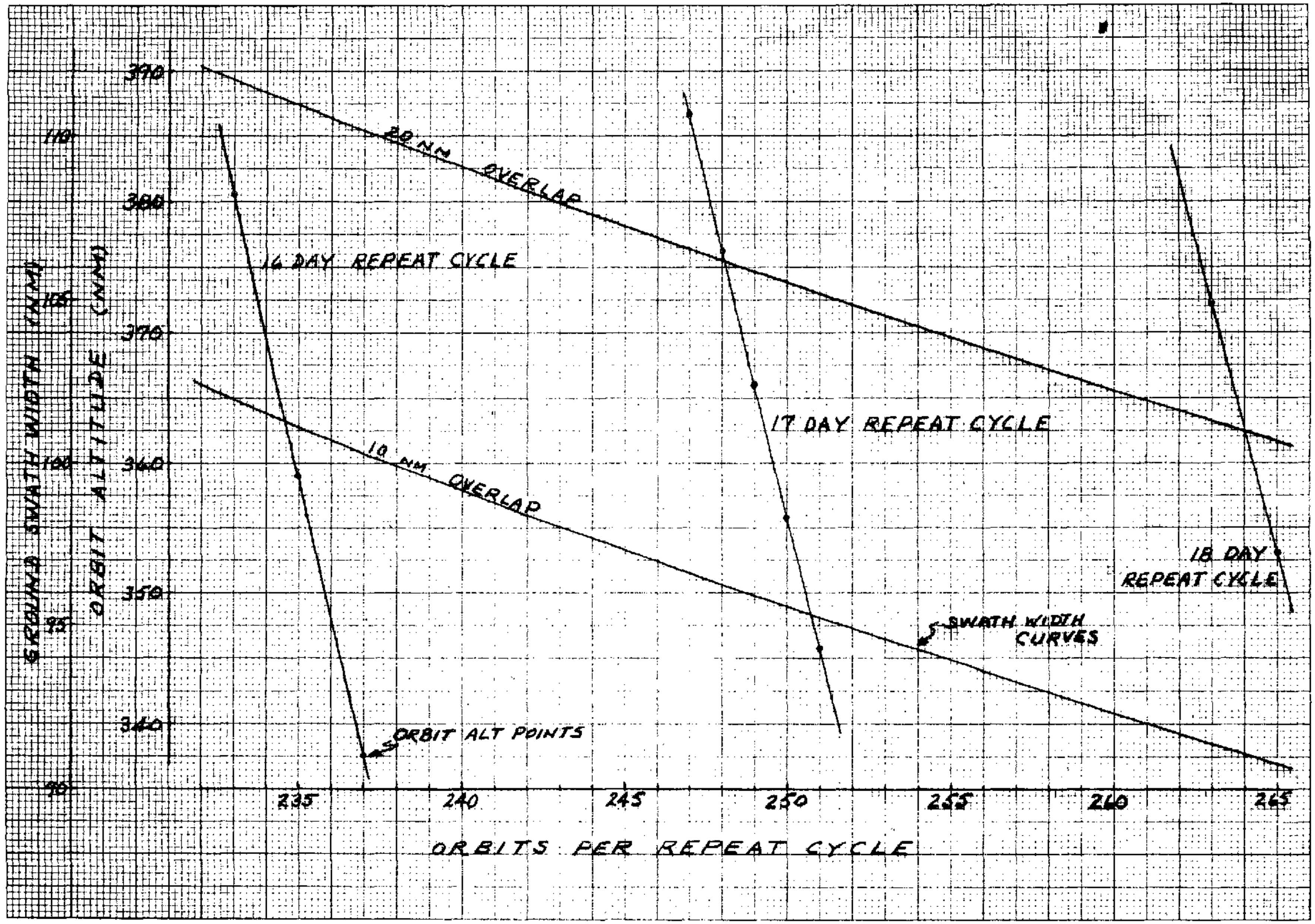
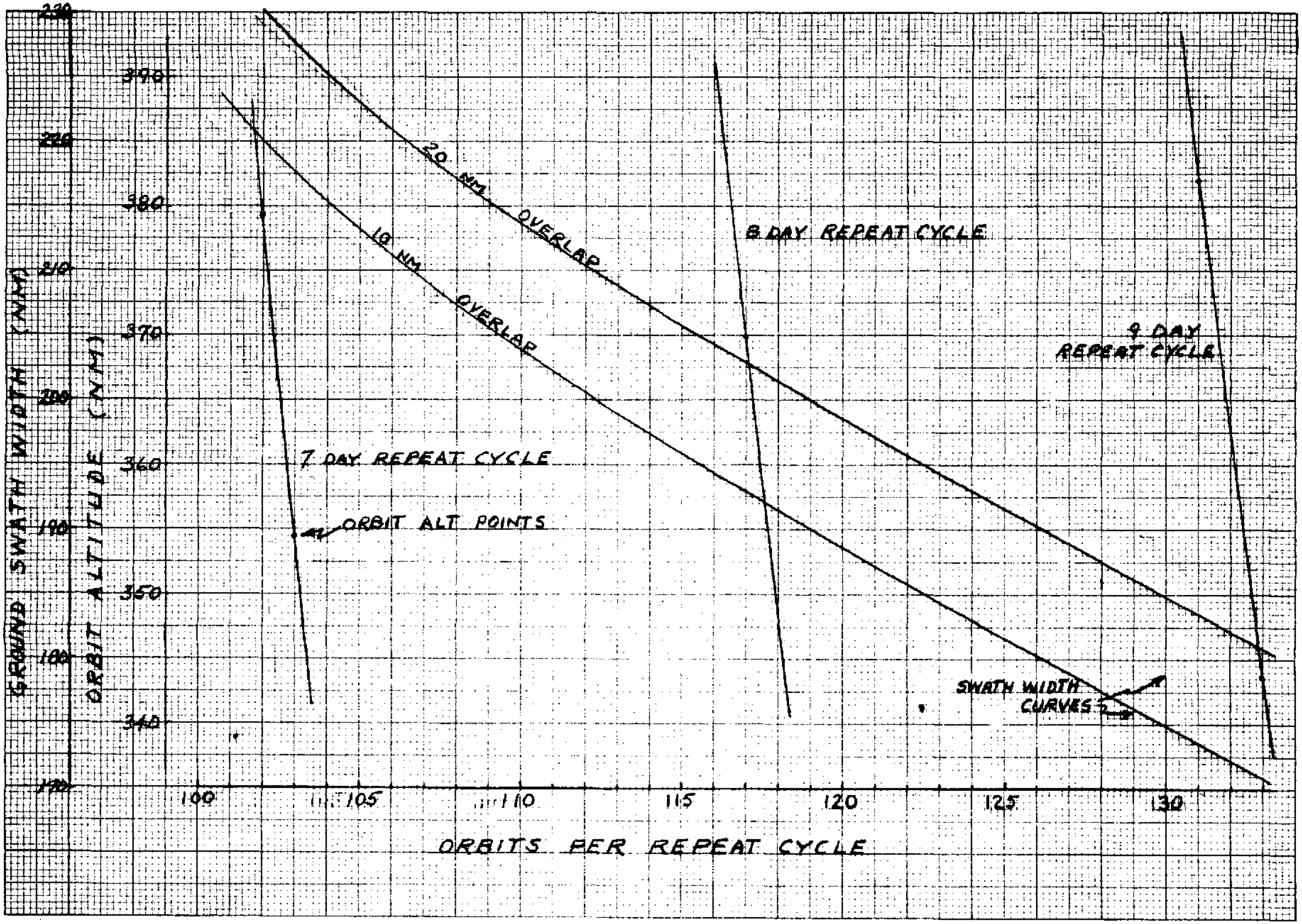


Fig. 3-9 Instrument Ground Swath Vs Repeat Cycle



(1) 5-23

Fig. 3-10 Instrument Ground Swath Vs Repeat Cycle

3.2.1.1.6 Radiometric Accuracy

The Observatory shall be capable of earth observation with the Thematic Mapper to the following relative radiometric accuracy (including central processing facility).

- Visual Spectral Bands (Large Area):

  ± 1.6% - Tape
  ± 5% - Film

- IR Spectral Band (Large Area):

  ± 1^{o}K - Tape
  ± 3^{o}K - Film

The Observatory shall be capable of earth observation with the Multispectral Scanner with the following relative radiometric accuracy.

- Visual Spectral Bands (Large Area)

  ± 2.5% - Tape
  ± 6% - Film

- IR Band (Large Area)

  ± 1^{o}K - Tape
  ± 3^{o}K - Film

3.2.1.1.7 Instrument Performance

Requirements for the performance of the Thematic Mapper and the Multispectral Scanner instruments are presented in Paragraph 3.7.2

3.2.1.2 SUPPORT EQUIPMENT PERFORMANCE CHARACTERISTICS

The support equipment shall be used to perform the functions required to: test, adjust, calibrate, appraise, gage, measure, assemble, disassemble, handle, transport, safeguard, store, actuate, service, repair, overhaul and maintain the S/C during all factory-to-launch operations, including checkout and launch site. As a minimum these functions shall include:

- Balance and determine the required mass properties of the EOS in its assembled configuration
- Monitor and evaluate performance of the S/C and its components and subsystems during all phases of testing

- Evaluate the end-to-end performance during functional, environmental and integrated system development, qualification, and acceptance tests of the EOS and its subsystems

- Verify static and dynamic mechanical interfaces

- Verify steady state and transient electrical interfaces between the Observatory and Launch Vehicle

- Simulate pyro interface and monitor and display responses and other pertinent parameters associated with pyro circuits

- Simulate those signals during functional and integrated system tests that would normally be transmitted from the ground to the Observatory and the Observatory to the ground

- Provide tracking, telemetry and command simulation to demonstrate all functional performance requirements at the acceptance level pertaining to the C and DH subsystem of the Observatory

- Perform end-to-end testing of the integrated S/C in its launch configuration during system checkout testing

- Perform servicing, environmental conditioning, electrical power, and grounding and mechanical support for S/C during all ground operations

- Provide the mechanical and electrical equipment necessary for S/C deployment, test, assembly, shipping, integration, and checkout at the factory and launch site

- Perform loading and detanking of propellants and pressurants for the S/C

In addition, support equipment shall be provided which can perform the above functions as applicable to assembly, integration and checkout of the S/C, S/S modules as inidividual units.

3.2.2 PHYSICAL CHARACTERISTICS

3.2.2.1 OBSERVATORY PHYSICAL CHARACTERISTICS

The S/C LRM Mission A Configuration is shown in Fig. 3-11. The coordinate system designating +Z as nadir and +X as flight path directions is shown in Fig. 3-18. The spacecraft-to-launch vehicle adapter shall be as described in Paragraph 3.7.1.4.3.5. There shall be no electrical interface between the S/C and the Delta/Titan Launch Vehicle. Existing provisions in the payload fairing shall be utilized for on-pad air conditioning and umbilical penetration to the maximum extent possible. Additional access doors and RF transparent panels as required may be installed in the fairing by coordination with the Delta Project. Size and location of such additions will be detailed in ICD (TBD).

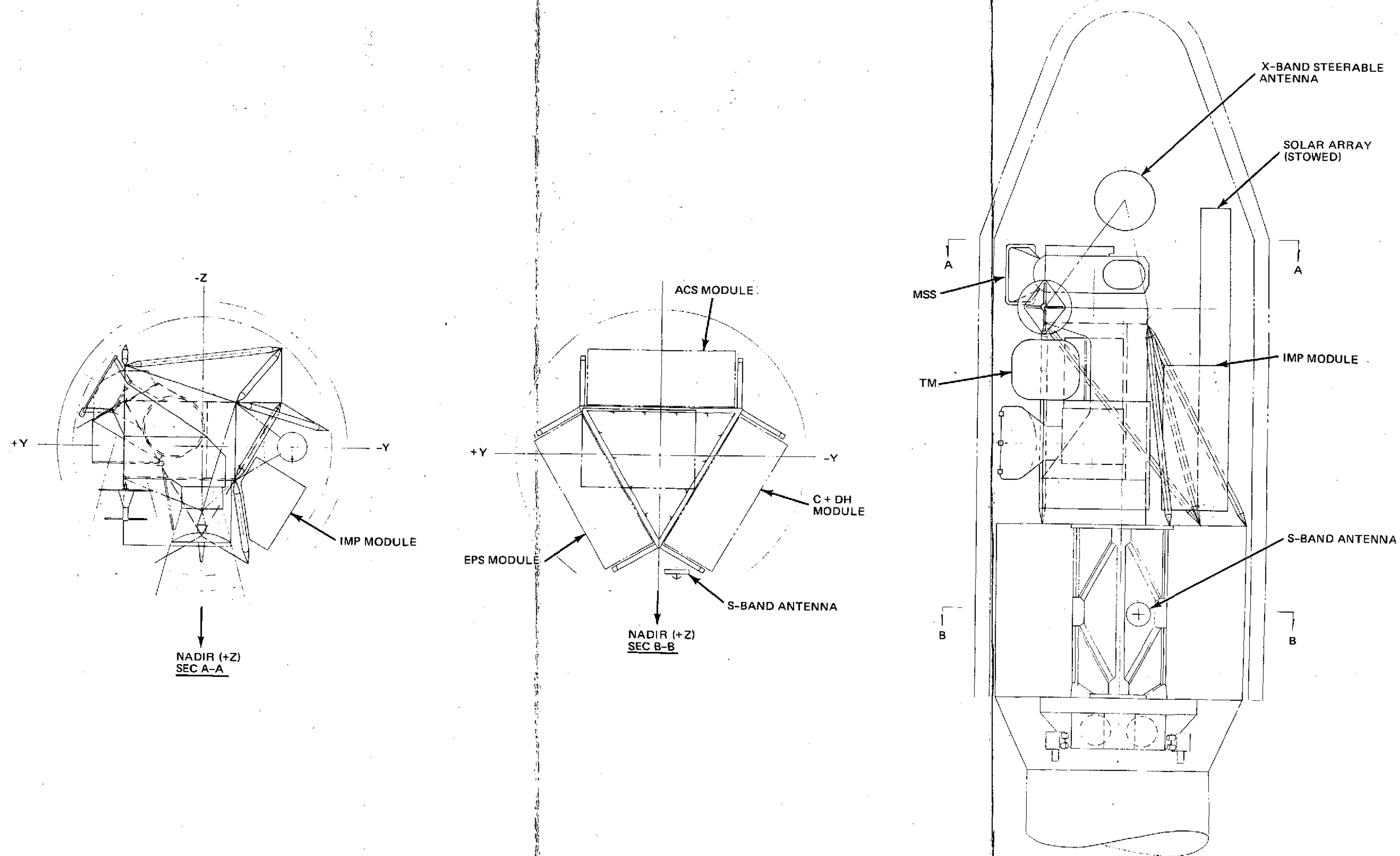


Fig. 3-11 EOS-A Configuration Delta L/V

3.2.2.1.1 Mass Properties

The limiting mass properties of the space vehicle and the subsystem modules shall be as shown in Table 3-2.

Table 3-2 Mass Properties LRM-A Mission

Space Vehicle	
Weight	2630 lb
Center of Gravity about Geometric *	6 inch Radius
Products of Inerta (P_{max}/I_{max})	3%
Subsystem Modules**	
Electrical Power	241 lb
Attitude Control	226 lb
Comm & Data Handling	209 lb
Orbit Adjust/RCS	100 lb

* Geometric centerline is defined as a line Perpendicular to the plane of the space vehicle attachment points, and centered with respect to them

** Complete module, including thermal control & structure

3.2.2.2 SUPPORT EQUIPMENT PHYSICAL CHARACTERISTICS

The physical characteristics of the support equipment shall be such as to provide ready access to interior parts, terminals and wiring, and for adjustments, calibration, complete circuit checking, and removal and replacement of parts. Doors or access plates shall be provided as necessary for access to controls, instruments, servicing provisions, and items requiring frequent maintenance. Fastening devices shall incorporate suitable locking means to prevent their working loose in service. In addition the following also applies:

3.2.2.2.1 Transport Equipment Controls and Displays

The Support Equipment enclosure for new end items of GSE shall be designed to facilitate packaging, shipping and storing of the equipment wherever possible. GSE that, due to weight and size, cannot be readily handled by two men during shipment shall have provisions for lifting by material handling equipment.

3.2.2.2.2 Support Equipment Controls and Displays

Controls shall be readily accessible, suitably arranged, and of such size and construction as to permit convenience and ease of operation under all service conditions. The setting, position, or adjustment of the controls shall not be affected by vibration, shock, or other service conditions. Controls shall operate freely, smoothly and without excessive binding, play, or backlash. Knobs and handles shall have high-impact strength and shall be firmly secured to their control shafts. The divisions and lettering on turning dials shall be suitably etched or printed with characters large enough to read under normal lighting conditions. The controls and displays shall be in conformance with MIL-T-21200L, Paragraph 3.1.5, and MIL-STD-454D, Requirements 28 and 42.

3.2.2.2.3 Mechanical Stops

Provision for mechanical stops shall be included for adjustable parts. The stops shall be sufficiently rugged to prevent damage to the mechanism.

3.2.2.2.4 Safety

- Interlocking Devices - Interlock devices shall be provided for protecting operating and maintenance personnel from injury by exposed voltages over 30 volts when servicing or adjusting any portion of the equipment
- Fire Hazards - Support equipment shall be designed to the extent possible with nonflammable materials and adequate protection devices such as fuses or circuit breakers.

3.2.3 RELIABILITY

3.2.3.1 QUANTITATIVE REQUIREMENTS

The on-orbit design life of the Observatory shall not be less than two years Mean Mission Duration (MMD) where the MMD shall be defined by the following equation for a survival life (T_L) of five years:

$$MMD = \int_0^{t_L} R\ (t)\ dt$$

R (t) is the value of the on-orbit observatory reliability function at time t.

3.2.3.2 RELIABILITY/MAINTAINABILITY PROGRAM

A reliability/maintainability program shall be conducted in accordance with the requirements of NASA specification NHB 5300.4 (1A).

3.2.4 MAINTAINABILITY: GROUND REFURBISHMENT

The Observatory shall be capable of being completely refurbished on the ground to the component level, within a period of six months.

3.2.5 SYSTEM EFFECTIVENESS

The EOS observatory and ground system design shall be evaluated using a multiparameter Figure of Merit representing System Effectiveness. System Effectiveness is an evaluation of a particular system design that measures its success at achieving program and mission objectives and requirements. The System Effectiveness shall combine performance capabilities of the system with the reliability, maintainability, and availability characteristics of the system to produce an overall assessment of the excellence with which the system is expected to achieve the program and mission objectives. Application of system effectiveness indicates the change in effectiveness associated with changes in design values or changes in mission objectives.

Effectiveness shall be measured by the expected number of equivalent scenes broadcast by the observatory segment during the program life. A larger number indicates greater effectiveness.

The effectiveness calculation for a satellite shall consider:

(1) For each instrument:

- Quality of instrument output (relative to TM)
- Quantity of instrument output (number of bits per day/bit in a TM scene X number of days in program = number of equivalent scenes)
- Availability of instrument output (uptime fraction over program life depends on the S/C subsystem support received, quality of support, and availability of support).

(2) For each subsystem not included in direct support of the instruments:

- Quality of support (per cent of objectives achieved)
- Availability of support (uptime fraction)

(3) Effectiveness of launch vehicle

- Reliability of launch
- Probability of spacecraft survival of launch environment

(4) Effectiveness of orbit - Quality of orbit (fractional score of chosen orbit at meeting orbit objectives)

The effectiveness of all instruments is the sum of the effectiveness of each instrument. The effectiveness of an instrument is the product of the quality, quantity and availability of the data gathered by that instrument.

The effectiveness of the launch vehicle is the product of the probability of launch success and the probability of S/C survival of the launch environment.

The effectiveness of the orbit is the weighted average score of the orbit parameters at meeting the orbit requirements.

The effectiveness of the satellite is the product of: (1) effectiveness of instruments; (2) effectiveness of S/C subsystems; (3) effectiveness of launch vehicle, and (4) effectiveness of orbit.

3.2.6 ENVIRONMENTAL CONDITIONS

3.2.6.1 OBSERVATORY ENVIRONMENTAL CONDITIONS

3.2.6.1.1 Transportation, Handling, and Storage Environment

The Observatory shall be capable of operating within specification limits after exposure to all of the following natural and induced environments, while in a non-operating condition, experienced during fabrication, storage, handling, transportation and erection at the launch site. Controlled environments shall be provided when necessary to bring the experienced natural and induced environments to levels less severe than those pertaining to launch, ascent and orbital mission phases. Major structural elements, except necessary lift/rotational hard points shall not be designed by these criteria.

3.2.6.1.1.1 Packaged Natural Environments

Ambient environments external to the shipping unit.

3.2.6.1.1.1.1 Altitude-Air Transport

The maximum range shall be from sea level to 15,000 meters.

3.2.6.1.1.1.2 Temperature

Surrounding air temperatures.

3.2.6.1.1.1.2.1 Air Transportation

$-40^{o}C$ to $+66^{o}C$

3.2.6.1.1.1.2.2 Truck Transportation

$-40^{o}C$ to $+66^{o}C$

3.2.6.1.1.1.2.3 Storage

$-3.9^{o}C$ to $+41.7^{o}C$

3.2.6.1.1.1.3 Solar Radiation

Equivalent to that specified in MIL-STD-810, Method 505.

3.2.6.1.1.1.4 Rain

Equivalent to that specified in MIL-STD-810, Method 506.

3.2.6.1.1.1.5 Humidity

Equivalent to that specified in MIL-STD-810, Method IV.

3.2.6.1.1.1.6 Fungi

Equivalent to that specified in MIL-STD-810, Method 508.

3.2.6.1.1.1.7 Atmospheric Corrosion

Equivalent to that specified in MIL-STD-810, Method 509.

3.2.6.1.1.1.8 Abrasion

Equivalent to that specified in MIL-STD-810, Method 510

3.2.6.1.1.2 Packaged Induced Environments

3.2.6.1.1.2.1 Sustained Acceleration-Hoisting

For all hoisting operations, a 2.0 g load hoisting capability shall be provided. When hoisting with lift rings, the 2.0 g load shall be applied to any one ring or combination of rings, whichever is critical.

3.2.6.1.1.2.2 Vibration

3.2.6.1.1.2.2.1 Air Transportation

Sinusoidal vibration applied as specified in MIL-STD-810, Method 514.1, Procedure X, to the levels indicated on Fig. 514.1-7, curve "AY". The levels shall be as follows:

Frequency Range Hz	Acceleration zero-to-peak ± 1g
5 - 17	2.54 mm. d.a.
17 - 50	1.5

3.2.6.1.1.2.2.2 Truck Transportation

Sinusoidal vibration applied as specified in MIL-STD-810, Method 514.1, Procedure X, to the levels indicated on Fig. 514.1-7, curve "AW". The levels shall be as follows:

Frequency Range, Hz	Acceleration zero-to-peak, $\pm$ 1g
5-5.5	25.4 mm d.a.
5.5-500	1.5

3.2.6.1.1.2.2.3 Shock

Equivalent to impact into a concrete abutment at a velocity of 2.31 m/sec.

3.2.6.1.2 Pre-Launch Natural Environment

The Observatory shall be capable of operating within specification limits during and after exposure to either (a) the following natural launch-site environment, or (b) a launch-site environment controlled by the Observatory Contractor to levels less severe than those pertaining to launch, ascent, and orbital mission phases.

3.2.6.1.2.1 Temperature

Surface air temperatures from -3.9^{o} to $+41.7^{o}$C and in accordance with NASA TMX-64589, Section II, Paragraph 2.6.

3.2.6.1.2.2 Solar Radiation

Equivalent to direct solar radiation of 1179 W/m^{2}, in accordance with NASA TMX-64589, Section II, Paragraph 2.5 and under the conditions specified in MIL-STD-810, Method 505.

3.2.6.1.2.3 Rain

Equivalent to rainfall of 64 mm/hr for a period of 2 hours, in accordance with NASA TMX-64589, Section IV, Paragraph 4.2.1 and under the conditions specified in MIL-STD-810, Method 506.

3.2.6.1.2.4 Humidity

Relative humidities up to 100 percent in accordance with NASA TMX-64589, Section III, Paragraph 3.2.16 and 3.2.1c and in a chamber as described in MIL-STD-810, Method 507.

3.2.6.1.2.5 Fungi

Equivalent to 28 days exposure to fungi in accordance with NASA TMX-64589, Section XI and under the conditions specified in MIL-STD-810, Method 508.

3.2.6.1.2.6 Atmospheric Corrosion

In accordance with NASA TMS-64589, Section X and under the conditions specified in MIL-STD-810, Method 509.

3.2.6.1.2.7 Abrasion

In accordance with MIL 64589, Section X under the conditions specified in MIL-STD-810, method 509.

3.2.6.1.2.8 Explosive Atmospheres

Explosive atmospheres surrounding non-hermetically sealed equipment as specified in MIL-STD-810, Methods 511, Procedure I.

3.2.6.1.2.9 Particulates

Prior to installation of the Launch Vehicle Fairing, the Observatory particulate contamination under visible and black light shall not exceed Level 300 of MIL-STD-1246.

3.2.6.1.3 Launch and Ascent/Descent Environment

The Observatory shall operate within specification limits during and after exposure to the environment specified below, experienced from start of countdown to separation of the Observatory from the Launch Vehicle. These environments are envelopes of those induced by all the anticipated Launch Vehicles which include the following:

Launch Vehicle	Fairing
Delta 2910	MDAC 2.44 m (96 in.) dia
Delta 3910	MDAC 2.44 m (96 in.) dia
Titan III B/NUS Titan III C-7	*LMSC P-123, 3.05m (120 in.) dia
Shuttle	Payload bay area

* Segments - - - - - A, B, C, D, and G

3.2.6.1.3.1 Acoustic Field

The highest acoustic environment occurs at launch and transonic flight regimens, and is generated by the Launch Vehicle's engines and aerodynamic pressure fluctuations. The maximum expected internal acoustic environment shall be shown in Table 3-3.

Table 3-3 Maximum Expected Flight Acoustic Level (Internal)
*(dB Re: 20μ Newton/m²)

OCTAVE BAND	
CENTER FREQUENCY (HZ)	SOUND PRESSURE LEVEL (DB*)
31.5	127
63	133
125	138
250	140
500	139
1000	137
2000	133.5
4000	131
8000	129
OVERALL	145.5
DURATION: 1 MINUTE	

(1) 5T-25

3.2.6.1.3.2 Sinusoidal Vibration

The sinusoidal vibration environment is an envelope of the Launch Vehicle's responses, at the Observatory/Launch vehicle interface, resulting from excitation of the Launch Vehicle low frequency modes due to various forcing functions (i.e., POGO, engine ignition, engine shutdown and sinusoidal transients occurring throughout the flight.) The maximum expected sinusoidal vibration levels shall be as shown in Table 3-4.

3.2.6.1.3.3 Random Vibration

The Observatory shall be subjected to broadband random vibration during launch and ascent. This excitation is predominantly due to the launch acoustic field, aerodynamic excitation and structure-borne transmitted vibration. The maximum expected structure-borne transmitted random vibration spectrum, at the Observatory/Launch interface, shall be as shown in Table 3-5.

3.2.6.1.3.4 Shock

Shock impulses are transmitted to the Observatory at separation of the Launch Vehicles stages, at engine ignition, at separation of the fairing and expected shock experienced by the Observatory, at the Observatory/Launch vehicle interface, shall be as defined by the shock responses spectrum shown in Fig. 3-12.

Table 3-4 Maximum Expected Flight Sinusoidal Levels

AXIS OF EXCITATION	FREQUENCY RANGE (HZ)	ACCELERATION ZERO-TO-PEAK ± (g)
	5 - 9.5	8.4 MM D.A.
LONGITUDINAL	9.5 - 15	1.5
(X-X)	15 - 21	4.0
	21 - 50	2.0
	50 - 200	1.5
	5 - 7.1	12.7 MM D.A.
LATERAL	7.1 - 22	1.3
(Y-Y) & (Z-Z)	22 - 200	1.0
SWEEP RATE: 4 OCTAVES/MINUTE/AXIS		

NOTES:

1. INPUT AT OBSERVATORY/LAUNCH VEHICLE INTERFACE
2. APPLIED ALONG EACH OF THE THREE ORTHOGONAL AXIS

(1) 5T-26

Table 3-5 Maximum Expected Flight Random Vibration

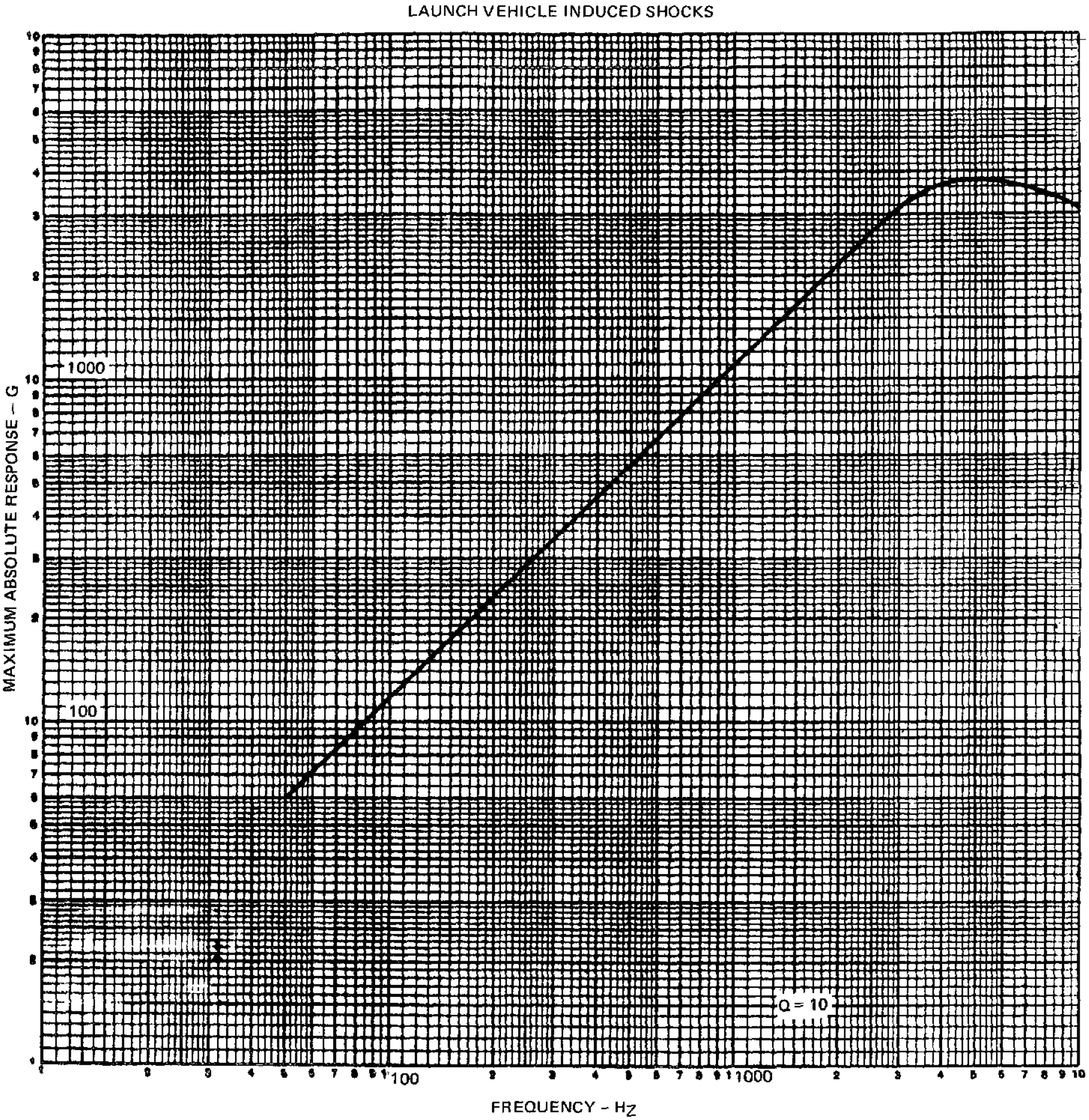
FREQUENCY RANGE (HZ)	ACCELERATION SPECTRAL DENSITY (g^2/HZ)	ACCELERATION OVERALL g-RMS
20- 500	+3DB/OCT	
500-1000	0.07	9.4
1000-2000	-6DB/OCT	
DURATION: 1 MINUTE/AXIS		

NOTES:

1. INPUT AT OBSERVATORY/LAUNCH VEHICLE INTERFACE
2. APPLIED ALONG EACH OF THE THREE ORTHOGONAL AXES

(1) T5-27

NOTE: FLIGHT LEVELS



(1) 5-28

Fig. 3-12 Shock Response Spectrum at Observatory/Launch Vehicle Interface

3.2.6.1.3.5 Sustained Acceleration

The maximum acceleration loads shall be determined from the worst case combined effects of quasi steady acceleration and transient response of the Observatory and Launch Vehicle System due to POGO, ignition and burnout of stages. The preliminary maximum expected limit load factors shall be as shown in Tables 3-6, 3-7, and 3-8 for Delta 2910, Titan III B/NUS and Shuttle respectively. Where the natural frequency of a component installed on its support brackets may couple with POGO, ignition and burn-out transients, the maximum predicted acceleration level shall account for possible amplification. These levels shall be determined from the dynamic load cycle. The Observatory coordinate system is shown in Fig. 3-18, Paragraph 3.7.1.3.4.2.

3.2.6.1.3.6 Pressure

Decreasing atmospheric pressures ranging from sea level conditions down to vacuum conditions associated with orbital altitude, occurring at a rate depending on the flight profile and venting schedule.

3.2.6.1.3.7 Temperature

Temperatures resulting from the following factors:

(a) Aerodynamic Heating - Surfaces which are exposed to the airstream will be subject to frictional heating. Maximum fairing wall temperature shall be specified in EOS-SS-240.

(b) Thermal radiation from aerodynamically heated walls

(c) Free molecule flow heating after fairing separation as specified in EOS-SS-240.

(d) Conductive and radiative heat transfer between equipment and structural members.

(e) Solar Irradiation - Depending upon the season, sunlit surfaces receive a radiation intensity of 1309 to 1400 W/M^2 over an area projected normal to the sun's rays. Tolerance and solar spectrum as specified in NASA SP-8005.

(f) Earth albedo and emitted radiation as specified in NASA SP-8067.

(g) Thermal radiation to free space after fairing separation.

3.2.6.1.4 Orbital Environment

The Observatory shall operate within specification limits during exposure to the following self-induced and natural environments experienced after separation of the Observatory from the Launch Vehicle.

Table 3-6 Limit Load Factors – Delta 2910 and Delta 3910 Launch Vehicle

CONDITION	LONGITUDINAL X	LATERAL Y OR Z
LIFT-OFF	+ 2.9 - 1.0	2.0
MAIN ENGINE CUTOFF	+ 12.3	0.65

T1-9
(1) 5T-30

Table 3-7 Limit Load Factors – WTR Titan III B/NUS Launch Vehicle

CONDITION	LONGITUDINAL X	LATERAL Y OR Z
LIFT-OFF	+ 2.3 - 0.8	2.0
STAGE I SHUTDOWN (DEPLETION)	+ 8.2 - 2.5	1.5
STAGE II SHUTDOWN (COMMAND)	+ 10.8 - 2.0	1.5

NOTES:
1. LOAD FACTOR CARRIES THE SIGN OF THE EXTERNALLY APPLIED LOAD.
2. INCLUDES BOTH STEADY STATE AND DYNAMIC CONDITIONS.

T1-10
(1) 5T-31

Table 3-8 Limit Load Factors – Payload Bay Shuttle

CONDITION	DIRECTIONS (3)		
	X	Y	Z
LIFT-OFF (1)	+ 1.7 ± 0.6	± 0.3	+ 0.8 + 0.2
HIGH Q BOOST	+ 1.9	± 0.2	- 0.2 + 0.5
BOOSTER END BURN	+ 3.0 ± 0.3	± 0.2	+ 0.4
ORBITER END BURN	+ 3.0 ± 0.3	± 0.2	+ 0.5
SPACE OPERATIONS	+ 0.2 - 0.1	± 0.1	± 0.1
ENTRY	± 0.25	± 0.5	- 3.0 + 1.0
SUBSONIC MANEUVERING	± 0.25	± 0.5	- 2.5 + 1.0
LANDING AND BRAKING	± 1.5	± 1.5	- 2.5
CRASH (ULTIMATE) (2)	- 9.5 + 1.5	± 1.5	- 4.5 + 2.0

NOTES
1. THESE FACTORS INCLUDE DYNAMIC TRANSIENT LOAD FACTORS.
2. THESE FACTORS ARE ULTIMATE AND ONLY USED TO DESIGN PAYLOAD SUPPORT FITTINGS. THE SPECIFIED CRASH LOAD FACTORS SHALL ACT SEPARATELY.
3. LOAD FACTOR CARRIES THE SIGN OF THE EXTERNALLY APPLIED LOAD. POSITIVE X, Y, Z DIRECTIONS EQUAL FORWARD, RIGHT AND DOWN.

T1-11
(1) 5T-32

3.2.6.1.4.1 Acceleration and Sustained Loads

Acceleration and sustained loads resulting from orbit adjust and reaction control system's firing shall be as defined in EOS-SY-105.

3.2.6.1.4.2 Vibration

Vibration due to operation of the orbit adjust and reaction control systems shall be as defined in EOS-SY-101 to EOS-SY-105.

3.2.6.1.4.3 Shock

Shock impulses due to activation of on-board pyrotechnically operated devices and impact of deployable devices at the end of their stroke shall be as defined in EOS-SY-101 to EOS-SY-105.

3.2.6.1.4.4 Vacuum

The atmospheric pressure at mission altitude shall be in the order of specific gas properties. The NASA atmospheric model as specified in NASA TMX-64589 shall be used to predict the gas properties of the orbital altitude region of the atmosphere.

3.2.6.1.4.5 Meteoroid

The meteoroid environment as defined in NASA TMX-64627, Paragraph 2.5, shall be used for the sporadic and stream meteoroid particle density particle velocity, flux-mass model, body shielding factor and other pertinent environmental data. The thermal control skins shall be designed to survive the space environment. For design, the average total meteoroid (average sporadic plus a derived average stream) environment shall be as given below. Since the mass density flux model given below includes a derived average stream environment, the damage due to stream meteoroids shall be evaluated.

(a) Particle Density - The mass density shall be 0.5 gm/cm^3 for all meteoroid particle sizes.

(b) Particle Velocity - The average meteoroid particle velocity shall be 20 km/sec.

(c) Flux-Mass Model - The average annual cumulative meteoroid flux-mass model in logarithmic plot shall be as described mathematically as follows:

$$\text{Log } N_t = -14.37 - 1.213 \text{ Log } m \quad ; \text{for } 10^{-6} \leqq m \leqq 10^{0}$$

$$\text{Log } N_t = -14.339 - 1.584 \text{ Log } m - 0.063 (\text{Log } m)^2 \quad ; \text{for } 10^{-12} \leqq m \leqq 10^{-6}$$

where: N_t = Number of particles/m^2/sec of mass m or greater

m = mass in grams

The gravitationally focused, unshielded flux, N, shall be multiplied by an appropriate defocusing factor for earth, G_e, and the shielding factor. The G_e factor applies to all missions and is obtained from the following equation:

$$G_e = 0.568 + 0.432/r$$

where:

r = The distance from the center of the earth in units of the earth's radius.

3.2.6.1.4.6 Temperature

The Observatory shall be capable of operating within specification limits during exposure to temperatures resulting from the following factors:

(a) Thermal radiation to free space

(b) Conductive and radiative heat transfer between equipment and structural members

(c) Solar irradiation - Depending upon the season, sunlit surfaces receive a radiation intensity of 1309 to 1400 W/M^2 over an area projected normal to the sun's rays. Tolerances and solar spectrum as specified in NASA SP-8005

(d) Earth albedo and emitted radiation as specified in NASA SP-8067.

3.2.6.1.4.7 Solar Radiation

The characteristics of solar radiation, including seasonal variation, spectrum and tolerances, shall be as specified in NASA SP-8005.

3.2.6.1.4.8 Geomagnetic Trapped Radiation Environment

The Observatory shall be capable of operating within specification limits during exposure to the geomagnetically trapped particle environment defined as follows:

- Electrons as defined in NASA SP-3024, Vol III
  - NASA, NSSDC 72-06
  - NASA, NSSDC 72-10
  - NASA, NSSDC 72-13
- Protons as defined in NASA SP-3024 Vol. IV through VI

3.2.6.1.4.9 Solar Flare Protons

The Observatory shall be capable of operating within specification limits during exposure to the environment as defined in NASA TM X-53865

3.2.6.1.4.10 Solar Flare Alpha Particles

The Observatory shall be capable of operating within specification limits during exposure to an alpha particle environment, which shall be taken as 10 percent of the proton environment as defined in NASA TM X-53865. This environment shall be reduced for protons remote from the activity peak cycle.

3.2.6.2 SUPPORT EQUIPMENT ENVIRONMENTAL CONDITIONS

The support equipment shall be capable of operating within specification limits during and/or after exposure to both controlled and natural environments experienced during test, transportation, handling, storage, prelaunch, and launch operations. The environmental parameters outlined in this section and MIL-STD-810 shall be adhered to.

3.2.6.2.1 Transportation, Handling and Storage Environment

The support equipment shall be capable of operating within specification limits after exposure to both controlled and natural environments while in a non-operating condition during transportation, handling and storage. The support equipment shall be protected from (a) air transportation, temperatures of $-40^{o}C$ to $+66^{o}C$ with an $18^{o}C$ variation per minute for 5 minutes, and (b) truck transportation temperature of $-40^{o}C$ to $+66^{o}C$ with a $3^{o}C$ variation per minute. The maximum range in altitude will occur during air shipment from sea level to 15,000 m at a maximum descent rate of 6 m/s.

3.2.6.2.2 Prelaunch Environment

The support equipment to be used at the launch site, not located in controlled environments, shall be designed to be capable of operating within specification limits during and after exposure to the following:

- Temperature - Surrounding air temperatures from $-3.9^{o}C$ to $41.7^{o}C$
- Humidity - Relative humidities up to 100% with conditions such that condensation takes place in the form of water or frost
- Fungus - Fungus equivalent to 28 days exposure to selected fungi as described in MIL-STD-810
- Sand and Dust - Graded wind-blown sand and dust equivalent to exposure for 6 hours in a sand and dust chamber

- Sunshine - Materials shall withstand the deterioration effects of direct sunlight equivalent to a solar radiation exposure of 1179 W/m^2
- Salt Fog - Salt fog equivalent exposure of 5% salt spray for 50 hours for equipment external to the Observatory
- Rain - Rain equivalent to 64mm per hour for 2 hours
- Explosive Atmospheres - Explosive atmospheres surrounding non-hermetically sealed equipment as described in MIL-STD-810, Procedure I, Method 511.

3.3 DESIGN AND CONSTRUCTION

3.3.1 MATERIALS, PROCESSES AND PARTS

The selection and application of materials, processes, and parts shall be in accordance with the requirements identified herein. Materials which may outgas under the operational temperature and vacuum environments and condense on the observatory optics shall be minimized. Design mechanical properties shall be governed by MIL-HDBK-5 for metallic materials, welds and fasteners, MIL-HDBK-17 for reinforced plastic and MIL-HDBK-23 for structural sandwich composites.

3.3.1.1 OBSERVATORY MATERIALS, PROCESSES AND PARTS

3.3.1.1.1 Polymer Materials

To minimize the possibility of optics contamination, polymeric materials shall not be used which produce greater than 0.1 percent volatile condensable material (VCM) when tested as described in NASA Report CR-89557, "Polymers for Spacecraft Applications", by R. F. Muraca and J. S. Whitteck. Exceptions to the 0.1 percent VCM requirement are permissible within environmentally/hermetically sealed containers, when operational temperature, and/or location minimize contamination, or when prior thermal-vacuum bakeout is employed to eliminate outgassing products. Materials shall not lose more than 1 percent of total weight after exposure for 24 hours at $125^{o}C$, and 1.333×10^{-4} newton/m^2 (1×10^{-6} Torr).

3.3.1.1.2 Lubricants

Lubricants shall meet the outgassisng requirements of Paragraph 3.3.1.1.1. Lubricants shall not degrade in their operational environment. Lubricants shall not be exposed to outgassing products which are incompatible with the lubricants. When specifying lubricants the Space Materials Handbook, NASA SP-8063 shall be used as a guideline along with such criteria as bearing or gear type, specific application, opposing surface materials etc.

3.3.1.1.3 Dissimilar Metals

Dissimilar metals shall not be used in contact with each other unless the metals are suitably protected against electrolytic and chemical corrosion to the extent that no contamination or operational impairment of useful life shall result. Metals shall be considered compatible if they are in the same grouping as specified in MSFC-SPEC-250.

3.3.1.1.4 Corrosion of Metal Parts

Metal parts shall be of corrosion resistant materials, or shall be processed to resist corrosion. Such corrosion resistant processes shall not prevent bonding compliance with MIL-B-5087. All parts, assemblies and equipment shall be finished to provide protection from corrosion in accordance with the requirements of MSFC-SPEC-250. Cadmium, zinc, or electro-deposited tin shall not be used as a finish.

3.3.1.1.5 Moisture and Fungus Resistance

Non-nutrient materials, as defined in MIL-STD-810, Method 508, shall be used whenever possible. If it is necessary to use nutrient materials outside of hermetically sealed containers, such materials shall meeet the fungus requirements specified in MIL-STD-454, Requirement 4.

3.3.1.1.6 Reaction of Materials

Materials shall not be used which may produce toxic, noxious or corrosive products under the ground operations environment. Observatory materials which are exposed to the fumes spillage, and combustion products of the propellants used in the Observatory segment shall be compatible therewith.

3.3.1.1.7 Drains

Drain holes and drainage provisions shall be in accordance with MIL-STD-454. Drain holes and drainage provisions shall be provided as required; when construction does not permit drainage provisions, affected areas of the Observatory shall be protected with a corrosion preventative finish.

3.3.1.1.8 Fasteners

All fasteners, shall comply with the requirements of MIL-STD-1515. No cadmium, zinc, or electrodeposited tin shall be used as a fastener finish. The use of silver coated fasteners in direct contact with titanium alloys shall be avoided.

3.3.1.1.9 Wiring

The fabrication, installation and inspection of Observatory cabling and wiring for the interconnection of electrical and electronic equipment, and the electrical wiring design for electromagnetic compatibility shall be in accordance with MIL-W-83575 (USAF) and S-320-G-1 (GSFC).

3.3.1.1.10 Stress Corrosion

The usage of metallic materials in structural elements and components shall be limited to those alloys which possess established stress corrosion cracking stress levels well above the calculated residual stresses in vehicle members. MSFC Drawing 10M33107 shall be utilized as a guideline for controlling stress corrosion.

3.3.1.1.11 Soldering

The soldering of sheet metals and electronic assemblies shall be performed in accordance with NASA Document NHB 5300.4 (3A) by certified operators.

3.3.1.1.12 Glass Fiber Reinforced Plastics (GFRP)

Glass fiber reinforced plastic parts shall meet the requirements of MIL-P-9400. Materials employed shall be in accordance with MIL-P-25421 shall require specific approval.

3.3.1.1.13 Electronic, Electrical, and Electromechanical (EEE) Parts

3.3.1.1.13.1 Parts Program

A Parts Program in accordance with NHB 5300.4 (1A) shall be implemented. Key requirements of this program shall be:

3.3.1.1.13.2 Parts Materials and Processes Control Board (PMPCB)

A PMPCB chaired by the Contractor and consisting of the Contractor and subcontractor's parts specialists shall be established. The PMPCB shall be the focal point for the selection, control and coordination of all Observatory Parts, Materials and Processes (PMP) activities. Parts control standardization and justification for the use of non-standard parts shall be in accordance with the Procurement Data Requirements Documents.

3.3.1.1.13.3 Selection

Observatory electronic, electrical and electromechanical parts selections for all new equipments and modified portions of off-the-shelf equipments shall be from the GFSC Preferred Parts List (PPL) and approved by the PMPCB. Parts on the PPL shall be selected with due regard to radiation susceptibility to minimize the impact of equipment redesigns for hardening for subsequent mission alternates. Off-the-shelf equipment parts shall be reviewed for compatibility with EOS life requirements and the PPL. Any deviations shall be fully justified, substantiated and approved by the PMPCB. The PMPCB shall be responsible for monitoring, updating, and administering the PPL.

3.3.1.1.13.4 Parts Manufacturer's Control

Systematic surveys of parts manufacturers shall be scheduled and performed to verify compliance to provide program visibility. Specific areas to be monitored are: process controls; screening and inspection test; quality assurance trends; traceability of product to raw materials; materials control; failure analysis capability; and implementation of corrective action.

3.3.1.1.13.5 Screening

EEE parts screening tests shall be incorporated in preferred parts procurement specifications.

3.3.1.1.13.6 Derating

The EOS Electronic Parts Derating Policy, based upon the derating guidelines of GFSC PPL, shall be imposed on circuit designs for all new equipments and modified portions of off-the-shelf equipments. Off-the-shelf equipments parts applications shall be reviewed for compatibility with the Derating Policy and the reliability requirements of the Observatory. Any deviations shall be fully justified, substantiated and approved by the PMPCB.

3.3.1.1.14 Cleanliness - General Requirements

Surface Cleanliness Requirements of the Observatory and support equipment used in proximity of the Observatory shall maintain Level TBD of MIL-STD-1246 up to launch. After fabrication, parts and assemblies shall be visibly clean and free from processing or handling damage and imperfections. There shall be no dirt, oil, grease, foreign particles, processing debris such as chips, fillings, loose or spattered solder on or within the hardware which might detract from the intended operation, function or appearance of the hardware. This includes particles that could loosen or become dislodged during the normal expected life of the equipment. All corrosive or foreign materials shall be removed. Whenever possible, the cleaning shall take place before parts are assembled into the equipment.

Cleaning processes shall have no deleterious effect on the equipment or parts. Observatory assembly and test operations shall be performed in a class TBD clean room in compliance with Fed-Std-209.

3.3.1.1.15 General Processes

3.3.1.1.15.1 Heat Treatment

Heat treatment of metal alloy parts shall meet the requirements of MIL-H-6875 (steel), MIL-H-6088 (aluminum alloys).

3.3.1.1.15.2 Welding

The suitability of the equipment, the welding processes, the welding supplies and the supplementary treatments selected shall be demonstrated through testing of welded specimens representative of the materials and joint configurations. Weld rod or wire used as filler metal on structural parts shall be fully certified and documented for composition, type, heat number, manufacturer, supplier, etc., as required to provide positive traceability to the end use item. No spliced wire spools shall be used. Welding shall be performed by welders qualified in accordance with MIL-T-5021.

(a) Aluminum

Series 5000, 6000 aluminum alloys and 2219 aluminum alloy may be either machined or manually welded.

(b) Steel

Dissimilar steels, free machining grades, high hardenability 400 series steels, steels heat treated above 140 ksi and A-286 shall not be welded.

3.3.1.1.15.3 Brazing

Brazing shall meet the requirements of MIL-B-7883. Subsequent fusion welding operations in the vicinity of brazed joints or other operations involving high temperature which might affect the brazed joint are prohibited.

3.3.1.1.15.4 Sandwich Construction

All sandwich assembly designs and constructions shall be in accordance with MIL-HDBK-23 and processed in accordance with MIL-A-25463 and MIL-A-9067 for metallic sandwich construction. All plastic sandwich construction shall be in conformance with

MIL-P-9400. Sandwich assemblies shall preclude the entrance and entrapment of water and other contaminants. Perforated metallic core shall not be used; the core material will be in accordance with MIL-S-7438.

3.3.1.1.15.5 Potting and Encapsulation

Where applicable, electrical and electronic assemblies may be potted or encapsulated for protection from environmental effects. Potting and encapsulating material selection shall be predicated upon environmental service conditions as well as electrical and physical property requirements. Particular attention shall be given to material outgassing characteristics in accordance with Paragraph 3.3.1.1.1.

3.3.1.2 SUPPORT EQUIPMENT MATERIALS, PROCESSES AND PARTS

3.3.1.2.1 Corrosion Resistance

Metals in all instruments/sensors shall either be of a corrosion resistant type or suitably treated to resist corrosion. Protective methods and materials for cleaning, surface treatment, and application of finishes and protective coatings shall be in accordance with the requirements of MIL-F-7179. Cadmium, zinc, or electrodeposited tin are permissible except in those areas which interface with the Observatory.

3.3.1.2.2 Fungus Resistance

Materials which are not nutrients for fungi shall be used to the greatest extent practicable. Where nutrient materials must be used outside of hermetically sealed containers, such materials shall be treated with a fungicidal agent in accordance with the requirements of MIL-STD-454.

3.3.1.2.3 Calibration

The Spacecraft Support Equipment shall be calibrated using requirements of MIL-C-45662.

3.3.1.2.4 Drains

Instrument/Sensor drainholes and drainage provisions shall be established in accordance with the requirements of MIL-STD-454.

3.3.1.2.5 Fasteners

All Instrument/Sensor fastener applications shall comply with the requirements of MIL-STD-1515.

3.3.1.2.6 Electrical, Electronic, and Electromechanical (EEE) Parts

3.3.1.2.6.1 Parts Program

A support equipment Parts Program in accordance with NHB 5300.4 (1A) shall be implemented. It shall be coordinated with the Space Systems Parts Program.

3.3.2 ELECTROMAGNETIC RADIATION

3.3.2.1 OBSERVATORY

The conducted and radiated electromagnetic emissions from the Observatory and the susceptibility of the observatory to conduct or radiate emissions shall be in accordance with MIL-STD-401/462, MIL-E-6051, and the ICD's for other EOS segments.

Electromagnetic compatibility (EMC) among all subsystems of the Observatory, between the Observatory and all other EOS segment, and between the Observatory and its launch and mission electromagnetic environments shall be in accordance with MIL-STD-461/462, MIL-E-6051.

3.3.2.2 SUPPORT EQUIPMENT ELECTROMAGNETIC RADIATION

EMC design efforts shall consider the following:

(a) Interface compatibility with the EOS.

(b) Sellers shall be monitored to ensure EMC for subcontracted assemblies.

(c) Support equipment shall be designed using MIL-STD-461 as a guide.

(d) The EMC interface safety margin of 6dB minimum for power and signal circuits and 20dB for pyro circuits shall be satisfied.

3.3.2.3 GROUNDING

3.3.2.3.1 Structure

All structural members of the Spacecraft shall be designed to provide electrical conductivity across all mechanical joints except where DC isolation is required for maximum electrical reliability. Conductive surface protection coatings shall be used at all joints.

3.3.2.3.2 Electrical

3.3.2.3.2.1 Central Ground Point

A Central Ground Point (CGP) shall be provided on the spacecraft structure in the vicinity of the power subsystem and C&DH subsystem interface connectors. The CGP shall be the bussing point to the structure for signal returns and AC circuit returns for carrier or servo systems employed in the subsystem or instruments. Power returns shall be bussed in the power module and then returned to the CGP to minimize common impendance in the power distribution circuitry.

3.3.2.3.2.2 Shield Grounding

All shields shall be grounded at each end of the nearest chassis or structural ground if feasible. For RF cables, the connector shell shall provide the electrical circuit connection to chassis or structure. All other shields and shield ground leads shall be used strictly for a shielding function (no signal or power currents on shields). The method(s) selected for terminating the shields of multi-conductor cables at connectors shall, of necessity be governed by the types of connectors being used. The preferred methods shall be: (a) grounding shields via electrically conductive connector shells; (b) the use of shield grounding studs adjacent to the connectors with very short jumpers between the shield termination and the stud; or (c) the use of one or more connector contacts to carry the shield return through the connector to a short ground return at the rear of the receptacle.

3.3.2.3.2.3 Signal Grounds

Signal circuit grounds, which normally carry only a few milliamperes of signal or DC currents, shall be returned to a separate bus at the CGP via the common signal ground bus in the module. Signal ground shall be isolated from power circuit ground. The signal ground shall be the point to which all single-ended control and data circuitry is referred. A signal ground lead shall be used between each module and the CGP. The signal ground bus within each module shall be the common for all input/output signals between assemblies within the module but shall be grounded to structure only at the CGP.

3.3.2.3.2.4 Party Line/Clock Grounds

The returns of party line and clock circuits from the C&DH module, which drive numerous loads in parallel, shall be isolated from signal ground except for a connection to the signal ground bus in the C&DH module. Circuit isolation shall be required if some circuits of this return bus carry relatively high signal current.

3.3.2.3.2.5 DC Power Circuit Grounds

Power circuit grounds from each module shall be tied to a common bus in the power system module. This bus shall be returned to the CGP.

3.3.2.3.2.6 AC Circuit Grounds

Two types of AC circuit grounds shall be employed. Alternating current power and carrier type circuits shall be returned on separate leads to an AC circuit bussing point at the CGP.

3.3.2.3.2.7 Chassis Grounding

All chassis shall normally be grounded directly to the module or structure. Electrical bonding devices shall be used across areas requiring thermal bonding or isolation to assure low ground circuit impedance.

3.3.2.3.2.8 Telemetry Circuit Grounds

Telemetry outputs shall normally be referenced to signal ground. Bilevel telemetry signals powered directly by the power bus shall be referenced to the power return.

3.3.2.3.2.9 Static Discharge

Provisions shall be made for static discharge or equalization of charge potential between the Observatory and Shuttle Orbiter during retrieval operations.

3.3.3.1 OBSERVATORY

Identification and marking of the spacecraft, the Mission Peculiar Components and Structures, and, the S/C to L/V Adapter shall conform to the requirements of MIL-STD-130 and MIL-STD-1247 or applicable Grumman STD Specs., i.e., GSS 4710-1 and/or GSS 4711.

3.3.3.2 SUPPORT EQUIPMENT NAME PLATE & PRODUCT MARKINGS

Identification and marking of support equipment shall be in accordance with MIL-STD-130 and MIL-STD-493, Appendix IX.

3.3.3 NAMEPLATES AND PRODUCT MARKINGS

3.3.4 WORKMANSHIP

All EOS hardware including detailed parts, subassemblies and installations shall be fabricated and assembled and finished to good commercial practice or in a manner which satisfies the workmanship standards specified in the current Government-approved Grumman drawing No. 659001.

3.3.5 INTERCHANGEABILITY

3.3.5.1 OBSERVATORY

The Observatory shall be compatible with the Delta 2910 and for follow-on missions the Weight Constrained Titan and the Titan IIIB launch vehicles as defined in Paragraph 3.1.6.2.1. Further, the Observatories will be compatible with the Shuttle Orbiter.

The Basic Spacecraft standard modules, communication and data handling, attitude and control, electrical power, and the orbit adjust/reaction control, shall be replaceable and interchangeable between Observatories.

3.3.5.2 SUPPORT EQUIPMENT

As applicable, the support equipment shall be interchangeable with interfacing equipment and facilities.

3.3.6 SAFETY

3.3.6.1 GENERAL

The Observatory shall be designed, fabricated, tested in plant, transported, operated at the launch base, launched, deployed and modifications made where necessary, to conform to the requirements of SAMTECM 127-1, the requirements of MIL-STD-1512 for pyrotechnics and MIL-STD-1522 for flight equipment. SAMSOM 127-8, Chapters 7 and 8 shall guide the implementation of the EOS accident prevention program.

3.3.6.1.1 Safety Analysis Reports

Safety Hazard Analysis Reports (HAR's) shall be generated to identify the deviations to the specified safety requirements for flight and ground SE.

3.3.6.2 SPACE VEHICLE SAFETY

Hazards to the ground crew and Shuttle Orbiter crew from hydrazine, pyrotechnics and X and Ku Band radiations, to the public from random reentry and to flight/ground support equipment and facilities from hazardous failure modes shall have as goals, satisfaction of these requirements:

(a) Hydrazine - No leaks, no spills, no toxic fume, splash or submersion per AFM 160-39, MIL-STD-1522 (USAF), and SAMTECM 127-1, Chapter 3.3.6.2.

(b) Pyrotechnics - No inadvertent initiation per MIL-STD-1512 (USAF) and SAMTECM 127-1 chapter 3.

(c) X and Ku Band Radiation - No exposures in excess of 10 mw/cm^2 per MIL-STD-454 Rqt. 1 and SAMTECM 127-1 Paragraph 3.3.12.7.

(d) Public Safety - No equipment materials that would survive entry heating and a mean number of casualties computed per SAMTECM 127-1 Paragraph 2.4 of less than 10^{-6}.

(e) Equipment/Facilities - Accidental damage preventing or impairing successful satellite deployment less than 10^{-6}.

3.3.6.3 SUPPORT EQUIPMENT

The SE shall utilize the requirements of Paragraph 3.3.6.1 and the specific requirements delineated in the following documents in the design of SE for safe operation.

3.3.6.3.1 Fluid and Mechanical Support Equipment Safety Requirements -

(a) AFSC DH1-X Section 6D

DN 6D1	All Items
DN 6D2	All except 7.1, 7.2, 7.3, 7.5, 10.1, 10.2, 11.3, 11.4, 11.5, and 11.6
DN 6D5	All except 1.2
DN 6D6	All except 3.7

(b) SAMTECM 127-1

3.3.6.3.2 Electrical Equipment

(a) AFSC DH1-X - Section 6D

DN 6D7	All
DN 6D8	All

(b) MIL-STD-454 Requirement 1 -

(c) SAMTECM 127-1

3.3.6.4 GROUND CREW SAFETY EQUIPMENT

Special items are required to protect ground crew personnel from EOS manufacture, test and operations environments. Table 3-9 identifies these items, potential source and applicable usage location.

3.3.6.5 SHUTTLE ORBITER CREW

The Observatory must provide immediate relay to the Orbiter Crew of Observatory parameters critical to the safety of the Crew while the Observatory is in the vicinity of the Orbiter and during Observatory retrieval. Provision must also be made for command override of critical Observatory functions by the Orbiter Crew and for fail operational design of Observatory critical data transmission, and command receipt, while the Observatory is in the vicinity/attached to the Orbiter.

Table 3-9 Safety Equipment

Item	Use MFR	TEST	VAFB
Scapesuits			•
Splash Suits			•
Hard Hats		•	•
Face Shields		•	•
Goggles	•	•	
Safety Glasses	•		
Portable Eye Wash			•
Scott Packs		•	•
O_2 Cannisters		•	•
Conductive Shoes		•	•
or Leg Stats		•	•
Wrist Stats		•	•
Grounding Mats		•	•
Earmuffs	•	•	
Jumper Cables	•	•	•

(1) 5T-33

3.3.7 HUMAN ENGINEERING

3.3.7.1 OBSERVATORY

New or modified Observatory equipment shall be designed in accordance with the requirements of Paragraphs 4.0 through 4.8, 5.9 and 5.13 of MIL-STD-1472.

3.3.7.2 SUPPORT EQUIPMENT

New or modified SE shall be designed using the requirements of Paragraphs 5.1 through 5.7 of MIL-STD-1472.

3.3.8 SOFTWARE DESIGN & CONSTRUCTION

The Observatory software shall be designed to interface and be compatible with the Spacecraft multiplex system. The Spacecraft telemetry system and other Spacecraft devices available to the computer input/output circuits. The Observatory software shall implement the Observatory function listed in Paragraph 3.7.1.8 and shall:

(1)5

(a) Be compatible with GSFC command formats as specified in Paragraph 3.7.1.1.5.2.2.1.

(b) Be compatible with GSFC multiplex data bus formats as specified in Paragraph 3.7.1.1.5.2.2.1.

(c) Be of a modular structure to permit enable/disable; modification or replacement of each module without disturbing the function of other modules, Paragraph 3.7.1.1.5.2.2.1.

(d) Minimize computer operating time.

(e) Minimize computer memory use.

(f) Provide flexibility to permit recovery in event of data processing interruption or failure.

3.4 DOCUMENTATION

All Observatory interface requirements shall be as specified in the ICD's identified in Paragraph 3.1.5. All other requirements shall be documented in the specifications shown in Fig. 3-13.

3.5 LOGISTICS

3.5.1 MAINTENANCE

The Observatory and its Support Equipment shall be maintained by replacement of equipment, as required, at the subsystem module level for the Observatory and to the line replaceable unit (meter, scope, drawer) for the Support Equipment.

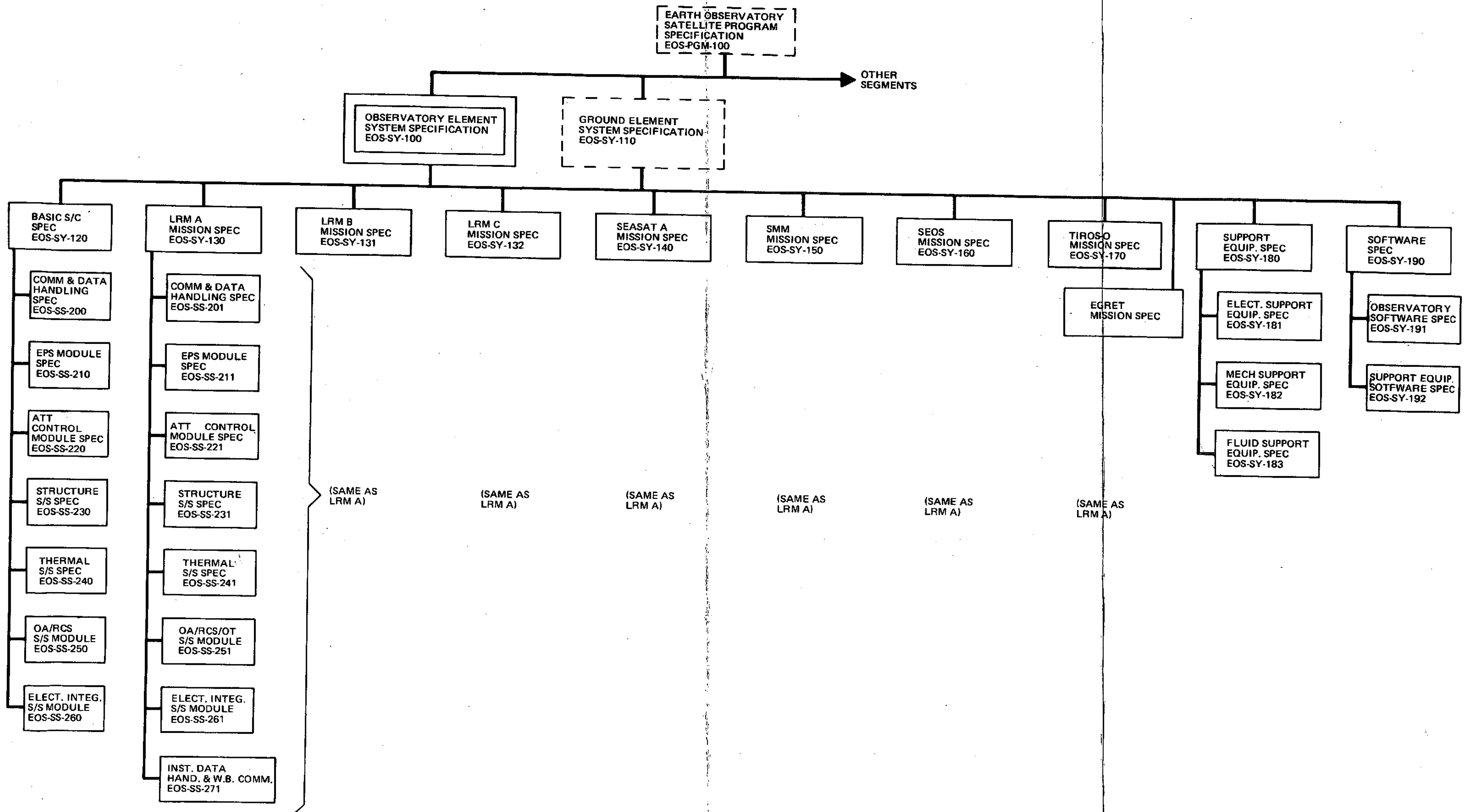
Repair of the S/S modules shall be to the replaceable unit within the module.

Repair of the replaceable units shall be performed by the units manufacturer.

3.5.2 SUPPLY

Spares shall consist of 25 percent of the total material content for one each non-redundant section of the Observatory. The following components shall be spared as complete subassemblies:

- One Power S/S Module
- One Communications and Data Handling S/S Module
- One Attitude and Control S/S Module
- One Orbit Adjust/RCS Module



(1) 5-15

Fig. 3-13 Specification Tree

3.5.3 FACILITIES AND FACILITES EQUIPMENT

The Observatory shall utilize the NASA assigned facilities at the Western Test Range (WTR).

- Space Launch Complex TBD for prelaunch and launch activities

3.6 PERSONNEL AND TRAINING

The contractor shall provide the necessary personnel to accomplish and support: the manufacture and assembly process; the operation of support equipment; the prelaunch and launch operation and control functions at the launch site; the post-launch clean up and analysis; the preparation of orbital plans; participation in interface working groups; the necessary tests for prelaunch and launch operations.

Contractor personnel shall receive preparatory training to ensure against personal injury or damage to equipment. Particular attention shall be given to any hazardous operations and to the cleanliness requirements of Observatory, particularly of the instrument sensors.

Indoctrination and familiarization shall be provided for GSFC personnel involved in the operation and support of the Observatory. This training shall be of the "on-the-job" type, including formal classroom instruction.

3.7 FUNCTIONAL AREA CHARACTERISTICS

3.7.1 BASIC SPACECRAFT SUBSYSTEM FUNCTIONAL CHARACTERISTICS

The Basic Spacecraft shall consist of a triangular structure to support the three basic modules, the ACS, EPS, and Communication and Data Handling modules. The Orbit Adjust/RCS Module shall attach to the lower bulkhead completing the Basic Spacecraft.

3.7.1.1 COMMUNICATIONS AND DATA HANDLING (C & DH)

The C&DH subsystem shall satisfy the Spacecraft requirements and be compatible with the operational requirements defined in the NASA/GSFC STDN Users Guide No. 101.1 and the Aerospace Data System Standards X-560-63-2, and TDRS users Guide No. X-805-74-176.

3.7.1.1.1 General Requirements

The C&DH subsystem shall provide the means of commanding the Spacecraft and payload instruments via the uplink, provide onboard instrumentation and data handling required for ground monitoring of the Spacecraft and payload status via downlink telemetry, provide for monitoring and control of Observatory functions that are critical to the safety of the Shuttle Orbiter Crew during retrieval, and transpond ranging signals for ground tracking of the Spacecraft.

3.7.1.1.2 Functions

The C&DH shall:

(a) Provide telemetry, tracking and command compatibility with STDN/Orbiter direct and relay (TDRS).

(b) Acquire, process, record, format and route data/commands from the appropriate Spacecraft subsystem modules.

(c) Execute ground/Orbiter commands in both real and delayed time.

(d) Provide on-board sequencing for Spacecraft functions scheduled to occur during and after launch prior to initial ground contact.

(e) Compress and store hi-lo mean deviation and latest selected housekeeping data in memory.

(f) Store all Spacecraft housekeeping data between ground contacts via a tape recorder (option).

3.7.1.1.3 Configuration

The major component of the C&DH subsystem shall be:

(a) STDN S-Band Transponder Assembly (includes transponder, 2 diplexers, hybrid and coaxial switch).

(b) TDRS S-Band Transponder

(c) TLM/CMD Antennas

(d) Computer

(e) Command Decoders

(f) Bus Controller/Formatters

(g) Remote Units

(h) Signal Conditioners

(i) Clock

The C&DH subsystem shall be configured as shown in Fig. 3-14.

3.7.1.1.4 Modes of Operations

The C&DH subsystem modes of operation shall be selected by execution of ground or stored commands. The C&DH subsystem modes of operation shall be as follows:

(a) Commands

    (1) Real time
    (2) Delayed
    (3) On-board computer generated

(b) Telemetry

    (1) Real time low data rate, selectable
    (2) Memory dump, medium data rate (direct only)
    (3) Combined low and medium data range (direct only)

(c) Ranging

    (1) Turnaround ranging
    (2) Turnaround ranging with command and/or low rate telemetry.

3.7.1.1.5 Performance Requirements

3.7.1.1.5.1 Communications Group

The communications group of the C&DH module provides telemetry tracking and command link compatibility with STDN direct and relay (TDRS) at S-Band.

3.7.1.1.5.1.1 Command

The command RF equipment of the C&DH module shall receive, demodulate and execute commands generated by the ground for control of the Spacecraft.

3.7.1.1.5.1.1.1 Link Considerations

3.7.1.1.5.1.1.1.1 STDN Direct

The Spacecraft command link shall receive command data from STDN direct with a minimum design margin of 6 dB above the signal level required for a 10^{-5} bit error rate under the following conditions:

(a) Frequency: 2025 to 2120 MHz

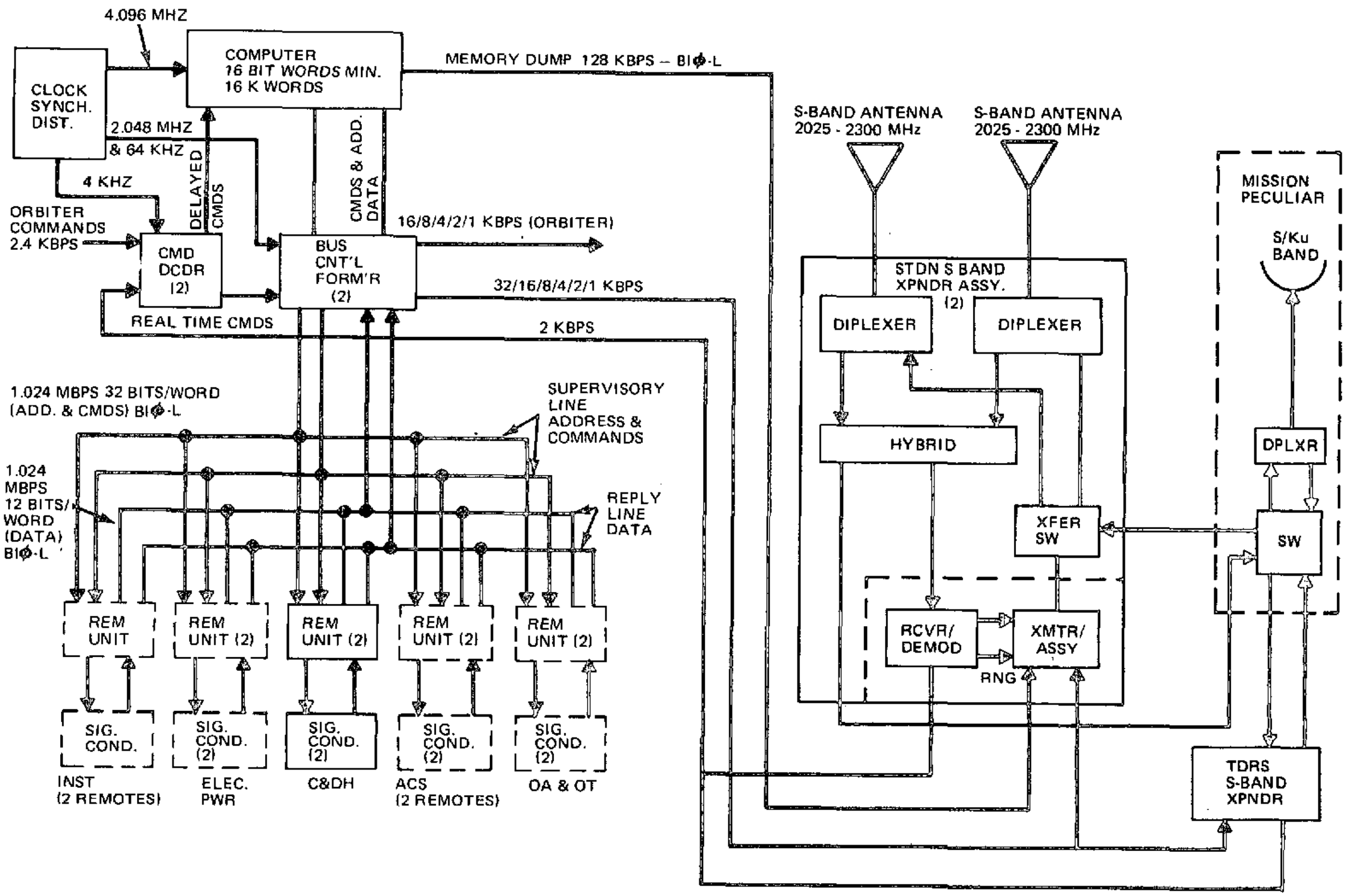


Fig. 3-14 Communications and Data Handling Subsystem

(b) Ground Antenna Size: 30-ft dish

(c) Minimum Elevation Angle of Ground Antenna: 5 degrees

(d) Minimum Ground Transmitted Power: 500 Watts

(e) Atmosphere Loss: 0.6 dB

(f) Polarization: RHCP

(g) Maximum Slant Range: 3040 Km

(h) Uplink data rate: 2 Kbps

(i) E/No Required: 12 dB

3.7.1.1.5.1.2.1.2 STDN Relay (TDRS)

The Spacecraft command link shall receive command data from STDN relay (TDRS) with a minimum design margin of 3 dB above the signal level required for a 10^{-5} bit error rate under the following conditions:

(a) TDRS Mode: Single Access

(b) Frequency: TBD MHz in the 2025 to 2120 MHz range

(c) Command bit rate: 2000 bps

(d) TDRS EIRP: 43.4 dBW

(e) Polarization: RHCP

(f) Polarization Loss: 0.5 dB

(g) Maximum Range (LOS): 42,000 Km

(h) E/No Required, DPSK: 9.9 dB

(i) Command Modulation: PN Spread Spectrum, PSK ($\pm 90^{\circ}$), biphase

(j) PN Chip Rate: 6.0 M chips/sec

(k) Command Decoding: Maximum Likelihood Decoding (Viterbi Decoder)
FEC Gain ($R = 1/2$, $K = 7$): 5.2 dB

3.7.1.1.5.1.1.2 Command Antenna(s)

The command antenna(s) shall be integrated into a dual frequency command/telemetry antenna assembly.

(a) Frequency: Each antenna shall receive a frequency f_r ± 10 MHz in the STDN uplink band of 2025 to 2120 MHz.

(b) Coverage: Each antenna shall provide the following coverage at the specified frequency when the antenna is mounted on the Basic Spacecraft:

- Conical sector of 130 degrees with a minimum gain of -4 dBi with respect to a right hand circularly polarized isotropic radiator, boresighted in the + and - Z axes.

(c) Polarization: Each antenna shall receive right hand circular polarizations.

3.7.1.1.5.1.1.3 Command Receiver (STDN Direct)

The command receiver shall be part of the S-Band integrated transponder and shall include a demodulation subassembly and shall have two major subassemblies:

Receiver Subassembly - Shall translate and detect S-Band RF signals for extraction of command and ranging modulations.

Demodulation Subassembly - Shall convert command modulated subcarrier to digital signals.

(a) Input Signal - Standard STDN PCM/PSK - FM/PM command signals shall be accommodated on a 70 KHz subcarrier. The bit rate shall be 2000 bps. The capability shall exist to accommodate a modulation index on the command channel of less than or equal to 3.

(b) Frequency Uncertainty - The best lock receiver frequency shall be within 15 parts in 10^6 of the assigned channel during all acceptance test environments and 20 parts in 10^6 during all qualification test environments. This stability shall be achieved within 10 minutes after turn on.

(c) Receiver RF Input - A phase modulated reception capability shall be provided for any STDN direct channel in the 2025 to 2120 MHz band. The channel tolerance and doppler capability shall be ±100 KHz. The specified performance shall be met over an input from -95 dBm to -30 dBm. No damage shall occur with ann RF input up to +3 dBm.

(d) Carrier Squelch - Acquisition of the uplink RF carrier shall initiate the transfer of power mode stand by to operate. This mode change shall be applied to the demodulator subassembly and other elements of the command equipment as required.

(e) Noise Figure - The noise figure of the receiver, measured at the input connector shall not exceed 7.5 dB for input signals less than -120 dBm.

(f) Demodulator Subassembly -

(1) Command Bit Error Rate 1 x 10^{-5} BER at -95 to -30 dBm, TBD modulation index.

(2) Command Decoder Squelch - All command output shall be placed at a zero logic level when squelch is imposed on the basis of the subcarrier signal level. Time to deactivate the squelch shall be 15 milliseconds after the 70 KHz subcarrier is no longer being received.

3.7.1.1.5.1.1.4 Command Receiver (STDN Relay - TDRS)

The command receiver shall be part of an S-Band TDRS Transponder and shall consist of the following major subassemblies:

- RF/IF
- Code Tracking Loop
- Carrier Tracking Loop
- Demodulator/Decoder
- Doppler Processor
- Local Oscillator.

(a) Receiver RF Input - A phase modulated reception capability shall be provided for any STDN relay forward channel in the 2025 to 2120 MHz band. The channel tolerance and doppler capability shall be ± 150 KHz. The specified performance shall be met over an input from TBD dBm to TBD dBm.

(b) Input Signal - Standard STDN PCM/PSK (summed) command signals combined with a PN code on a PM carrier shall be accommodated. The command rate shall be 2000 bps and a PN code rate of 6 M Chips/sec.

(c) Noise Figure - The noise figure of the receiver, measured at the input connector shall not exceed 7.5 dB for input signals less than TBD dBm.

(d) Demodulator - The command bit error rate shall be 1×10^{-5} at a signal level of TBD dBm and TBD modulation index.

(e) Decoder - Code Rate: 1/2

- Constraint Length: K = 7 bits
- Path Selection: Most likely according to metrics
- Metric Storage: 4 bits with clamping
- Decoder Input Quantization: 3 bits
- Path Delay: 5 constraint lengths

3.7.1.1.5.1.1.5 RF Coupler (3dB Hybrid)

(a) VSWR, 1.2:1 to maximum reference to 50 ohms at any part with other parts terminated.

(b) Insertion loss, 3.4 dB maximum at f_r ±10 MHz (includes 3 dB power split).

(c) Isolation, 30 dB minimum between two input ports.

3.7.1.1.5.1.1.6 Diplexer (Receiver Channel)

(a) VSWR, 1.2 to 1 maximum referenced to 50 ohms.

(b) Insertion Loss; 0.8 dB maximum at f_r ±10 MHz.

(c) Receive/Transmit Channel Isolation, 60 dB minimum.

3.7.1.1.5.1.2 Telemetry

The telemetry RF equipment of the C&DH module shall provide modulation and transmission of telemetry to the STDN direct and relay (TDRS) at S-band.

3.7.1.1.5.1.2.1 Link Considerations

3.7.1.1.5.1.2.1.1 STDN Direct

The Spacecraft telemetry link shall transmit telemetry data to STDN direct with design margin of 6 dB above the signal level required for a 10^{-5} bit error rate under the following conditions:

(a) Frequency: TBD MHz in the 2200 to 2300 MHz range

(b) Ground Antenna Size: 30-ft dish

(c) Minimum Elevation Angle of Ground Antenna: 5^{o}

(d) CCIR Power Flux Density Limits:

- Elevation of User $<5^{o}$, - 154 dBw/4 KHz/M^2.
- Elevation of User $>5^{o}$, $<25^{o}$, $-154 + \frac{\theta - 5^{o}}{2}$, θ = elevation angle
- Elevation of User $>25^{o}$, -144 dBw/KHz/M^2

(e) Atmosphere Loss: 0.6 dB

(f) Polarization: RHCP

(g) Maximum Slant Range: 3040 Km

(h) E/No Required: 12 dB

(i) Data Rates: Low - Selectable, 32 Kbps, 16 Kbps, 8 Kbps, 4 Kbps, 2 Kbps, 1 Kbps
Medium - 128 Kbps

3.7.1.1.5.1.2.1.2 STDN Relay (TDRS)

The Spacecraft telemetry link shall transmit telemetry data to STDN relay (TDRS) with a design margin of 3 dB above the signal level required for a 10^{-5} bit error rate under the following conditions:

(a) CCIR Power Flux Density Limits:

- Elevation of User $<5^{o}$, -154 dBW/4KHz/M^2
- Elevation of User $>5^{o}$, $<25^{o}$, $-154 + \theta - 5^{o}$, θ = elevation angle
- Elevation of User $>25^{o}$, - 144 dBW/4 KHz/M^2

(b) Polarization: RHCP

(c) Narrow Band Data Rate: Selectable, 8 Kbps, 4 Kbps, 2 Kbps or 1 Kbps.

(d) Data Modulation: PN Spread Spectrum, PSK ($\pm 90^{o}$), biphase

(e) PN Chip Rate: 6.0 Mchips/sec

(f) Data Coding:

- Convolutional Encoding, Rate (R) = 1/2,
- Constraint length (K) = 7, FEC Gain = 5.2 dB

(g) Maximum Range (LOS): 42,000 Km

(h) E/No Required, PSK: 9.9 dB

(i) TDRS Antenna Gain: 36 dB

(j) Pointing Loss: 0.5 dB

(k) Polarization Loss: 0.5 dB

(l) TDRS Ts: $824^{o}K$

(m) Transponder Loss: 2.0 dB

(n) Demodulation Loss: 1.5 dB

(o) EOS EIRP Required: 8.0 dBW

(p) Frequency: TBD (2200 to 2300 MHz range)

3.7.1.1.5.1.2.2 Telemetry Antenna(s)

The telemetry antenna(s) shall be integrated into a dual frequency command/telemetry antenna assembly.

(a) Frequency - Each antenna shall transmit at a frequency f_t ± MHz in the STDN downlink band of 2200 to 2300 MHz

(b) Coverage - Each antenna shall provide the following coverage at the specified frequency when the antenna is mounted on the EOS vehicle.

- Conical sector of 130° with a minimum gain of +3 dB at 40° to 65° off-axis and a minimum gain of -6 dB increase to +3 dB from foresight to 40° off-axis.

(c) Gain Reference: Right hand circularly polarized isotropic radiator

(d) Polarization: Each antenna shall transmit right hand circular polarization

3.7.1.5.1.2.3 Telemetry Transmitter (STDN Direct)

The telemetry transmitter shall be part of the S-Band integrated transponder and shall have two major subassemblies:

Baseband Subassembly - Shall combine ranging signals, medium rate PCM signals or an internally generated 1.024 MHz subcarrier modulated by low rate PCM. The combined baseband shall be level controlled, filtered and routed to the transmitter subassembly.

Transmitter Subassembly - Shall accept the baseband signal and phase modulate a coherent uplink derived signal which is subsequently multiplied and amplified to the required RF frequency and power level.

(a) Frequency - Output frequency shall be selectable as one of the STDN downlink channels in the 2200 to 2300 MHz frequency range. The frequency shall remain within ±0.0001% of the assigned frequency during all environmental testing.

RF Power Output - Minimum power output under worst-case specified environment and a 24 VDC input voltage shall be 2 W for any medium data rate mode and 0.2 W for the low data rate mode. The maximum RF power output shall be such that the CIRR requirements of Paragraph 3.7.1.1.5.1.2.1 are not exceeded. Rated power shall be provided into a load of 50 ohms at a maximum VSWR of 1.5:1 at any phase angle. No damage to the transmitter shall occur if the load isopen or shorted.

(c) Modulation Type - Phase modulation shall be employed. Polarity shall be such that a negative-going voltage produces leading phase.

(1) Low Rate PCM - The low rate PCM shall be bi-phase modulated onto a crystal-controlled subcarrier having a center frequency of 1.024 MHz.

(2) Medium Rate PCM - The medium rate PCM shall directly bi-phase modulate the RF carrier.

(3) Low Rate PCM and Medium Rate PCM - With simultaneous inputs to the low rate and medium rate PCM channels so as to form a common baseband in the summation amplifier; the transmitter output modulation index shall be equal to the sum of the individual modulation indices over the combined range of TBD radians with an accuracy of ±TBD percent.

(d) Data Rates -

(1) Low Rate: selectable, 32 Kbps, 16 Kbps, 8 Kbps, 4 Kbps, 2 Kbps, or 1 Kbps

(2) Medium Rate: 128 Kbps.

(e) Harmonic Distortion - Harmonic Distortion shall not exceed 3% of 100 KHz and 500 KHz, at a modulation index of 1.0 radian.

3.7.1.1.5.1.2.4 Telemetry Transmitter (STDN Relay-TDRS)

The telemetry transmitter shall be part of an S-Band TDRS transponder and shall consist of the following major subassemblies:

Transmitter Subassembly - Shall accept the baseband signal and phase shift key a coherent uplink derived signal which is subsequently multiplied and amplified to the required RF frequency and power level.

Baseband Subassembly - Shall combine ranging signals and PCM data. The combined signal shall be level controlled, filtered and routed to the transmitter subassembly.

(a) Frequency - Output frequency shall be selectable as one of the TDRS return channels in the 2200 to 2300 MHz frequency range.

(b) RF Power Output - Minimum power under worst-case specified environment and a 24 VDC input voltage shallbe 10 W. Rated power shall be provided into a load of 50 ohms at a maximum VSWR of 1.5:1 at any phase angle. No damage to the transmitter shall occur if the load is open or shorted.

(c) Modulation Type: Phase modulation shall be employed

(1) Low Rate PCM - The low rate PCM shall be capable of bi-phase modulating the carrier.

(d) Data Rates:

(1) Low Rate: selectable, 8 Kbps, 4 Kbps, 2 Kbps, or 1 Kbps

(e) Encoding: Convolutional Encoding

- Code Rate (R): 1/2

- Constraint Length (K): 7

3.7.1.1.5.1.2.5 Diplexer (Transmit Channel)

(a) VSWR: 1.2 to 1 maximum reference to 50 ohms

(b) Insertion Loss: 0.5 dB maximum @ f_t ± MHz

(c) Receive/Transmit Channel Isolation: 60 dB minimum

3.7.1.1.5.1.2.6 RF Coaxial Transfer Switch

The RF Coaxial Switch shall be a latching type switch with a position indicator circuit.

(a) VSWR: 1.2 to maximum reference to 50 ohms

(b) Insertion Loss: 0.2 maximum at f_t ± 10 MHz

(c) Isolation: 60 dB minimum

(d) Switch Time: 20 ms maximum (RF to RF)

3.7.1.1.5.1.3 Ranging

The ranging equipment shall receive and coherently retransmit STDN direct and/or relay (TDRS) ranging signals. The turnaround ranging channel shall utilize the receiver and transmitter equipment used for command and telemetry.

Input power and temperature transponder calibration curves for delay shall be used to reduce the uncertainty in time delay at 500 KHz to less than five nanoseconds as a goal.

The S/C ranging channel shall meet the STDN's modulator index interface requirements (uplink and downlink).

3.7.1.1.5.1.3.1 Link Considerations

3.7.1.1.5.1.3.1.1 STDN Direct

The Spacecraft ranging link shall transpond STDN direct ranging signals with a minimum design margin of 6 dB at the MFR specified sensitivity and tracking bandwith into the STDN's 30 foot antennas under worst-case link conditions. Link parameters are as stated in Paragraphs 3.7.1.1.5.1.1.1 and 3.7.1.1.5.1.2.1.1.

Ranging signals will consist of a number of different frequency tones phase modulated on the uplink carrier. The Spacecraft receiver-transmitter will transpond the signals and the ground ranging system will determine range by measuring the phase shift of the tones. The ranging frequencies are 500 KHz, 100 KHz, 20 KHz, 4 KHz, 300 Hz, 160 Hz, 40 Hz and 10 Hz.

3.7.1.1.5.1.3.1.2 STDN Relay (TDRS)

The Spacecraft ranging link shall transpond STDN relay (TDRS) ranging signals with a minimum design margin of 3db at the TDRS ground station under worst-case link conditions. Link parameters are as stated in Paragraph 3.7.1.1.5.1.1.1.1 and 3.7.1.1.5.1.1.1.2.

The ranging signal will consist of a PN spread spectrum coded signal that is transponded at the Spacecraft. Measurement of propagation time will result in Spacecraft range. Doppler information will be obtained by using a reconstructed carrier component of the Spacecraft transmitted signal.

3.7.1.1.5.1.3.2 Ranging Antennas

The ranging antennas shall be dual frequency and are defined in Paragraphs 3.7.1.1.5.1.1.2 and 3.7.1.1.5.1.2.2.

3.7.1.1.5.1.3.3 Ranging Receiver (STDN Direct)

The ranging receiver shall be part of the S-Band integrated transponder as defined in Paragraph 3.7.1.1.5.1.1.3. In addition, it shall include the following:

(a) Input Signal - Standard STDN tone ranging signals shall be accommodated consistent with the ground ranging equipment capabilities.

(b) Coherent Drive - The receiver shall be capable of furnishing a coherent drive signal to the transmitter. The output frequency shall be 1/120 of the transmitter exciter output frequency.

(c) Output Signal - The receiver shall provide a demodulated ranging signal to the transmitter.

3.7.1.1.5.1.3.4 Ranging Receiver (STDN Relay - TDRS)

The ranging receiver shall be part of the S-Band TDRS transponder as defined in Paragraph 3.7.1.1.5.1.1.4. In addition, it shall include the following:

(a) Input Signal - Standard STDN/TDRS PN ranging signals shall be accommodated consistent with the ground ranging equipment capabilities.

(b) Coherent Drive - The receiver shall be capable of furnishing a coherent drive signal to the transmitter.

(c) Output Signal - The receiver shall provide a demodulated ranging signal to the transmitter.

3.7.1.1.5.1.3.5 Ranging Transmitter (STDN Direct)

The ranging transmitter shall be part of the S-Band integrated transponder as defined in Paragraph 3.7.1.1.5.1.2.3. In addition, it shall include the following:

(a) Utilize a coherent drive signal from the receiver

(b) Ranging Signal - The transmitter shall be capable of accepting a tone ranging signal from the receiver, combining it with low rate PCM data and amplifying the resultant signal to 2 watts at S-Band.

3.7.1.1.5.1.3.6 Ranging Transmitter (STDN Relay-TDRS)

The ranging transmitter shall be part of the S-Band TDRS transponder as defined in Paragraph 3.7.1.1.5.1.2.4. In addition, it shall include the following:

(a) Utilize a coherent drive signal from the receiver.

(b) The transmitter shall be capable of accepting a PN ranging signal from the receiver, combining it with PCM data and amplify the resultant signal at S-Band.

3.7.1.1.5.1.3.7 RF Coupler (3-dB Hybrid)

The RF coupler shall be identical to that described in Paragraph 3.7.1.1.5.1.1.4.

3.7.1.1.5.1.3.8 Diplexer

The diplexer shall be identical to that described in Paragraph 3.7.1.1.5.1.1.5 and 3.7.1.1.5.1.2.4.

3.7.1.1.5.1.3.9 RF Coaxial Transfer Switch

The RF coaxial switch shall be identical to that described in Paragraph 3.7.1.1.5.1.2.5.

3.7.1.1.5.2 Data Handling Group (DHG)

The DHG of the C&DH module acquires, processes, records, formats and routes data/commands from/to the appropriate EOS subsystem modules.

3.7.1.1.5.2.1 On-Board Computer

A general purpose digital computer shall be included in the C&DH module. Computer major elements shall include nonvolatile memory, central processing unit, input/output unit, internal timing unit and power regulator unit. The computer shall communicate with all Spacecraft subsystems and devices through time shared use of the telemetry and command group. Computer characteristics which are considered essential are as follows:

Computer Characteristics	Essential
Minimum Word length (BITS)	16
Required Memory Size (K words)	16
Memory Module Size (K WORDS)	8
Maximum Memory to Register Add time (μ sec)	5
Maximum Memory to Register Multiply time (μ sec)	50
Maximum Memory to Register Divide time (μ sec)	80
Number of Index Registers	1
External interrupts	8
Internal interrupts	1
Instruction types:	add, multiply, divide and, or, exclusive or, complement branch, conditional branch branch and mark read, store
Input/Output Channels (serials	8

3.7.1.1.5.2.2 Command Decoder

3.7.1.1.5.2.2.1 Bit Rates, Coding and Formats

3.7.1.1.5.2.2.2 Input Commands from Ground

The command rate from the receiver/demodulator shall be 2Kbps NRZ-L. Each command message shall be composed of 40 bits which shall be defined in accordance with Fig. 3-15. For synchronization, each single command or each sequence of commands will be preceded by an introduction and synchronization code as shown. For the case of delayed commands, two commands are required: The first (Part 1) contains time tag and the second (Part 2) contains the command itself. Since the Spacecraft address bits are unique assignments for each mission, the command decoder must readily accommodate the different address possibilities.

3.7.1.1.5.2.2.3 Output Commands to Computer

The command rate at the output to the computer shall be a 2Kbps Bi-phase L encoded.

Each command message shall be composed of 28 bits. Twenty four bits are defined as the command or computer load data segment of the ground input command word format shown in Fig. 3-15. This segment will be proceeded by four Bi -L synch bits defined bits defined below:

Bit Pos	Bit Qty	Function (Word Synch)
0-2	3	1 1/2 bits (+), followed by 1 1/2 bits (-1)
3	1	Fixed Logical "I"

3.7.1.1.5.2.2.4 Output Commands to the Data Bus Controller/Formatter

The command rate at the output to the Data Bus Controller/Formatter shall be 50 commands/sec. The Data Bus Controller/Formatter will strobe the command decoder once every 16 milliseconds. Data transfer rate will be 1.024 Mpps, Bi-phase L encoded.

Each command message shall be composed of 28 bits. Format is same as that defined in Paragraph 3.7.1.1.5.2.2.2 for output commands to computer.

3.7.1.1.5.2.2.5 Command Execution Rate

The command decoder shall be capable of executing 50 commands per second from the ground.

Probability of executing a false command shall be less than 1×10^{-10} for any input signal condition.

3.7.1.1.5.2.3 Multiplex Data Bus System

The Data Bus System characteristics are separated into three major sections:

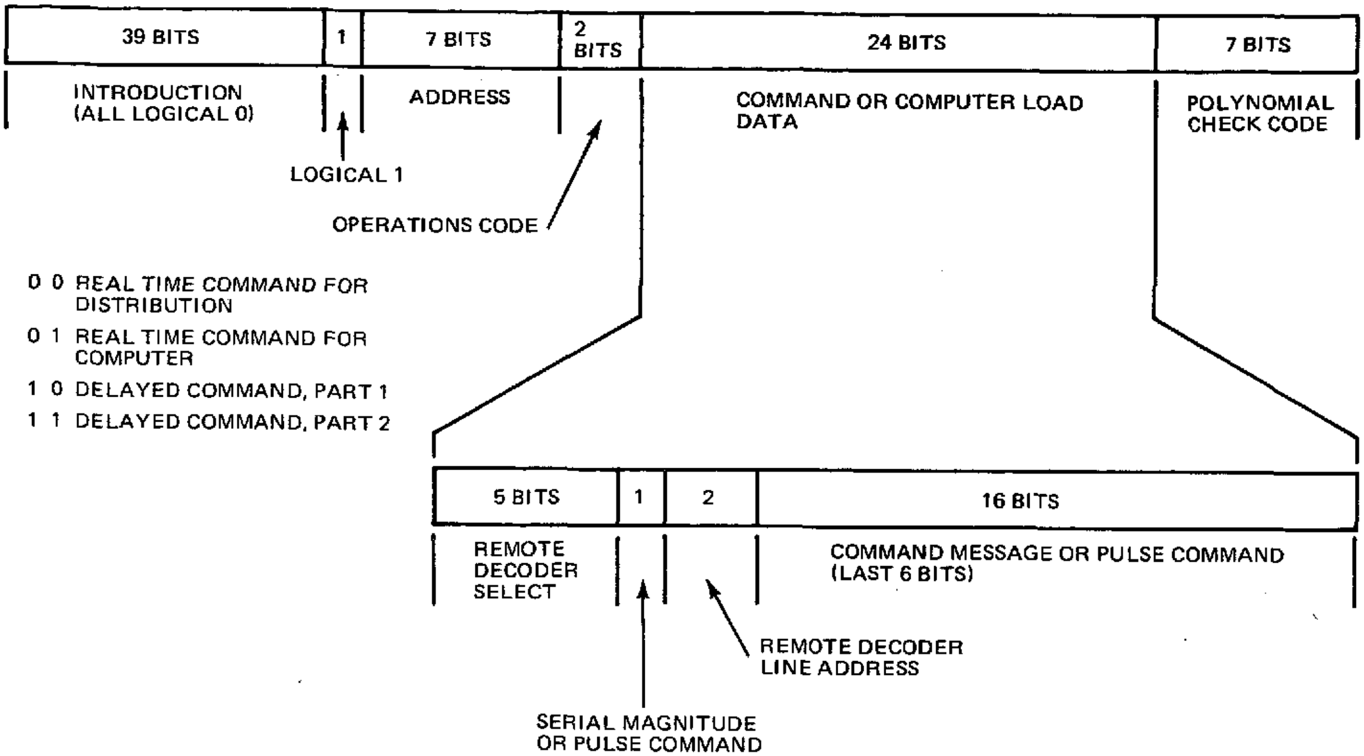
- Multiplex Data Bus Characteristics
- Characteristics unique to the Bus Controller/Formatter
- Characteristics unique to the Remote Units

3.7.1.1.5.2.3.1 Bus Characteristics

(a) Transformer Coupled

(b) Full Duplex System (shielded twisted pair = 1 bus)

- One bus line for command and addresses from the Bus Controller/Formatter (This line is designated the Command and Address Line)
- One bus line for data return from Remote Units (This line is designated Reply Line)



(1) 5-34

Fig. 3-15 Command Word Format

(c) Bit Rate: 1.024 Mbps

(d) Code: Biphase L coded data (Manchester type II)

(e) Word Sync: 3 Bits Illegal Manchester followed by Logical "1".

(f) Word Size: 32 Bits on Command & Address line
12 Bits on Reply Line (8 Bits of Data)

(g) Word Rate: 32 KHz

(h) Response Time *: 64 to 66 u sec.

(i) Clock on Command and Address Line is continuous

(j) Manchester data on Reply Line is phased relative to Command and Address Line bit rate

(k) Up to 64 Remote Units may be tied on bus

*Response Time is defined as the time from the end of the message parity bit to the start of the return data sync word.

(1)5

(l) Multiplex Data Bus Format From Controller/Formatter to Remote Unit

Bit Pos	Bit Qty	Standard Header Word Synch
0-2	3	1 1/2 bits (+), 1 1/2 bits (-)
3	1	Fixed Logical "1."
4-6	3	Specifies 1 of 8 message types
		Message Type I (Ser. Mag. CMD)
7-11	5	Specifies 1 of 32 Remote Units
12	1	Specifies Remote Unit A or B
13-16	2	Specifies 1 of 4 CMD Lines to User
17-22	16	Specifies Magnitude CMD Value
23-30	1	Parity
31		
		Message Type II (Pulse CMD)
7-11	5	Specifies 1 of 32 Remote Units
12	1	Specifies Remote Unit A or B
13-16	4	Not used
17-22	6	Selects 1 of 64 Outputs
23-30	8	Not Used
31	1	Parity
		Message Type III (TM Address)
7-11	5	Specifies 1 of 32 Remote Units
12	1	Major Frame Indicator
13	1	Minor Frame Indicator
14	1	TM Word Indicator
15-16	2	Specifies 1 of 4 Signal Types
17-22	6	Selects 1 of 64 Inputs
23-26	4	Allows Expansion to 1024 Inputs
27-30	4	Not Used
31	1	Parity

(m) Command/Address Line Time Slot Allocations:

(1) Every other 32 bit time slot on Command/Address Line is allocated to transmission of MUX*Channel Addresses (16 K addresses/sec).

a. Every other MUX Channel Address slot is allocated for TM transmission (8 K addresses or 64 Kbps max).

b. Every other MUX Channel Address slot is allocated for computer use as required.

(2) Every other 32 bit time slot on Command/Address Line is allocated to transmission of commands.

a. Ground commands can only occupy every 256th slot (one per 16 ms).

b. Pulse commands from computer can occupy only one slot per 16 ms.

c. Serial magnitude commands from computer can occupy TBD slots per sec.

(n) Redundancy - Busses to be redundant, Bus Controller/Formatter to transmit on selected Command/Address Bus - Remote Unit to transmit on both Reply Data Busses.

*MUX: Multiplexer section of remote unit

3.7.1.1.5.2.3.2 Bus Controller/Formatter Characteristics

(a) Command Execution Rate - The Bus Controller/Formatter shall be capable of distributing 62.5 commands per second from the computer while simultaneously executing 50 commands per second from the ground.

(b) Bus Data Rates - Bus Controller/Formatter shall be capable of acquiring up to 32 Kbps data for the computer while simultaneously acquiring up to 32 Kbps of data for transmission to the ground. The 32 Kbps of telemetry data shall also be fed to the computer.

(c) Telemetry Output Format - Telemetry output data rates shall be command selectable at 32, 16, 8, 4, 2, and 1 Kbps. The telemetry format shall be structured in minor frames of 128 eight bit words. For the baseline C&DH subsystem, the telemetry format shall be controlled by the computer and as a minimum each minor frame shall contain synchronization code, Spacecraft time, command verification data, and the four subcommutator words. Capability shall exist for dwelling on the subcommutator as well as any minor frame word.

(d) Spare Outputs - The unit shall have 2 spare data outputs with voltage levels defined below:

- Logical "1"	+12 to +17V @ 4 ma
- Logical "0"	0 to +.5V

3.7.1.1.5.2.3.3 Remote Unit Requirements

The remote units shall be capable of providing the following signal input and output interface to users.

(a) Remote Multiplexer

Each multiplexer (section) shall have 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 16 inputs for serial digital signals.

All inputs of the multiplexer shall have an input impedance of 10 megohms minimum in the normal mode and 10K ohms minimum during sampling. The multiplexer shall be capable of surviving a short circuit of +35 VDC maximum on any one input for an indefinite time.

(1) Analog Inputs (Digitized to 8 bits)

Range	0 to +5 VDC
Z Source	5K ohms maximum
Accuracy	+30 mv

(2) Bilevel Digital Inputs

Logical "1"	+3.5 to +35 VDC
Logical "0"	-1.0 to +1.5 VDC
Fault Tolerance	-20 to +40 VDC
Z Source	5K ohms minimum; 10 ohms maximum

(3) Serial Digital Inputs (8 bits/word)

Clock Rate	64 KHz
*Gate Width	Envelopes 8 clock pulses
Input Data	
Logical "1"	+3.5 to +12 volts
Logical "0"	-1.0 to +1.5 volts
Z Source	500 ohms maximum

*These signals are multiplexer outputs with the same voltage and impedance characteristics as those shown for pulse commands in the following paragraph.

(b) Remote Decoder

Each remote decoder (section) shall have 64 pulse command ouputs and 4 serial magnitude command outputs. The relay drivers shall be packaged in groups of 4 and shall not necessarily be contained in the decoder housing. Pulse commands shall serve as relay driver inputs.

(1) Pulse Commands*

Pulse Duration	4 ms minimum
Logical "1"	+12 to +17V @ 4 ma
Logical "0"	0 to +.5V
R Source @ "0"	8.0K ohms maximum

(2) Magnitude Commands*

Clock Rate	16 KHz
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.

(3) Relay Driver Commands

Level	24 ±4 VDC @ 50 ma
Duration	100 ms minimum
Ground	Relay coil return provided at driver

(c) Remote Unit Alternate Power Configuration - Remote units as per configuration No. 1 & No. 2 power strobed for economy with 16 KHz square wave TBD volts ±2%. Power "ON" only when required.

3.7.1.1.5.2.4 Spacecraft Clock

The Spacecraft clock shall contain the frequency source for timing of C&DH functions and all other Observatory functions as required. The primary clock frequency of 4.096 MHz shall be generated by an oscillator with a stability of ±1 part in 10^6 per day.

The clock unit shall contain a frequency divider chain and clock driver circuitry which will specify 24 bits of Spacecraft time (LSB = 1.024 seconds) and make a variety of clock signals available to other Spacecraft subsystems. As a minimum, these clock signals shall be 4.096 MHz, 2.048 ,Hz and 64 KHz. The 2.048 MHz signal shall be distributed via a differential line driver feeding a balanced two-wire line. Drivers requiring a signal return through signal ground shall not be allowed.

3.7.1.1.5.2.5 Signal Conditioner Unit

The Signal Conditioning Unit shall provide circuitry required to:

(a) Condition to proper level, form and mode, all pick-up point signals which are not normalized.

(b) Provide buffering and isolation when required by the design limitations of the monitored equipment.

(c) Route all signals specified in (a) and (b) to their proper destinations.

(d) Provide excitation signal power to remote sensors as required.

The quantity of circuit types and ranges shall be in accordance with the EOS measurement/command list.

The Signal Conditioning Unit shall contain, but is not limited to, a combanation of the following signal conditioning input types:

(a) DC Amplifier

(b) DC Attenuator

(c) Signal Isolation Buffer

(d) Variable Resistance to DC Voltage Converter

(e) Discrete (Bilevel) Signals

In addition, any pulse stretchers, latching relays and digital to analog converters required for command implementation may be located in this assembly.

3.7.1.1.5.2.6 Sensors

Sensor types which satisfy the requirements of the Observatory Measurements List will be provided to convert the measurement to a compatible signal for data processing. The sensors consist of two basic designs that contain either integral electronics or are powered by the sensor excitation power provided by the Signal Conditioner Unit.

3.7.1.1.6 Physical Requirements

The C&DH shall be housed in a standard module as specified in Paragraph 3.7.1.4 excluding components external to the standard module (eg. antenna). Component physical requirements shall be specified in Spec EOS-SS-200. The weight allocation for the C&DH is specified in Paragraph 3.2.2.1.1.

3.7.1.1.7 Interface Requirements

3.7.1.1.7.1 Mechanical Interfaces

The mechanical interfaces between the C&DH module and the Spacecraft structure shall be in accordance with Paragraph 3.7.1.4

3.7.1.1.7.2 Thermal Interfaces

The thermal design shall maintain all of the C&DH subsystem equipment within the limits specified for all mission phases. The general heat sink operating temperature range shall be $21^{\circ}C \pm 11^{\circ}C$. Specific components requiring deviations from this value and temperatures for all modes shall be as specified in Paragraph 3.7.1.5.

The C&DH subsystem module shall be designed to reject all equipment heat dissipation to space. The thermal design shall minimize the module heat sink gradients. The C&DH subsystem module shall be made thermally independent from other subsystem modules and from the mounting structure, by using insulation and low conductance mounts.

3.7.1.1.7.3 Structural Interfaces

The structural interfaces between the C&DH module and the Spacecraft shall be in accordance with Paragraph 3.7.1.4

3.7.1.1.7.4 Electrical Interfaces

All C&DH module electrical interfaces with the Spacecraft shall be via the Spacecraft Interface Connector(s) Umbilical Connection interfaces with prelaunched and resupply equipment shall also be via the Spacecraft Interface Connector(s). Electrical interfaces, for prelaunch test operations only, shall be made via the Test Connectors. All electrical interfaces between assemblies and interface connectors within the C&DH Module shall be made with the C&DH module harness.

3.7.1.1.7.4.1 Connectors

(a) Spacecraft Interface Connector - The Spacecraft interface connector shall provide the electrical connect/disconnect between module and Spacecraft. The connector (s) shall be designed for and physically positioned to assure interchangeability of modules. Specific design requirements shall be: blind mate capability; anti-bind roll-off shell (angular disconnect capability); maximum axial movement of structure without affecting continuity; and highly reliable contacts with self-alligning capabilities.

(b) Test Connectors - Test connectors shall be provided as applicable on major assemblies and at one side of the subsystem module. The test connectors shall provide the capability, to the maximum extent possible, to determine degradation or the flight worthiness of the assemblies and module without the need for demating connectors in flight circuits. All outputs to test connectors shall contain isolation circuitry.

3.7.1.1.7.4.2 Harness

The module harness shall provide all electrical interfaces between subsystem assemblies within the module and to the module/structure interface 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 subsystem electrical assemblies. It shall be possible to remove electrical assemblies without removing the harness. Harness and cable assembly practices shall meet the intent of MIL-W-5088 unless specified otherwise in this specification.

Wire types shall be lightweight, abrasion resistant, and space qualified. Coaxial cable shall conform to MIL-C-17. Wire size shall be determined by : Circuit steady state current; voltage drop compatible with unit performance requirements; thermal environment; connector termination capabilities; minimum wire gauge 24 awg high strength copper alloy; bundle capacity; and minimum weight. Connectors shall be of the removal crimp contact type where feasible and shall meed environmental requirements for space application.

3.7.1.1.7.4.3 Power

The electrical power required to operate the C&DH Module will be provided by the Power Module. The prime power shall be at a voltage of +28 ±7 VDC with detailed characteristics as defined in Specification EOS-SS-250. Any conditioning of the nominal +28 VDC power to other power types, voltage levels or regulation tolerances shall be provided within the C&DH module.

The C&DH module power distribution circuitry shall contain devices to protect the power busses from short circuits. The bus protection circuitry shall be provided for all loads except those which are non-redundant and critical to mission success. Protected loads and detailed Bus Protection requirements shall be in accordance with Specification EOS-SS-250.

3.7.1.1.8 Instrumentation Requirements - Telemetry

The C&DH system shall be instrumented with telemetry circuitry which can indicate and locate operating modes or failures on a system or component level.

This telemetry shall include, but not limited to the following functions:

- Commandable modes
- RF power output
- Receiver signal quality
- Temperature
- Power Converter Voltages

The electrical interface characteristics for the telemetry circuitry is specified in Paragraph 3.7.1.1.

3.7.1.1.9 Test Point Connectors

Test point connectors shall be provided as applicable on major assemblies and at one side of the subsystem module. The test connectors shall provide the capability, to the maximum extent possible, to determine degradation or the flight worthiness of the assemblies and module without the need for demating connectors in flight circuits. All outputs to test connectors shall contain isolation circuitry.

3.7.1.1.10 Ground Support Equipment

The C&DH will require a S/S Module Checkout Test Bench capable of powering the module, performing all tests required for the integration of its components and for its

C-2

acceptance test. The bench shall be sufficiently flexible to provide for maintenance of the module after its removal from the EOS in a failed mode. In addition, the flexibility shall extend to permit the use of the bench within the NASA LPS (Launch Processing System) during Shuttle operations.

The bench shall be designed such that bench failure shall not induce failures within the module. (Refer to Paragraph 3.1.3.1 for additional GSE.)

3.7.1.2 ELECTRICAL POWER SUBSYSTEM

3.7.1.2.1 General Requirements

This specification outlines the requirements for an Electrical Power Subsystem (EPS) which will be used to supply power to a modularized spacecraft. The EPS will consist of a Power Module and a mission peculiar solar array and array drive as required. The EPS shall be adaptable to a variety of missions ranging from near earth to synchronous altitudes.

3.7.1.2.2 Functions

The EPS shall provide for solar energy conversion, energy storage, power control, distribution and monitoring of unregulated +28 VDC power to the Spacecraft throughout the full duration of each mission. Provisions shall be included for powering of the Spacecraft from external power sources during any mission phase from prelaunch (ground) operations through on-orbit resupply or retrieval operations. Any conditioning of the unregulated 28 VDC power to other power types, voltage levels or regulation tolerances shall be provided within the subsystems or payloads.

3.7.1.2.2.1 Solar Energy Conversion

Photovoltaic devices (solar cells) shall be used as the primary source of electrical energy for the Spacecraft through the full duration of each mission. The solar cell array and any associated functions and mechanisms shall be mission peculiar and optimized to specific mission requirements.

3.7.1.2.2.2 Energy Storage

Secondary (rechargeable) batteries shall be used to supplement solar array power during those mission phases or orbital periods when solar array power is unavailable or insufficient to support the Spacecraft load. The batteries shall be charged whenever solar array power capability exceeds the Spacecraft load demand.

3.7.1.2.2.3 Power Control

The EPS shall include positive provisions for control of the solar array power output and battery discharge/charge. Automatic control functions shall maintain safe conditions within the EPS during all phases of the mission. Command capability shall exist for overriding all automatic functions except those considered necessary for normal operations or survival of the Spacecraft during emergency or abnormal conditions.

Solar array control functions shall allow efficient utilization of available solar array energy and shall include positive means of limiting maximum voltages impressed upon the batteries and Spacecraft busses.

Battery control shall be such to maintain a safe state-of-charge in the batteries for normal operations. Abnormal battery conditions which could be considered unsafe or which can affect the capability of the batteries in satisfying the mission shall be automatically detected and appropriate corrective action initiated. Command override of the automatic functions shall be provided.

Battery charge control techniques shall be limited to reliable flight proven methods.

3.7.1.2.2.4 Power Distribution

The EPS shall contain the necessary functions to control and distribute to the spacecraft subsystems and payloads, unregulated +28 VDC power supplied by ground or shuttle based power equipment or power derived from the spacecraft solar array and/or batteries.

The distribution circuitry shall include protective features that eliminate single point failures in the power distribution network. During ground test and orbital resupply or retrieval operations, a hard line control capability to arm and disable the power input/output circuitry to the power module shall be provided.

3.7.1.2.2.5 Monitoring

Provisions shall be incorporated to determine and evaluate the EPS flight worthiness, degradation, status and performance during ground, flight and resupply or retrieval operations. In-flight telemetry data shall be primarily limited to those functions necessary to determine abnormal or emergency conditions and for control and operation. Hardline connections for test and integration shall provide the capability, to the maximum extent possible, to determine degradation or flight worthiness of the assemblies and subsystem without the need for demating connectors in flight circuits. Prelaunch and/or resupply operations monitoring shall be hardline connections via an umbilical circuit and shall be primarily limited to those functions necessary to insure safety of the subsystem and the Spacecraft.

3.7.1.2.3 Configuration

The EPS configuration shall include a mission peculiar solar cell array (refer to Paragraph 3.7.3.1.2.1) and associated mechanisms and a Power Module containing rechargeable storage batteries, circuitry for solar array control, battery charge control, power distribution and control, power and signal interfaces and command and telemetry equipment.

The EPS design shall utilize proven concepts and existing equipment designs, especially those with proven flight performance, to the maximum extent practicable within the constraints of the EPS performance requirements.

3.7.1.2.3.1 Basic EPS Configuration

The basic functional configuration of the EPS shall be as shown in Fig. 3-16. The basic EPS configuration requires that at least two and as many as six hermetrically sealed, rechargeable Nickel-Cadmium batteries be electrically paralleled with an electrically isolated section of the solar array and connected directly to the Power Module load bus without any active, series regulation elements.

The section of the solar array connected directly to the batteries/bus, hereafter called the Auxiliary Solar Array, shall only be used to supply the spacecraft load and shall not be allowed to recharge the batteries. Power Module circuitry shall be provided to assure positive voltage-limiting of the auxiliary solar array throughout the full duration of each mission.

Battery recharge power and the average Spacecraft load not supplied by the auxiliary solar array shall be provided by a second electrically isolated section of the solar array. The power output from this array section, hereafter called the Main Solar Array, shall be routed through a series, pulse-width-modulated buck regulator(s) which shall serve as a central battery charge controller.

The Power Module shall contain the distribution bus which supplies power to the subsystems and the payload. The distribution circuitry for internal power subsystem module loads shall contain devices to protect the power busses from short circuits. The bus protection circuitry shall be provided for all loads except those which are non-redundant and/or critical to mission success. For Observatory/Spacecraft load distribution, individual groups of power contacts shall be provided on the module/structure interface connector assembly for each of the major subsystems and for the payload. Bus protection for each of these loads shall be provided only in the using equipment during flight; however, bus protection circuitry shall be provided for use during the early phases of Spacecraft/Observatory integration and test.

The Power Module shall contain hardline control circuitry via the Spacecraft umbilical connector for arming and disabling the power input/output circuitry during ground tests and during orbital resupply or retrieval. The Spacecraft shall have the capability for being powered by ground based power supplies during test operations and Shuttle-based power sources during resupply or retrieval operations. Power inputs for this purpose shall be designed to include circuitry to protect the Spacecraft from shorts to ground on these power input lines. Power input shall be via the module/Spacecraft interface connector and the umbilical connector which will be mounted on the Spacecraft structure.

Command and telemetry interfaces to the communications and data handling subsystem shall be via remote command decoding and telemetry multiplexing circuitry housed in the power subsystem module. This circuitry will operate on a "party-line" principle to minimize module-to-module wiring interfaces and will be supplied to the Power Subsystem contractor from the Command and Data Handling Subsystem contractor.

3.7.1.2.3.2 EPS Configuration Options

The EPS design shall incorporate configuration options which will permit optimization of power capability, reliability/redundancy and weight according to specific mission requirements without redesign or significant modification of assemblies or harnesses.

3.7.1.2.3.2.1 Main/Auxiliary Solar Array Ratio

The basic Power Module configuration shall accommodate, without redesign, the capability to support none, part or all of the average Spacecraft load via the Auxiliary Solar Array. For those missions where the entire Spacecraft load is supplied by the Auxiliary Array, array control provisions may be physically located on the array. The ratio of power supplied to the Power Module from the Main and Auxiliary Solar Array shall be mission peculiar.

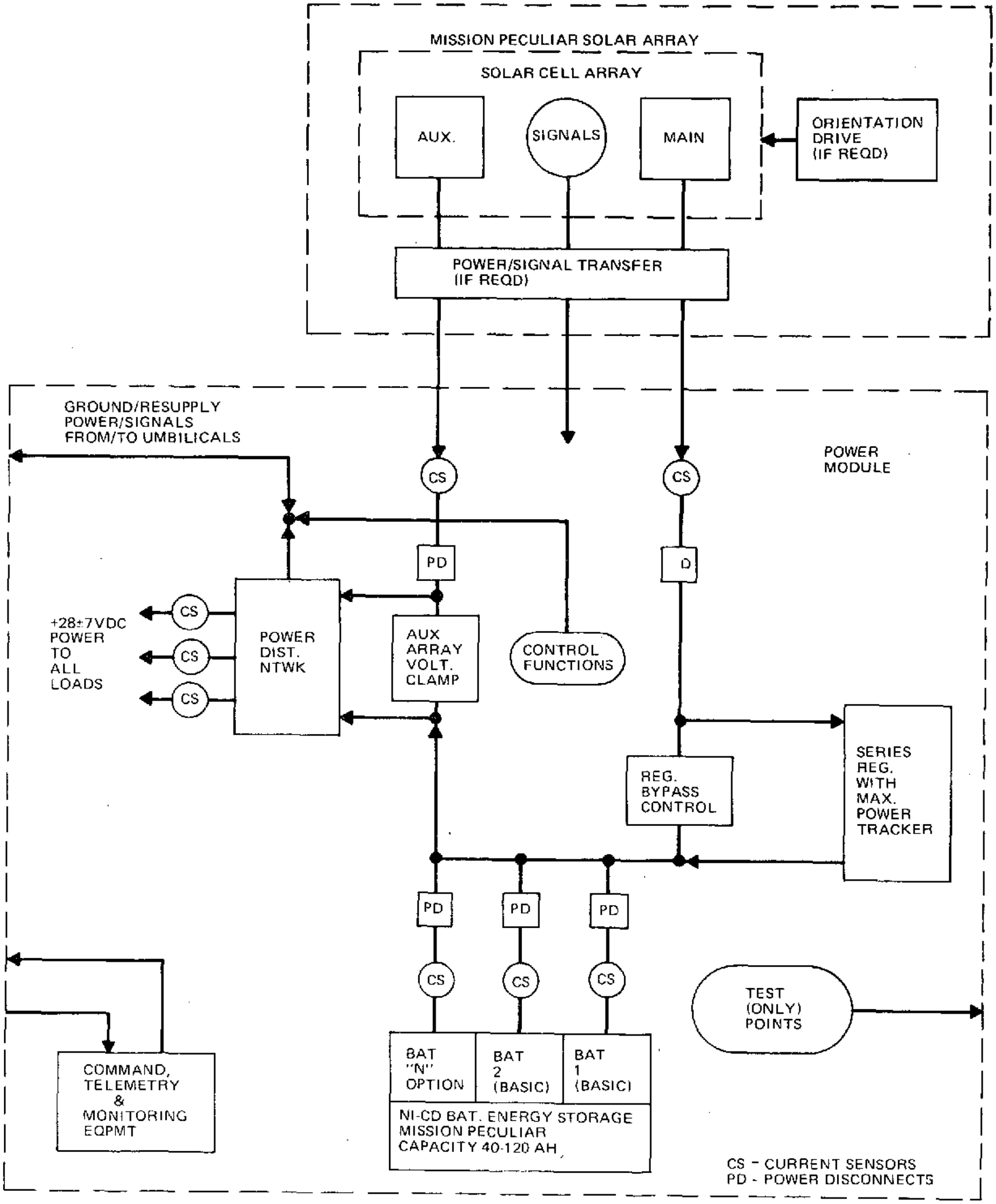


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Fig. 3-16 Basic Electrical Power Subsystem Functional Configuration

3.7.1.2.3.2.2 Battery Energy Storage Capacity

The batteries and charge control circuitry shall be of such a design to permit tailoring of the subsystem energy storage capability to the needs of specific missions. The minimum battery capacity shall be at least 40 Ampere-Hours and it shall be possible to increase system capacity to at least 120 Ampere-Hours. The interconnecting harness must be designed to handle full system capabilities but should be sectionalized to permit removal of unrequired sections where Spacecraft weight is critical.

3.7.1.2.3.2.3 Redundancy

Subsystem design shall be based on the use of parallel redundancy of the operational or standby types. The degree of redundancy provided shall be selectable on a mission-to-mission basis by the addition or omission of remote decoders, remote multiplexers, batteries and charge control equipment. The harness shall be designed to provide capabilities for operation in the fully redundant mode. Assemblies and possibly harness sections shall be omitted when lesser degrees of redundancy are required.

3.7.1.2.3.2.4 Battery Reconditioning

On-orbit battery reconditioning shall be a mission peculiar option of the EPS. Individual batteries shall be capable of being electrically isolated from the remaining "on-line" batteries and deep discharged and charged.

3.7.1.2.3.3 Electromagnetic Compatibility

The power subsystem module shall be designed to minimize the radiation of self generated noise and shall be shielded to preclude the possibility of susceptibility to EMI from Spacecraft or external sources. For specific missions it shall be possible to incorporate additional shielding to further reduce radiation or susceptibility. System design shall be based on the suppression of noise at its source and the containment of self-generated noise within the generating assembly. Good design practices in chassis design, EMC filtering, grounding, bonding etc. shall be employed through the program. Detailed EMC and grounding requirements are defined in Paragraph 3.3.2.

3.7.1.2.3.4 Connectors

3.7.1.2.3.4.1 Spacecraft Interface Connector

The Spacecraft interface connector shall provide the electrical connect/disconnect between module and Spacecraft. The connector(s) shall be designed for and physically positioned to assure interchangeability of modules. Specific design requirements shall be: blind mate capability; anti-bind roll-off shell (angular disconnect capability); maximum axial movement of structure without affecting continuity; and highly reliable contacts with self aligning capabilities.

3.7.1.2.3.4.2 Test Connectors

Test connectors shall be provided as applicable on major assemblies and at one side of the subsystem module. The test connectors shall provide the capability, to the maximum extent possible, to determine degradation or the flight worthiness of the assemblies and module without the need for demating connectors in flight circuits. All outputs to test connectors shall contain isolation circuitry.

3.7.1.2.3.5 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 subsystem electrical assemblies. It shall be possible to remove electrical assemblies without removing the harness. Harness and cable assembly practices shall meet the intent of MIL-W-5088 unless specified otherwise in this specification.

Wire types shall be lightweight, abrasion resistant, and space qualified. Coaxial cable shall conform to MIL-C-17. Wire size shall be determined by: circuit steady state current; voltage drop compatible with unit performance requirements; thermal environment; connector termination capabilities; minimum wire gauge 24 awg high strength copper alloy; bundle capacity, and minimum weight. Connectors shall be of the removable crimp contact type where feasible and shall meet environmental requirements for space application.

3.7.1.2.4 Modes

3.7.1.2.4.1 Battery Modes

Battery discharge will occur during all orbital periods when solar array energy is unavailable or insufficient to support the Spacecraft load demands. During normal eclipse (dark) periods, all batteries shall be operated in parallel, passively sharing the load power. Sunlight period peak load requirements in excess of the capability of the solar array shall be supplied by the batteries.

After passing through a discharge period, the batteries shall automatically enter a charge mode provided sufficient Main Solar Array power is available. The primary battery charge mode shall include an initial charge phase followed by a voltage limited, taper charge phase.

During initial charging, the batteries shall receive all available Main Solar Array energy in excess of the load demand. This phase of battery charging shall be terminated when the battery voltages rise to a preselected battery voltage limit. A minimum of four battery temperature-compensated-voltage-limits shall be provided with the operating level selectable by command. Upon reaching the selected voltage limit, the battery currents shall be reduced (tapered) such to maintain the battery voltage at the temperature compensated level.

For some orbital conditions or missions, it may be desirable to limit the battery overcharge currents or recharge energy input. Battery current controlled modes shall be available that allows either a battery trickle-charge or an "on-line float" mode.

3.7.1.2.4.2 Main Solar Array Control Modes

The Main Solar Array power output to the Power Module shall be controlled by an active series voltage regulator. When the Main Solar Array power available exceeds the battery and Spacecraft loads demands, the series regulator shall operate in a voltage limited mode. In this mode the output voltage of the series regulator shall be limited to a preselected level dictated by battery voltage and temperature characteristics.

At any time during the orbital sunlight period, when battery voltages are less than the preselected voltage limit, the series regulator shall be capable of automatically entering a mode where the Main Solar Array is operated at its maximum power point.

As an alternative to the maximum power tracking mode during initial battery charging, the series regulator shall be capable of being by-passed or shunted such that Main Solar Array power is transferred directly to the batteries and Spacecraft load with no active series regulation. This shunt mode shall be terminated when battery voltages reach the preselected voltage limit. The series regulator shall then assume control of the Main Solar Array and shall automatically operate in either the voltage limited or maximum power track modes.

3.7.1.2.4.3 Mission Peculiar Modes

The EPS shall be capable of special operational modes that are peculiar to specific mission requirements.

For missions that include Spacecraft/Observatory resupply or retrieval requirements, the EPS shall be capable of being commanded via the umbilical hardline interface into a quiescent mode where all input/output power is disabled. During this mode, Spacecraft power shall be supplied by Shuttle-based power sources.

For EPS configurations that include battery reconditioning provisions, this mode shall be utilized, as required, to maintain battery discharge voltages. Only one battery at a time shall be in the recondition mode with the remaining batteries "on-line" supporting the Spacecraft load.

3.7.1.2.5 Performance Requirements

The EPS shall be capable of satisfying the following minimum requirements throughout the full duration of each mission when matched to a suitable mission peculiar solar array. Detailed EPS performance requirements and characteristics shall be as specified in Specification EOS-SS-210.

3.7.1.2.5.1 Bus Characteristics

During normal flight operations the EPS shall supply, via a two wire distribution network, prime power at +28 ±7 volts VDC to all Spacecraft subsystems and payloads.

3.7.1.2.5.2 Power Output

The power output capability of the EPS shall be mission peculiar. The Power Module shall be capable of being adapted to satisfying, as a minimum, orbital average and peak power requirements in the range of;

(a) Orbital average: 400 to 1500 W

(b) peak loads: up to 3 KW for 10 minutes, day or night.

3.7.1.2.5.3 Batteries

The power module batteries shall be capable of supporting the Spacecraft subsystems and payload power requirements during normal operations through the full duration of each mission when solar array power is unavailable or insufficient.

3.7.1.2.5.4 Battery Charging

Battery charge control shall be such to maintain a safe state-of-charge condition in the batteries at all times during the mission. The control shall be fully automatic with command control override on all functions necessary to alter the automatic mode into a safe operating mode under possible abnormal conditions.

The primary battery charge control technique shall limit battery charge voltages to a level that is consistant with battery temperature and charge rates. The charge voltage limit shall be capable of being adjusted in discrete steps during flight to account for battery, EPS, Spacecraft and mission variations and tolerances.

3.7.1.2.6 Physical Requirements

All EPS functions, with the exception of the solar array and associated mechanisms, shall be physically located in the Power Module. The Power Module will be a standardized structure of approximately 48 inches by 48 inches by 18 inches deep, as described in Paragraph 3.7.1.4. The total weight of the Power Module shall be mission peculiar, depending upon redundancy, energy storage capacity and other specific mission requirements. Basic module weight data is specified in Paragraph 3.2.2.1.1. Detailed physical requirements shall be as stated in Specification EOS-SS-210.

3.7.1.2.7 Interface

3.7.1.2.7.1 Mechanical Interfaces

The mechanical interfaces between the Power Module and the Spacecraft structure shall be in accordance with Paragraph 3.7.1.4. The EPS assemblies and Power Module mechanical interfaces shall be in accordance with details contained in Specification EOS-SS-210.

3.7.1.2.7.2 Thermal Interfaces

The thermal design shall maintain all of the power subsystem equipment within the limits specified for all mission phases. The general heat sink operating temperature range shall be $21^{o}C \pm 11^{o}C$. Specific components requiring deviations from this value and temperatures for all modes shall be as specified in Paragraph 3.7.1.5.

The power subsystem module shall be designed to reject all equipment heat dissipation to space. The thermal design shall minimize the module heat sink gradients. The power subsystem module shall be made thermally independent from other subsystem modules and from the mounting structure, by using insulation and low conductance mounts.

3.7.1.2.7.3 Electrical Interfaces

All Power Module electrical interfaces with the spacecraft shall be via the Spacecraft Interface Connector per Paragraph 3.7.1.2.3.4.1. Umbilical connection interfaces with prelaunch and resupply equipment shall also be via the Spacecraft Interface Connector. Electrical interfaces, for prelaunch test operations only, shall be made via the Test Connectors per Paragraph 3.7.1.2.3.4.2. All electrical interfaces between assemblies and interface connectors within the Power Module shall be made with the Power Module Harness per Para. 3.7.1.2.3.5.

Detailed descriptions of all interfaces are defined in Specification EOS-SS-210.

3.7.1.2.7.4 Command and Data Handling Interface

A remote multiplexer and decoder unit shall be dedicated to the module to provide a standard interface between module data and command signals and the multiplex data bus system which is controlled from within the C&DH module. The remote unit shall be capable of providing the signal input and output interface as defined in Paragraph 3.7.1.1.5.2.3, Section I, entitled Remote Unit Characteristics.

3.7.1.2.8 Instrumentation Requirements

Instrumentation shall be provided that allows monitoring of the EPS during flight, test and resupply operations. All instrumentation circuitry shall be buffered from the circuitry being monitored to prevent loss of critical functions due to failures in the instrumentation equipment or shorts in the wiring. A detailed list of instrumented functions and parameters is contained in Specification EOS-SS-210.

3.7.1.2.8.1 Telemetry

Telemetry data shall be primarily limited to those functions necessary for control and operation of the power subsystem 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. For instance, an indication of equipment status should not be based on monitoring that a command to achieve that status has been issued.

3.7.1.2.8.2 Test

Test connectors shall be provided as applicable on major assemblies and at one side of the power subsystem module. The test connectors shall provide the capability, to the maximum extent possible, to determine degradation or the flight worthiness of the assemblies and module without the need for demating connectors in flight circuits. A separate group of connectors shall be provided for test, reconditioning and storage of the batteries. All outputs to test connectors except those to individual cells of the batteries shall be provided for test, reconditioning and storage of the batteries. All outputs to test connectors except those to individual cells of the batteries shall contain isolation circuitry. Isolation impedances must be high enough to permit proper circuit operation if the test point is accidentally shorted to ground. Test points on individual assemblies shall be brought out to connectors used for test only. Where feasible, all test points shall then be routed through a test circuit harness to the required test connector area on the side of the module. Low level circuits or those susceptible to noise or capacitive loading of the test cable may be terminated at the test connector on the assembly.

All analog and bilevel data outputs to telemetry shall be available at the test connectors to permit full time (non-sampled) monitoring of data during specific phases of the test program.

3.7.1.2.8.3 Resupply

For those missions that include resupply or retrieval requirements, hardline instrumentation provisions, via the Power Module/Spacecraft Interface Connector, shall be provided for all functions and parameters that are indicative of safety conditions within the subsystem.

3.7.1.2.9 Ground Support Equipment

The Contractor shall provide all ground support equipment (GSE) necessary for test and operation of the power subsystem module during integration and laboratory and environmental tests. This shall include bench test equipment (BTE) necessary for bench and environmental tests of power subsystem assemblies prior to integration into the module and all peripheral GSE necessary for controlling, powering, monitoring, and loading the integrated power subsystem during laboratory and environmental tests.

The BTE shall be sufficiently flexible to provide for maintenance of the module after its removal from the EOS in a failed mode. In addition, flexibility shall extend to permit the use of the BTE within the NASA LPS (Launch Processing System) during Shuttle operations. The peripheral GSE shall include but not limited to the following items:

(a) C&DH subsystem module simulator (command generator, a simple onboard computer simulator and control and formatting of multiplexer data).

(b) Telemetry and test connector output display, monitoring and recording.

(c) Battery charge and conditioning equipment.

(d) Solar array simulator and ground power source.

(e) Load banks to simulate each subsystem and the payload.

Design of items c, d, and e above, shall be suitable for use by the system integration contractor during integration and prelaunch operations.

The GSE shall be designed for fail safe operation. That is, ground support equipment failures or interruption of the main power sources shall not induce failures in the power subsystem module. Batteries shall be provided in the GSE as necessary to permit powering down of the subsystem during extended periods of loss of commercial power.

Additional GSE is specified in Paragraphs 3.2.1.2 and 3.2.2.2.

3.7.1.3 ATTITUDE CONTROL SUBSYSTEM MODULE

This specification establishes the performance and design requirements for the Attitude Control Subsystem (ACS) which forms a part of the EOS Spacecraft.

The ACS serves to stabilize and orient the Spacecraft in 3 axes after it separates from the launch vehicle. The ACS uses radiant energy from the sun and rate information for initial attitude control. The prime control will be performed by 3-axis rate-integrating gyros, a general-purpose digital computer for its prime controller element, three reaction wheels, three torquer bars, and attitude control jets (Orbit Adjust/Reaction Control System) as its control actuators. The ACS also generates and conditions status and monitoring signals for telemetering to the ground.

3.7.1.3.1 General Requirements

The ACS will provide the capability to point the Spacecraft toward the earth within specified tolerances and other associated capabilities for a Spacecraft with the following physical characteristics.

3.7.1.3.1.1 Spacecraft Mass Properties

Spacecraft mass properties are given in Table 3-10. The allowable tolerances on C.M. location shall be ±0.1 ft in the X direction and ±0.03 ft in the Y and Z directions. The tolerances on moments of inertia about the X, Y, and Z axes shall be ±25% of nominal. The cross-product moments of inertia shall not exceed ±3% of the maximum diagonal moment of inertia.

3.7.1.3.1.2 Disturbance Due to Instruments

The disturbance torques introduced by any of the GFE experiments shall not exceed 5×10^{-3} ft-lb.

3.7.1.3.1.3 Flexibility Parameters

The Spacecraft dynamic characteristics, i.e., natural frequencies and mode shapes, generalized masses and modal damping values shall be the following:

(a) Modal Frequencies (Fundamental)
Solar Panel 1.0 Hz - roll, pitch, and yaw

(b) Damping ratio, viscous:
Solar Panel - 0.001 Nominal, 0.0001 to 0.01 range

3.7.1.3.2 Functions

The ACS shall perform the following functions:

(a) Provide an accurately-pointed stable earth-referenced platform with low jitter amplitude to allow proper instrument operation.

(b) Provide inertial attitude hold for:

(1) Maximum power (solar array normal to the sun line)

(2) Retrieval operation

Table 3-10 Spacecraft Mass Properties

Stowed Configuration: Nominal CM Station (in): X = 100.72 Y = 0.68 Z = 2.62	Nominal Inertia Matrix (slug-ft^2): I_{xx} = 368 I_{xy} = 3.5 I_{xz} = 63 I_{yx} = -3.5 I_{yy} = 1365 I_{yz} = -1.1 I_{zx} = 63.1 I_{zy} = -1.1 I_{zz} = 1409
Deployed Configuration: Nominal CM Station (in): X = 97.79 Y = 8.06 Z = 2.62	Nominal Inertia Matrix (slug-ft^2): I_{xx} = 1485 I_{xy} = 23.6 I_{xz} = 67.3 I_{yz} = 23.6 I_{yy} = 1244 I_{yz} = 11.3 I_{zx} = 67.3 I_{zy} = 11.3 I_{zz} = 2432
Jet Lever Arms: Coupled Configuration: Roll 10.5 in Pitch, Yaw 7 in	

3.7.1.3.3 Configuration

The ACS shall consist of the equipment arranged as shown in Fig. 3-17 and as listed below:

(a) Coarse Sun Sensor

(b) Coarse Sun Sensor Electronics

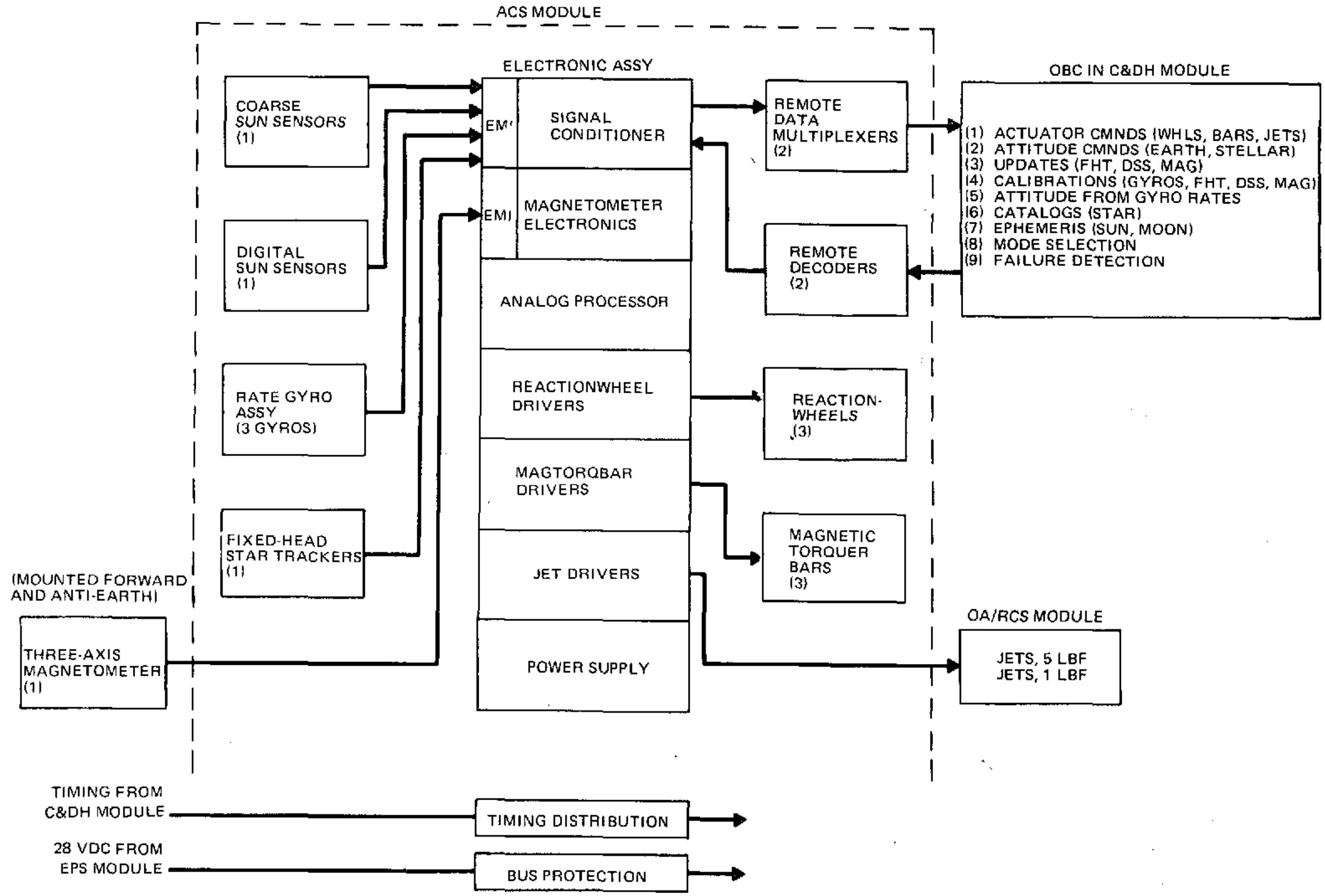

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Fig. 3-17 Block Diagram of Attitude Control System

(c) Digital Sun Sensor

(d) Digital Sun Sensor Electronics

(e) Fixed Head Tracker consisting of:

(1) Star Aspect Sensor

(2) Low Voltage Power Supply

(3) Bright Object Sensor

(4) Earth Albedo/Sun Shade

(f) Rate Gyro Assy (3 gyros)

(g) Magnetometer (3 probes)

(h) Electronics Assy consisting of:

(1) Magnetometer Electronics

(2) Signal Conditioning (MUX, Decoders)

(3) Analog Processor

(4) Reaction Wheel Drivers

(5) Magtorquer Drivers

(6) Jet Drivers

(7) Power Supply

(i) Reaction Wheels (3)

(j) Torquer Bars (3)

The following additional equipment shall be incorporated into the ACS module:

(a) Bus Protection Assembly

(b) Test Connector

(c) Remote MUX/Decoder

(d) Wiring Harness

(e) Timing Distribution

3.7.1.3.4 Modes

3.7.1.3.4.1 Launch Mode

During the launch mode the ACS shall be in a de-energized condition except for the following:

(a) The Fixed Head Tracker shall have the sun shutter circuitry energized so that the sun shutter will be closed if the sun comes into the field of view of the sensor.

(b) The ACS shall energize portions of the OA/RCS by external commands. The conditions shall be maintained until an external command is received to terminate them.

3.7.1.3.4.2 Control Modes

The ACS shall provide the necessary control modes to acquire the earth's center and to execute attitude control commands for purposes of earth pointing and orbit control maneuvers. Earth pointing is defined as the +Z (yaw) axis of the spacecraft directed toward the center of the earth, the +X (roll) axis in the orbit plane and in the direction of motion; and the +Y (pitch) axis perpendicular to the orbit plane. Earth pointing and roll, pitch, and yaw angle errors relative to it are depicted in Fig. 3-18. A summary of the modes is given in Table 3-11.

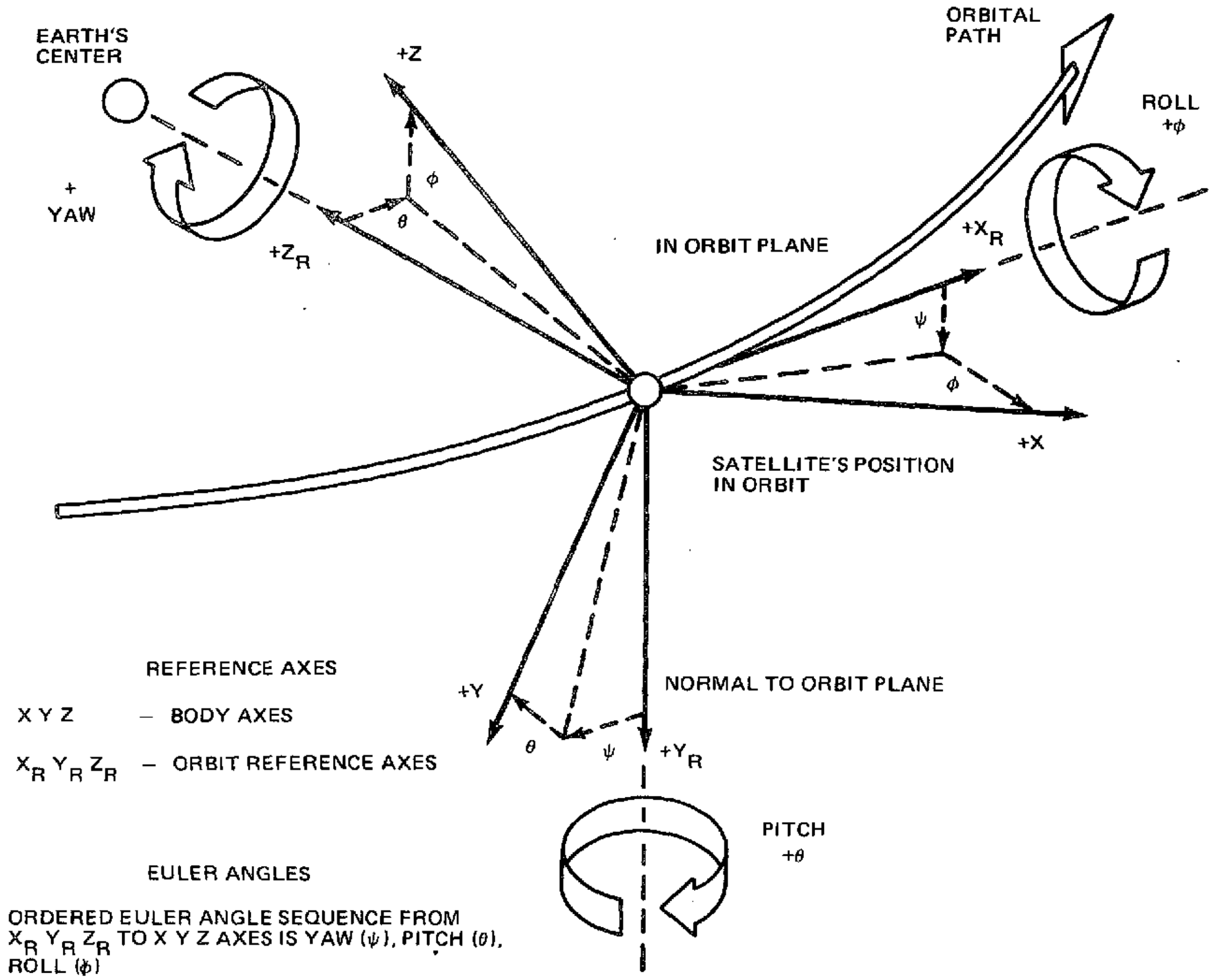
3.7.1.3.5 Performance Requirements

3.7.1.3.5.1 Rate Damping

After separation, the ACS shall reduce the angular rates about each of the three body axes to less than $\pm 0.03^{\circ}$/sec. The ACS shall be in the rate change mode. Rate damping shall normally be accomplished using the three-axis RGA, the OBC, and the attitude control jets of the OA/RCS. Rate damping shall be complete within 10 minutes maximum (5 minutes nominal) starting from rates of 1.0°/sec about each of the three body axes, with Spacecraft inertias as specified in Table 3.7.1.3-1. RGA gyro null offsets shall be compensated on-board using values computed in the OBC and supplied by the ground. A block diagram of this mode is shown in Fig. 3-19.

3.7.1.3.5.2 Coarse Sun Acquisition

Following rate damping the ACS shall acquire the Sun by causing the -Z axis of the Spacecraft to point to the sun to $< 2^{\circ}$. Coarse sun acquisition will be accomplished using the CSS, RGA, the OBC and the RCS jets. Sun acquisition to within $\pm 32^{\circ}$ (FOV of the DSS) of the spacecraft -Z axis shall be completed in less than 20 minutes (10 minutes nominal) from any initial orientation following rate damping to $\pm 0.03^{\circ}$/sec. A block diagram of this mode is shown in Fig. 3-20.



(1) 5-3 **Fig. 3-18 Definition of Earth-Pointing Orbit Reference Axis and Yaw, Pitch, and Roll Angles**

Table 3-11 Summary of ACS Modes

NO.	MODE	PURPOSE
1	RATE CHANGE	NULL RATES AFTER BOOSTER SEPARATION. GENERATE ORBIT RATE ABOUT THE PITCH AXIS IN PREPARATION FOR THE EARTH-POINTING ATTITUDE HOLD MODE.
2	COARSE SUN ACQUISITION	ACQUIRE THE SUN FOR SOLAR POWER AND IN PREPARATION FOR FINE SUN ACQUISITION AND FOR SUBSEQUENT GUIDE STAR ACQUISITION.
3	FINE SUN ACQUISITION	POINT TOWARD THE SUN WITH INCREASED ACCURACY. UPDATE ATTITUDE IN PREPARATION FOR SUBSEQUENT GUIDE STAR ACQUISITION.
4	RATE HOLD	HOLD SELECTED RATE ABOUT SUNLINE FOR GUIDE STAR ACQUISITION (ALTERNATIVE: SLEW ABOUT SUNLINE TO ATTITUDE FOR GUIDE STAR ACQUISITION AFTER UPDATING USING DSS AND MAGNETOMETER). BACKUP FOR EARTH-POINTING. HOLD ORBIT RATE ABOUT PITCH AXIS PRIOR TO EARTH-POINTING ATTITUDE HOLD. BACKUP FOR DEPLOYMENT, RETRIEVAL, AND SERVICE OPERATIONS.
5	SLEW	CHANGE ATTITUDE FROM PRESENT ATTITUDE TO ANOTHER IN PREPARATION FOR NEXT EVENT, SUCH AS EARTH-POINTING.
6	EARTH-POINTING ACQUISITION HOLD	POINT THE INSTRUMENTS AT THE EARTH AND X AXIS IN THE DIRECTION OF FLIGHT TO PERFORM THE EOS MISSION.
7	INERTIAL-POINTING ATTITUDE HOLD	POINT THE INSTRUMENTS TOWARD A SELECTED POINT IN SPACE WITH THE ROLL ANGLE ABOUT THIS LINE IN SPACE CHOSEN FOR MAXIMUM SOLAR POWER. PERFORM A STELLAR MISSION. HOLD AN ATTITUDE SUITABLE FOR DEPLOYMENT, RETRIEVAL, OR SERVICING.
8	SURVIVAL	SURVIVE IN CASE OF FAILURES IN OTHER MODES. MAXIMUM SOLAR POWER IS OBTAINED. RETRIEVAL OR SERVICING CAN BE ACCOMPLISHED. SOLUTIONS TO FAILURES CAN BE WORKED OUT.

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3.7.1.3.5.3 Fine Sun Acquisition

When the -Z axis of the Spacecraft comes within the FOV of the DSS ($\pm 32^{\circ}$) and the body rates are less than $\pm 0.03^{\circ}$/sec, control of the Spacecraft will be transferred to the DSS. Fine Solar Acquisition will be accomplished using the DSS, the RGA, the OBC, the three reaction wheels, and magnetic unloading of the reaction wheels. (If required, unloading will be performed using the RCS jets.) Fine Sun Acquisition to within 5° of the -Z axis shall be completed in less than 10 minutes maximum (5 minutes nominal) starting with the -Z axis 20 degrees from the sun line. The Spacecraft -Z axis will be held to 0.1° and a final angular rate of 0.03°/sec. A block diagram of this mode is shown in Fig. 3-21.

3.7.1.3.5.4 Rate Hold

In the inertial attitude hold mode, the ACS will hold the attitude of the Spacecraft at any inertially referenced attitude. Before in-orbit calibration of the gyros, the drift tolerance will be $\pm 0.03^{\circ}$/hr, but after in-orbit calibration, the ACS shall maintain rate drift to 0.003°/hr.

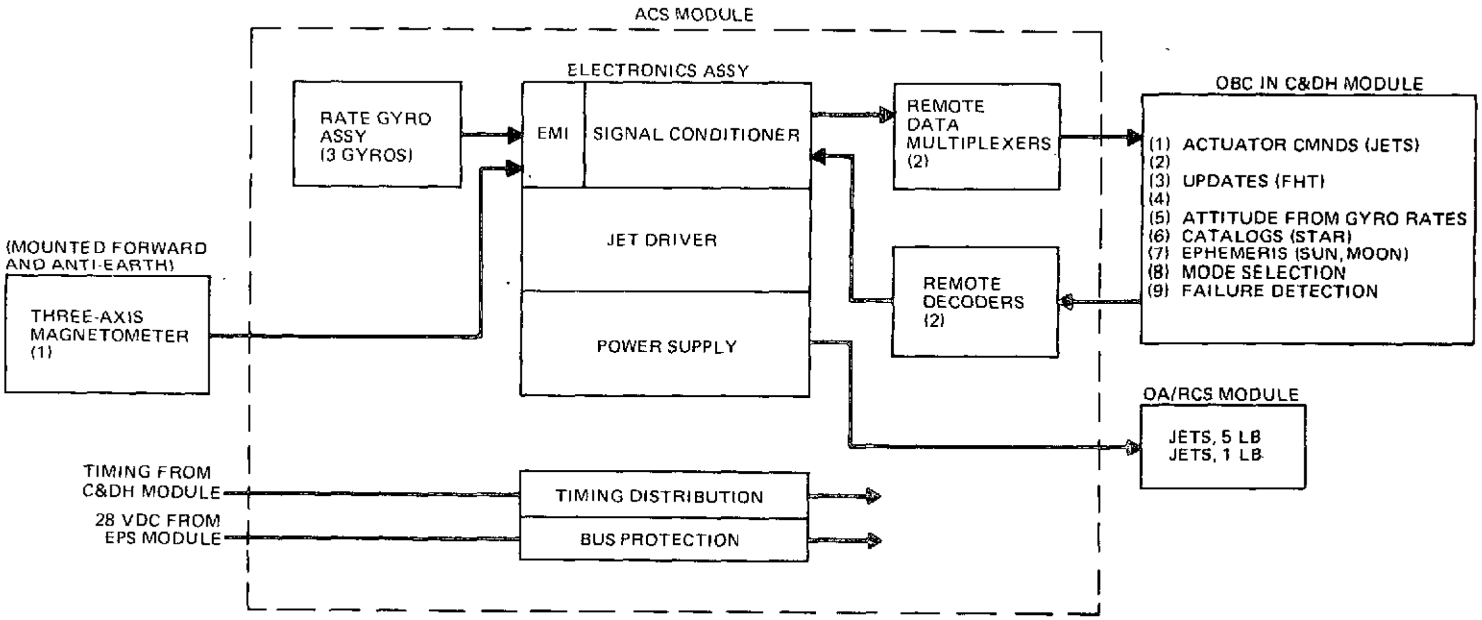

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Fig. 3-19 Equipment Used in Rate Change Mode

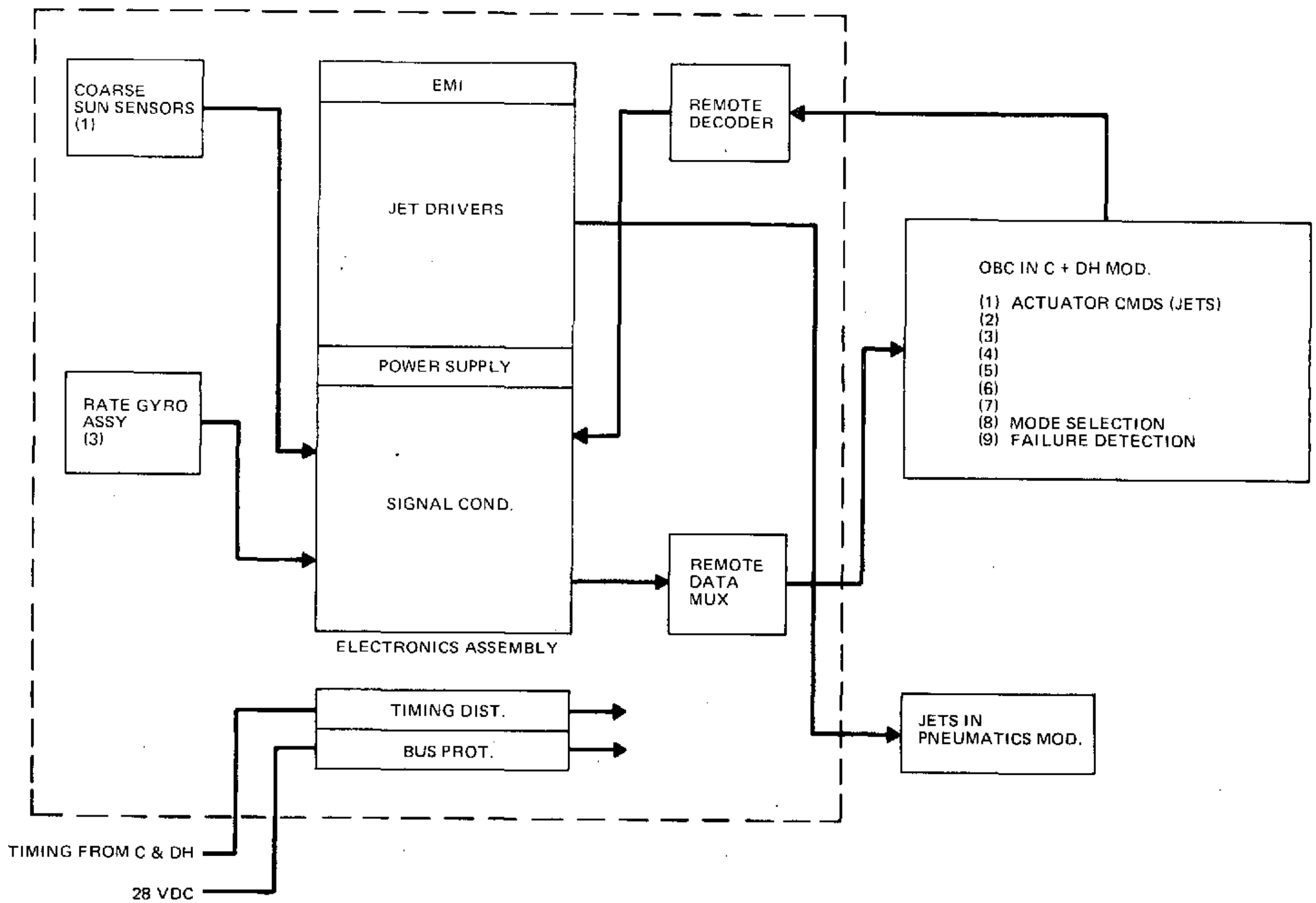

Fig. 3-20 Equipment Used in Coarse Sun Acquisition Mode

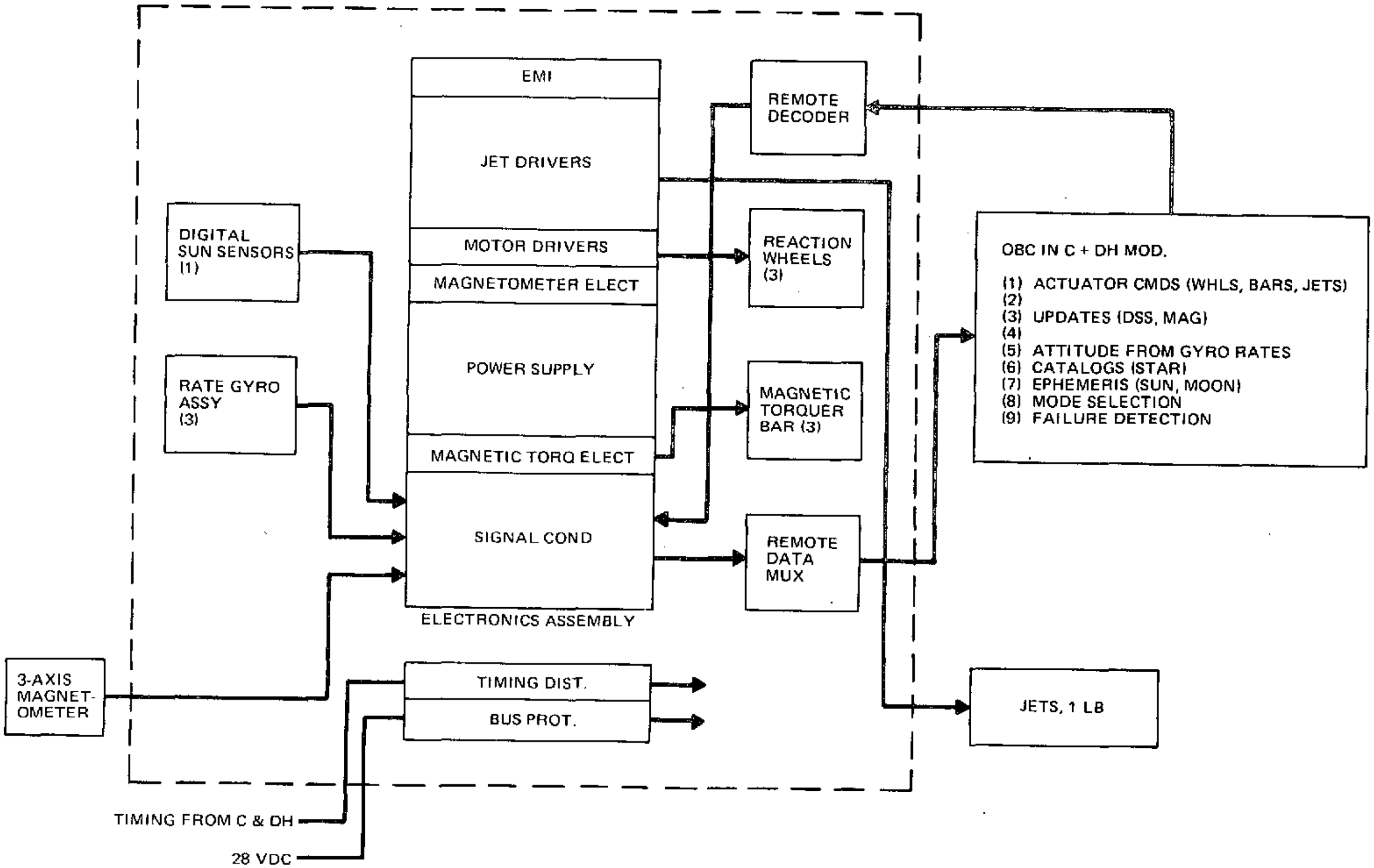

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Fig. 3-21 Equipment Used in Fine Sun Acquisition Mode

Inertial attitude hold will be accomplished using the 3-axis RGA, the OBC, the three reaction wheels with magnetic unloading, and, if required, jet unloading. Gyro systematic and random drift rate will cause the Spacecraft to drift. A block diagram of this mode is shown in Fig. 3-22.

3.7.1.3.5.5 Earth Acquisition

To achieve earth acquisition the spacecraft will be sequentially slewed using the RGA, Magnetometer, OBC, reaction wheels and magtorquers (jets, also, if required, for unloading and the FHT continues to update) to an attitude which, at a point in the orbit, results in an earth pointing attitude. Just before arriving at this specific point in the orbit, the pitch rate is commenced. On arriving at this point in the orbit, the pitch rate shall equal orbit rate, the instruments will point at the earth, and the Spacecraft X-axis will point in the direction of flight.

Earth Acquisition will utilize the RGA, FHT, Magnetometer, OBC, reaction wheels, magtorquers, and the RCS jets. The modes used are the slew, rate hold, and rate change modes. The block diagrams of the slew and rate change modes are shown in Fig. 3-19 and 3-23.

3.7.1.3.5.6 Earth-Pointing Attitude Hold

Once earth acquisition has been accomplished, earth pointing will be maintained by holding the pitch rate equal to the orbit rate. Roll, pitch and yaw errors will be slewed by the inertial sensors. The RGA assembly will be updated by the FHT, and corrections will be made to the rate commands based on stellar updates. Orbit regression will be compensated by including slight oscillatory rate commands in the roll and yaw axes. Attitude commands are computed by first calculating the satellite ephemeris.

The ACS shall provide the capability to point the yaw +Z axis of the Spacecraft toward the earth's centroid with the following performance levels:

- Pointing accuracy (per axis) $\pm 0.01^{\circ}$
- Rate stability over 30 minutes $\pm 10^{-6}$ $^{\circ}$/sec
- Jitter up to 30 seconds $\pm$ 1 sec
- Jitter up to 20 minutes $\pm$ 2 sec

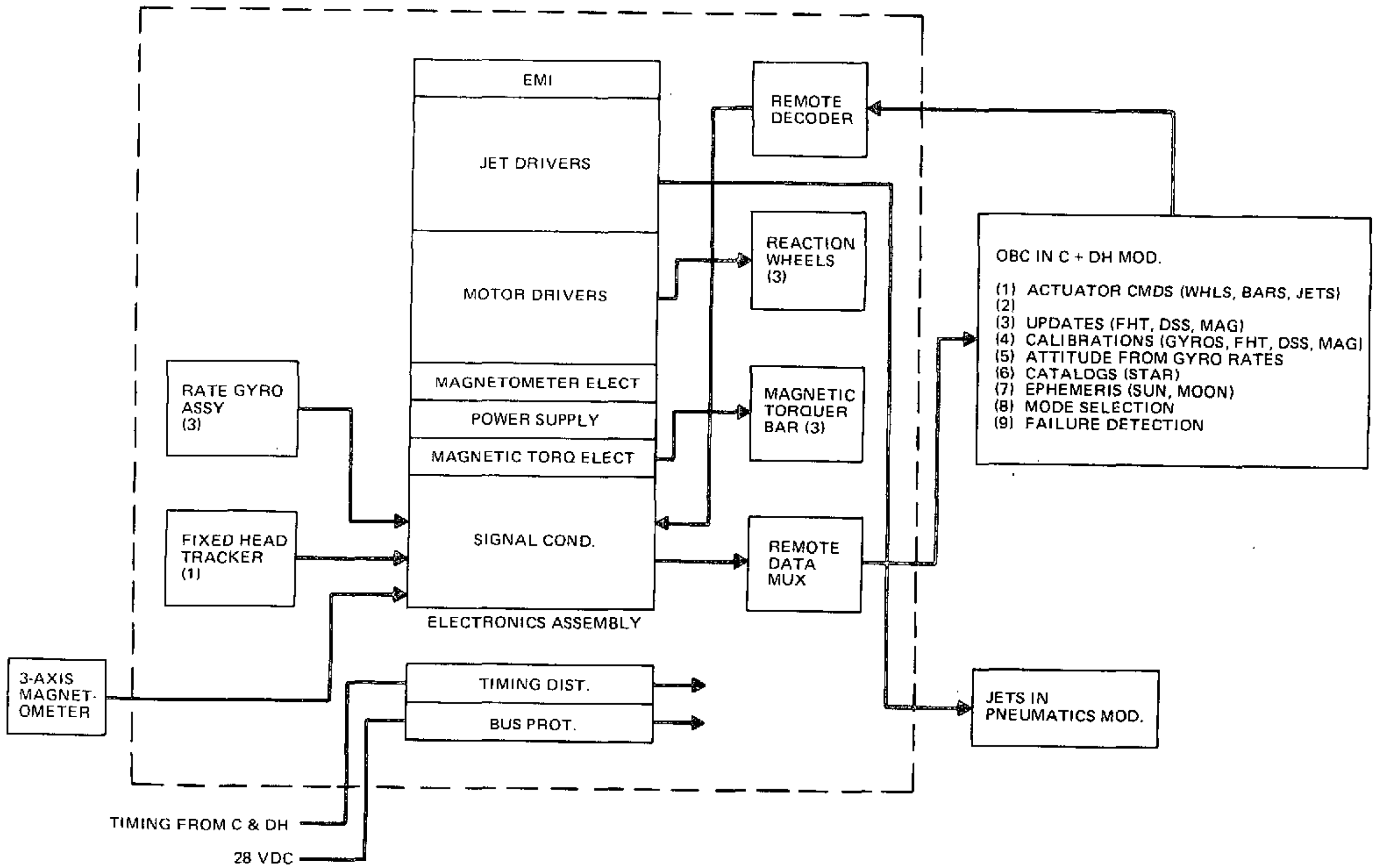

Fig. 3-22 Equipment Used in Rate Hold Mode

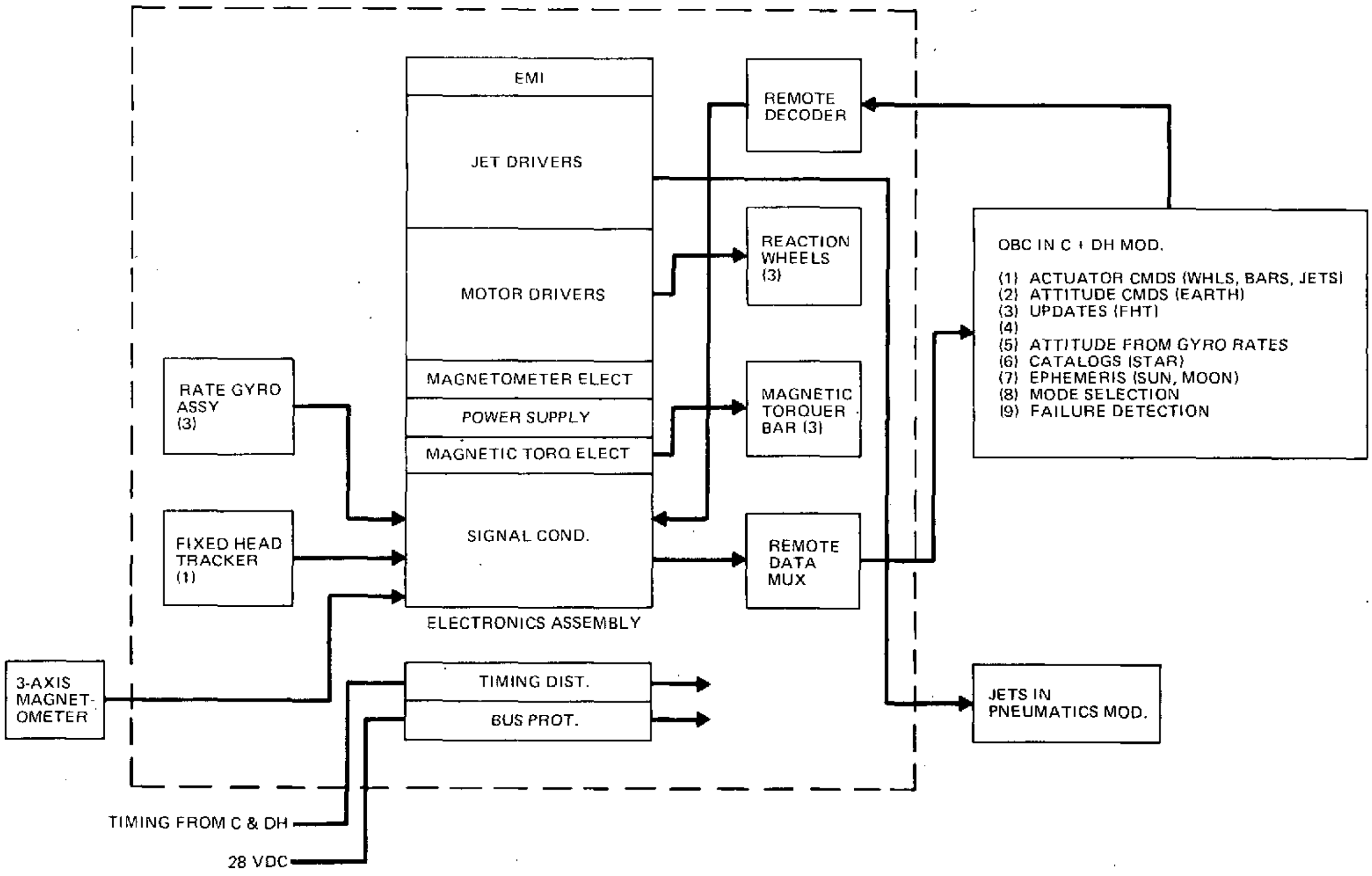

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Fig. 3-23 Equipment Used in Slew Mode

Earth pointing will be accomplished using the 3-axis rate gyro assy, the OBC, the three reaction wheels, magnetic unloading, and, if required, RCS unloading. A block diagram of this mode is shown in Fig. 3-24.

3.7.1.3.5.7 Inertial - Pointing Attitude Hold

The ACS will have the capability for holding any inertial attitude and continue holding that attitude for a time interval of one hour. The pointing requirements including in-orbit calibration are as follows:

Pointing Accuracy 0.01°

Pointing stability over 30 minutes $\pm 10^{-6}$ °/sec, jitter 2 sec

The ACS will also have the capability of taking signals from the stellar instrument package and using them for control purposes. The pointing requirements in this case are as follows:

Pointing Accuracy 0.01 sec

Jitter ± 0.003 sec

To perform inertial (stellar) pointing the ACS will use the 3-axis rate gyro assy, instrument optics, OBC, 3 reaction wheels, magnetic unloading, and, if required, RCS unloading. A block diagram of this mode is shown in Fig. 3-25.

3.7.1.3.5.8 Survival Mode

The ACS will be capable of holding the solar array normal to the sunline during periods of failures or operational difficulties. Performance requirements are $\pm 7^{\circ}$ total and Spacecraft rates less than 0.05°/sec. This mode shall be maintained for an unlimited period of time.

The equipment required for the survival mode are the CSS, magnetometers, analog processor (part of the electronics assembly), three reaction wheels, magnetic unloading, and if required, RCS unloading.

A block diagram of this mode is shown in Fig. 3-26.

3.7.1.3.6 Physical Requirements

The ACS shall be housed in a standard module as specified in Paragraph 3.7.4.4 excluding components external to the standard module (e.g., magnetometers). The weight allocation for the ACS module is specified in Paragraph 3.2.2.1.1.

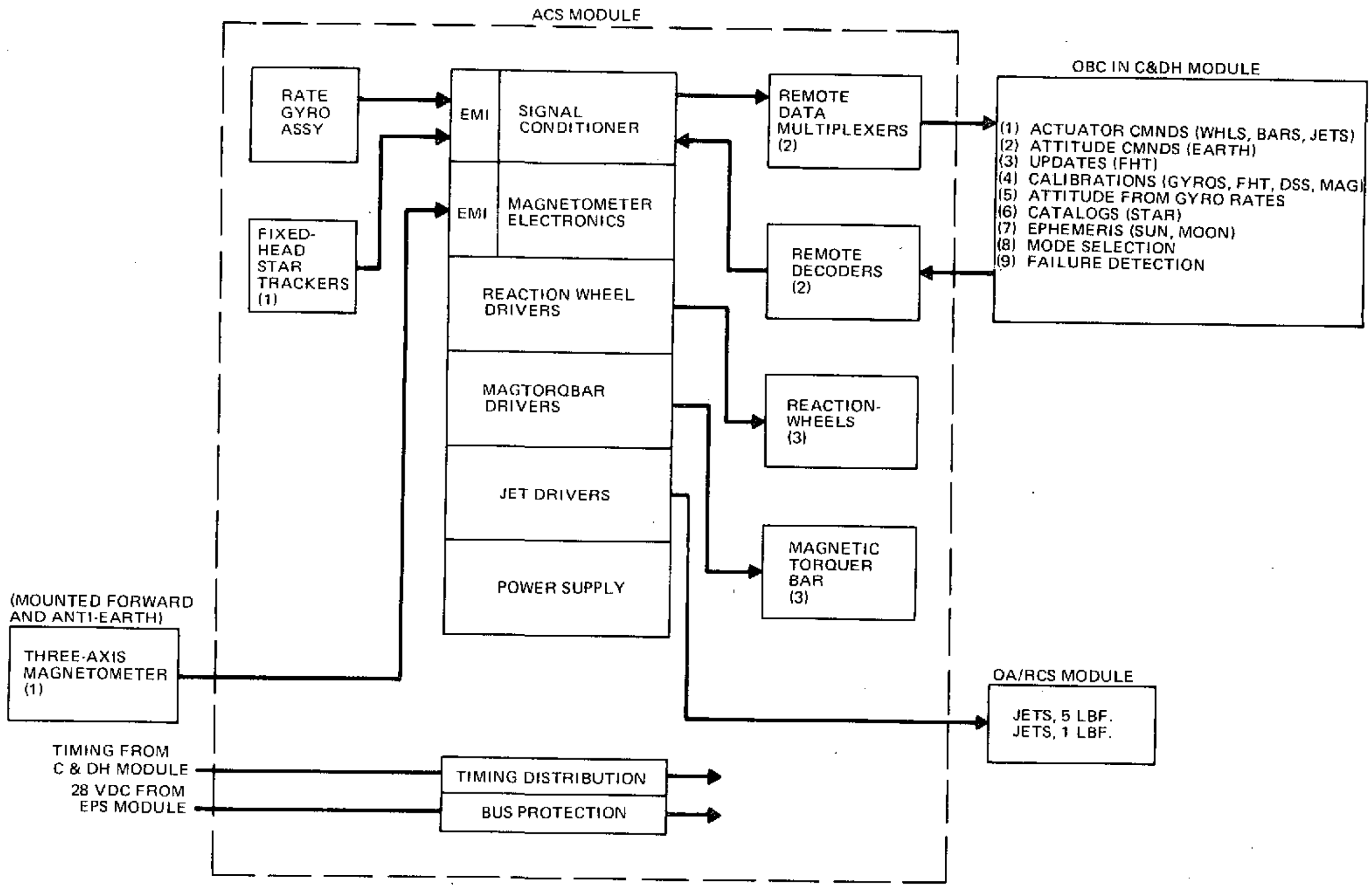

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Fig. 3-24 Equipment Used in Earth-Pointing Attitude Hold Mode

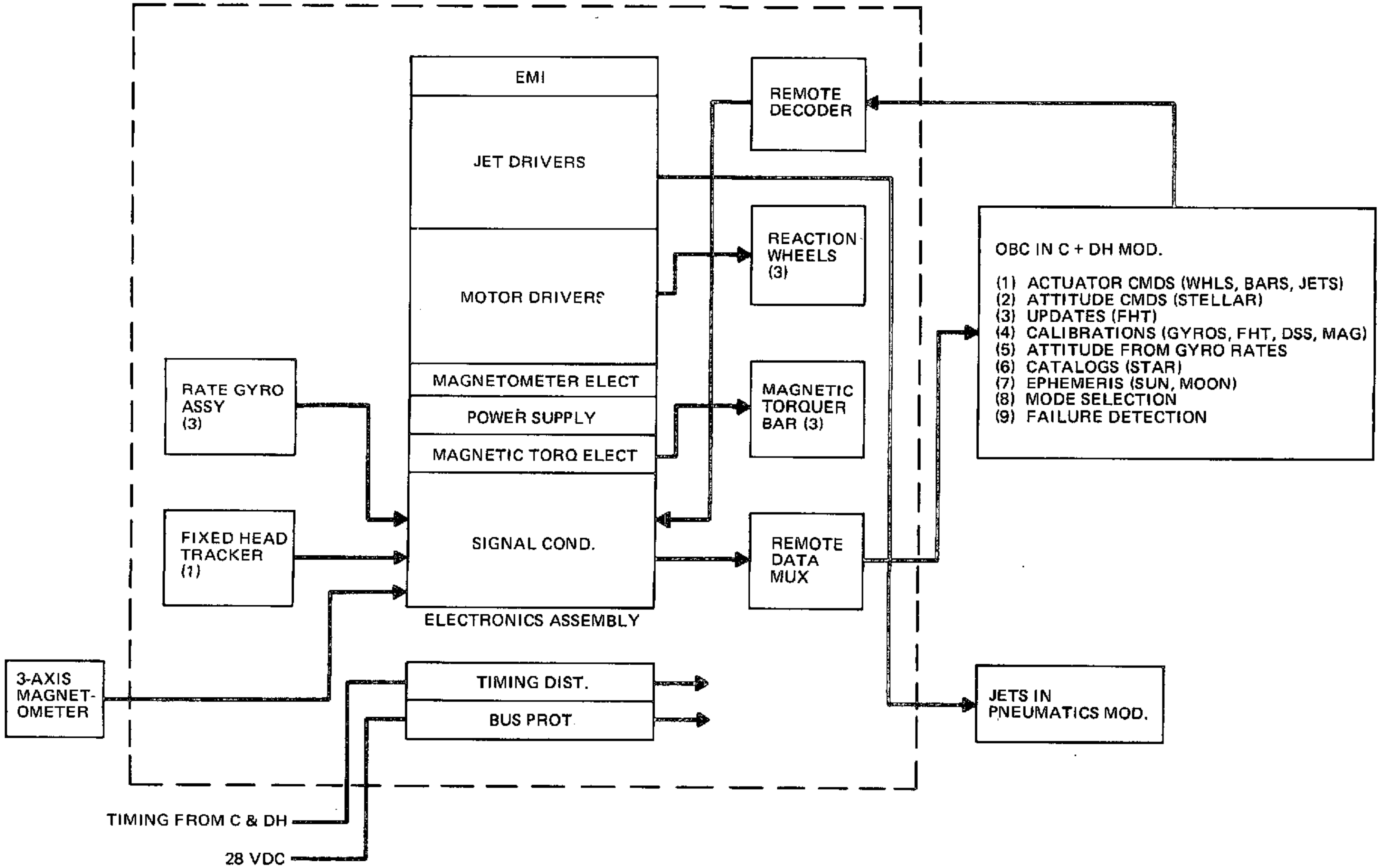

Fig. 3-25 Equipment Used in Inertial – Pointing Attitude Hold Mode

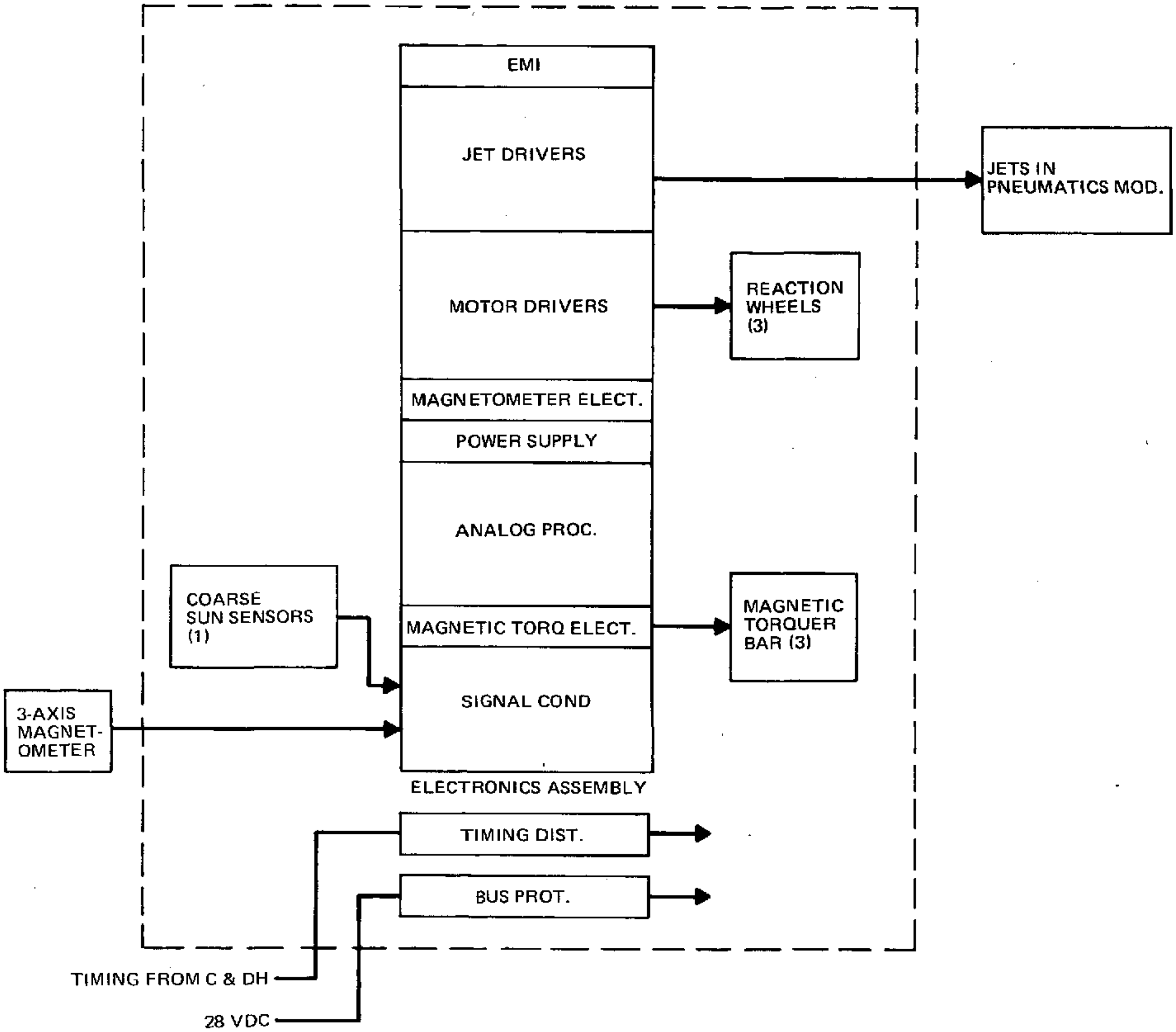

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Fig. 3-26 Equipment Used in Survival Mode

3.7.1.3.7 Interface Requirements

3.7.1.3.7.1 Thermal Interfaces

The thermal design shall maintain all of the ACS subsystem equipment within the limits specified for all mission phases. The general heat sink operating temperature range shall be $21^{\circ}C \pm 11^{\circ}C$. Specific components requiring deviations from this value and temperatures for all modes shall be as specified in Paragraph 3.7.1.5.

The ACS subsystem module shall be designed to reject all equipment heat dissipation to space. The thermal design shall minimize the module heat sink gradients. The ACS subsystem module shall be made thermally independent from other subsystem modules and from the mounting structure, by using insulation and low conductance mounts.

3.7.1.3.7.2 Mechanical Interfaces

The mechanical interfaces between the ACS module and the Spacecraft structure shall be in accordance with Paragraph 3.7.1.4.

3.7.1.3.7.3 Power

The ACS shall be designed to operate from the power module whose primary power is 28 ± 7 VDC. The ACS equipment will provide their own power conditioning.

3.7.1.3.7.4 Command and Data Handling

Command and Data Handling design requirements shall be as specified in Paragraph 3.7.1.1.

3.7.1.3.7.5 Bus Protection Assembly

The distribution circuitry for subsystem module loads shall contain devices to protect the power busses from short circuits. The bus protection circuitry shall be provided for all loads except those which are non-redundant and/or critical to mission success. The criteria for device selection, sizing, and derating shall be: steady state current; transient current/voltage; thermal environment; bus short-circuit capabilities, and redundancy requirements. Prime and redundant protection circuitry shall be packaged in the module in such a manner as to facilitate ease of prelaunch checkout verification.

3.7.1.3.7.6 Spacecraft Interface Connector

The Spacecraft interface connector shall provide the electrical connect/disconnect between module and Spacecraft. The connector(s) shall be designed for and physically be positioned to assure interchangeability of modules. Specific design requirements shall be: blind mate capability; anti-bind roll-off shell (angular disconnect capability); maximum axial movement of structure without affecting continuity, and highly reliable contacts with self aligning capabilities.

3.7.1.3.7.7 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 subsystem electrical assemblies. It shall be possible to remove electrical assemblies without removing the harness. Harness and cable assembly practices shall meet the intent of MIL-W-5088 unless specified otherwise in this specification. Wire types shall be lightweight, abrasion resistant, and space qualified. Coaxial cable shall conform to MIL-C-17. Wire size shall be determined by: circuit steady state current; voltage drop compatible with unit performance requirements; thermal environment; connector termination capabilities; minimum wire gauge 24 awg high strength copper alloy; bundle capacity, and minimum weight. Connectors shall be of the removable crimp contact type where feasible and shall meet environmental requirements for space application.

3.7.1.3.8 Instrumentation Requirements

3.7.1.3.8.1 Telemetry

The ACS shall be instrumented with telemetry circuitry which can indicate and locate any failure on the system or on a component level.

The types of telemetry functions that are required for the ACS are:

- Mode determination
- Logic functions
- Error signals from sensors
- Drive signals
- Power status (on/off)
- Pressure levels (hermetically sealed units)
- Thermal conditions

The electrical characteristics for the telemetry circuitry which interfaces with the ACS shall be as specified in Paragraph 3.7.1.1.

3.7.1.3.8.2 Test Points

Test connectors shall be provided as applicable on major assemblies and at one side of the subsystem module. The test connectors shall provide the capability, to the maximum extent possible, to determine degradation or the flight worthiness of the assemblies and module without the need for demating connectors in flight circuits. All outputs to test connectors shall contain isolation circuitry.

3.7.1.3.9 Ground Support Equipment

The ACS will require a S/S Module Checkout Test Bench capable of powering the module, performing all tests required for the integration of its components and for its acceptance test. The bench shall be sufficiently flexible to provide for maintenance of the module after its removal from the Observatory in a failed mode. In addition, the flexibility shall extend to permit the use of the bench within the NASA LPS (Launch Processing System) during Shuttle operations.

The bench shall be designed such that bench failure shall not induce failures within the module. (Refer to Paragraph 3.1.3.1 for additional GSE.)

3.7.1.4 STRUCTURE SUBSYSTEM

The structure shall possess sufficient strength, rigidity, and other characteristics required to survive critical loading conditions that exist within the envelope of mission requirements. The structure shall survive those conditions in a manner that does not reduce the probability of mission success. The design shall: (a) be based upon the structural design principles and assumptions listed in NASA SP-8057, and (b) satisfy the requirements of EOS-SS-230 (TBD).

3.7.1.4.1 General Requirements

3.7.1.4.1.1 Design Approach

The structure shall possess sufficient strength, rigidity and other necessary characteristics to survive the critical loading conditions that exist within the envelope of mission requirements. It shall survive those conditions in a manner that insures the successful completion of the mission.

The methods used in structural and dynamic analysis shall be rational and conservative. The term "rational" denotes methods that are based on the accepted principles of mechanics. Conservative methods are those for which any necessary assumptions conform to accepted engineering practice and are chosen in such a way that critical loads and stresses are not underestimated.

3.7.1.4.1.2 Design Environments

The Spacecraft structure shall be designed to meet the environments specified in Paragraph 3.2.7. The environmental phenomena corresponding to each design condition shall include all factors that can influence the structural design, and typically include heating, vibration, shock and acoustics, in addition to quasi-steady and dynamic loads. Consideration shall be given to the deteriorating effect of prolonged exposure to the space environment. Where appropriate all such phenomena shall be determined statistically.

3.7.1.4.1.2.1 External and Internal Load Distribution

Analyses for the determination of external loads from the design environment shall employ conservative methods and assumptions. Determination of internal structural load distributions shall be rational analyses. These analyses shall include the effects of deformations, temperatures, and material and geometric nonlinearities on internal load distribution.

When internal pressure effects in a combined load condition are stabilizing or otherwise beneficial to structural load capability, the minimum anticipated internal operating pressure shall be used with a safety factor of 1.0 to arrive at an ultimate internal pressure in the ultimate loads analysis.

3.7.1.4.1.2.3 Misalignment and Dimensional Tolerances

The analysis of all loads, load distributions, and structural adequacy shall account for the effects of allowable structural misalignments, control misalignments, and other permissible and expected dimensional tolerances.

3.7.1.4.1.2.4 Dynamic Loads

Dynamic loads shall be considered for all phenomena expected in each design environment. The calculation of all dynamic loads shall include the effects of vehicle structural flexibilities and damping, and coupling of structural dynamics with the control system and the external environment.

3.7.1.4.1.2.5 Repeated Loads and Thermal Fatigue

The structural design shall account for the effects of repeated loads and elevated temperature. The design structural adequacy of the Spacecraft in flight shall not be impaired by fatigue damage resulting from exposure to non-flight and launch environments.

3.7.1.4.1.2.6 Vibrational and Acoustical Loadings

The effects of the vibrational and acoustical environments shall be accounted for in design wherever practicable by rational analysis of the response of the dynamic system to the environment.

3.7.1.4.1.2.7 Creep Deformation

The effects of permanent creep deformation shall be considered by rational methods of analysis. Where not otherwise critical, i.e., creep buckling, etc., a permanent deformation of 1 percent shall be considered as the maximum permissible value.

3.7.1.4.1.2.8 Thermal Stresses

The effects of thermal stresses where significant shall be combined with the appropriate load stresses when calculating required strength. Thermal stresses shall be combined with load stresses in a rational manner. For relieving thermal stresses when combined with load stresses, a safety factor of 1.0 shall be used. For additive thermal stresses, the safety factors of Paragraph 3.7.1.4.1.7 shall be used.

3.7.1.4.1.2.9 Malfunctions

The structure shall not be designed to withstand loads produced by any system malfunction that would otherwise result in failure to accomplish the mission.

3.7.1.4.1.3 Material Properties and Allowables

3.7.1.4.1.3.1 Sources

Material strengths and other mechanical and physical properties shall be selected from NASA approved sources of reference, such as MIL-HDBK-5, and MIL-HDBK-17, and from contractor test values when appropriate. Strength allowables and other mechanical properties used shall be appropriate to the loading conditions, design environments, and stress states for each structural member.

3.7.1.4.1.3.2 Single Load Path Structures

For single load path structures, the minimum guaranteed values, (A) values in MIL-HDBK-5, shall be used.

3.7.1.4.1.3.3 Multiple Load Path Structures

For multiple load path structures, the 90 percent probability values, (B) values in MIL-HDBK-5B, shall be used. These values are to be consistent with overall vehicle reliability requirements.

3.7.1.4.1.4 Strength Requirements

3.7.1.4.1.4.1 At Limit Load

The structure shall be designed to have sufficient strength to withstand simultaneously the limit loads, applied temperature and other accompanying environmental phenomena for each design condition without experiencing excessive elastic or plastic deformation. Transportation and handling loads shall be considered for design of the structure and kept lower than flight induced loads.

3.7.1.4.1.4.2 At Ultimate Load

The structure shall be designed to withstand simultaneously the ultimate loads, applied temperature and other accompanying environmental phenomena without failure. No factor of safety shall be applied to any environmental phenomena except loads.

3.7.1.4.1.4.3 Margin of Safety

Margin of safety is defined as: $MS = \frac{1}{R} - 1$ where R is the ratio of applied load (or stress, when applicable) to the allowable load (or stress).

Determination of the factor R shall include the effects of combined leads or stresses (interaction). It shall be a criterion for structural design that no margin of safety be less than zero.

3.7.1.4.1.5 Stiffness Requirements

3.7.1.4.1.5.1 Under Limit Loads

The structure shall not experience excessive deformations at limit loads and in the appropriate design environment. In particular, the stiffness of all portions of the structure shall be sufficiently great that deflection under limit loads does not produce inadvertent contact or interference between adjacent parts of the structure or between the structure and fairing or between the structure and the booster interface.

3.7.1.4.1.5.2 Under Ultimate Loads

Structural deformation shall not precipitate structural failure during any design conditions and environment at ultimate load or less.

3.7.1.4.1.5.3 Dynamic Properties

The structural dynamic properties of the EOS Spacecraft and payload shall be such that interactions with control system dynamics does not result in unacceptable degradation of control system performance. This requirement may impose a minimum natural frequency requirement on the structure.

3.7.1.4.1.5.4 Minimum Frequency

The stiffness of the Spacecraft structure, restrained at the Spacecraft/Launch Vehicle Interface, shall be designed to result in fundamental frequencies greater than 35 Hz in the longitudinal axis and 15 Hz in the lateral axes.

3.7.1.4.1.5.5 Component and Attachment Stiffness

The fundamental natural frequency of all components shall be 50 Hz, or greater, when mounted on the Spacecraft structure. Analyses and/or tests will be required to establish component flexibilities when installed on the Spacecraft. Components which cannot meet this criterion must be individually modeled in the Spacecraft dynamic loads analyses so as to account for response amplification effects.

3.7.1.4.1.6 Thermal Requirements

The design of the vehicle shall account for the effects of temperature. The temperature requirements and results of analysis specified in Paragraph 3.7.3.1.5 shall be used to determine thermal effects on the structure. The structure thermal effects which will be considered include thermal stresses and deformations, and mechanical and physical property changes.

3.7.1.4.1.7 Loads

3.7.1.4.1.7.1 Flight Loads

The structural factors of safety for all externally applied loads shall be as follows:

- Limit factor of safety = 1.0
- Ultimate factor of safety = 1.5

3.7.1.4.1.7.2 Non-Flight Loads

The above factors of safety shall apply to all non-flight loads.

3.7.1.4.1.7.3 Pressure Vessels

The limit pressure shall be equal to the maximum relief valve setting for pressure vessels. Where the pressure is a relieving load to externally applied loads the limit pressure is the minimum regulated pressure. The limit factor is equal to 1.0. All pressure vessels shall be subjected to proof pressure test during acceptance testing. The proof factor as applied to limit loads shall be established by fracture mechanics considerations as given in NASA SP-8040. These methods shall be used to establish the proof factors, ultimate safety factors and service life of the pressure vessel.

For preliminary design the yield factor of safety shall be 1.25 and ultimate factor equal to 2.0.

3.7.1.4.1.8 Dynamic Environmental Safety Factors

The Observatory shall be designed and certified to withstand the dynamic environments specified in Paragraph 3.2.7 increased by the appropriate safety factors noted below.

3.7.1.4.1.8.1 Acoustic Levels

The design and certification test factor shall be + 4 dB applied to the maximum expected internal acoustic environmental spectrum and the overall level. The exposure time duration shall be 2 minutes.

3.7.1.4.1.8.2 Sinusoidal Levels

The design and certification test factor shall be +3.5 dB applied to the maximum expected sinusoidal vibration environmental levels. The logarithmic frequency sweep rate shall be 2 octaves/minute along each of the three orthogonal axes.

3.7.1.4.1.8.3 Random Levels

The design and certification test factor shall be +3.5 dB applied to the maximum expected random vibration environmental spectrum and the overall root-mean-square level. The exposure time duration shall be 2 minutes along each of the three orthogonal axes.

3.7.1.4.1.9 Flight Vehicle Mission Phases

The Spacecraft shall be capable of withstanding all load conditions, and all environments to which it is exposed, in all phases of assembly, transportation, and flight specified herein. All items and components shall be designed for the most severe environmental conditions with consideration of both operational and nonoperational states.

3.7.1.4.1.9.1 Ground Phase

The structural design shall account for all environments to which the structure and its component parts are exposed during manufacturing, handling, transportation, erection and storage. Except for local support structures, the ground loads shall not govern design of the structure.

3.7.1.4.1.9.2 Prelaunch and Erection Phases

The structure shall be capable of sustaining all prelaunch and erection load conditions.

3.7.1.4.1.9.3 Launch Release

The structure shall be capable of sustaining all load conditions as may be experienced during launch operations.

3.7.1.4.1.9.4 Powered Flight

The structure shall be designed for the entire powered flight environment.

3.7.1.4.1.9.5 Orbit Phase

The Spacecraft shall be designed for all geophysical environments and loading conditions associated with launch vehicle separation and orbital flight. The design of the vehicle and its parts shall be based on, but not limited to, consideration of the following conditions.

3.7.1.4.1.9.5.1 Maneuvering Loads

The vehicle shall be designed for loads resulting from orbit adjust and reaction control equipment maneuvers for changing orbits, station keeping and attitude control.

3.7.1.4.1.9.5.2 Deployment of Appendages

The Spacecraft shall be designed to accommodate loads induced by the deployment of solar arrays, antennas and all other appendages.

3.7.1.4.1.9.5.3 Meteoroid

The structure shall be designed in allowance with the requirements of Paragraph 3.2.7.4.5.

3.7.1.4.1.9.5.4 Radiation Environment

The effect of both natural and artificial radiation environment shall be considered in designing the structure, radiators, solar panels, etc., including not only the deterioration and induced radiation effects on the materials, but also the shielding that may be required for sensitive equipment. The radiation environment is specified in Paragraphs 3.2.7.4.7, 3.2.7.4.8 and 3.2.7.4.9.

3.7.1.4.2 Functions

The structure subsystem shall provide:

(a) All primary and secondary structures required to provide adequate support and protection to all subsystem equipment items such that they will withstand all natural and induced environmental forces to which the Spacecraft shall be exposed during all ground and flight phases of the mission.

(b) Adequate rigidity in those areas where subsystem equipment items requiring critical alignment are mounted and whose geometry is critical to achieving mission objectives.

(c) For all required mechanical interfaces between the spacecraft and:

(1) The launch vehicles (Delta 2910 and Space Shuttle)

(2) Ground support equipment

(3) Launch pad handling equipment

(d) Sufficient space for adequate access to permit efficient preflight and on orbit servicing, maintenance, and replacement of subsystem modules and equipment items.

(e) For the prevention of structural deformations of a magnitude sufficient to:

(1) Cause structural failure

(2) Jeopardize the proper functioning of equipment items

(3) Endanger the functional characteristics of the S/C at any time during all ground and flight phases of the mission.

3.7.1.4.3 Configuration

The structure subsystem shall consist of the following elements as shown in Fig. 3-27.

(a) Spacecraft core structure

(b) Three subsystem modules (ACS, EPS & C&DH)

(c) Orbit adjust/RCS module

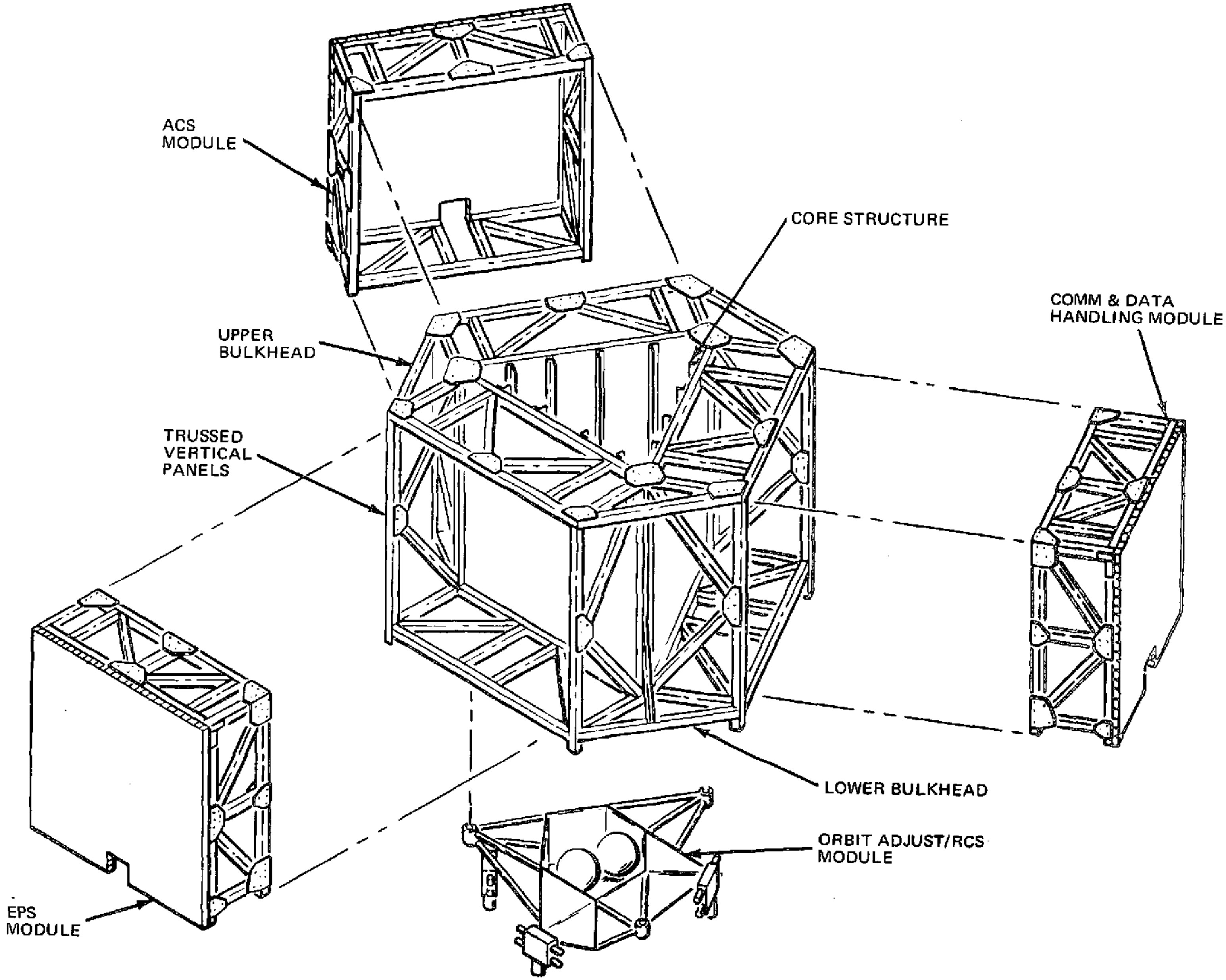

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Fig. 3-27 Basic Spacecraft Structure

3.7.1.4.4 Performance Requirements

3.7.1.4.4.1 Spacecraft Core Structure

The core structure illustrated in Fig. 3-28 contains:

(a) A hollow triangular structure between the adapter and the mission peculiar instrument support structure.

(b) Support structure for three replaceable subsystem modules.

(c) Thermal shielding compatible with shielding and/or radiation requirements of the individual subsystem module and of the OA/RCS module.

(d) Support Provisions for:

(1) Six separation spings and the Spacecraft orbital release machanisms compatible with Delta interstage adapter.

(2) Six Spacecraft Shuttle support fittings and three passive probes for Shuttle orbiter flight support system (retrieval).

(3) Mounting three subsystem and one OA/RCS module.

(4) Omni antennas - (2) S-band.

(5) Launch pad and Shuttle orbiter umbilical disconnects.

(6) Pyro controls for release and deployment units.

(7) Attach fittings of the mission peculiar instrument section.

3.7.1.4.4.2 Subsystem Modules

Each subsystem component complement shall be mounted in a structure assembly such as shown in Fig. 3-29 shall contain:

(a) A 48 x 48 x 18 inch tubular frame with structural provisions for mounting three latching mechanisms and a minimum of three guide and roller assemblies.

(b) A 48 x 48 inch aluminum honeycomb prime mounting and heat radiating surface outboard.

(c) Internal partitions and shelves as required to structurally isolate subsystem items.

(d) Thermal shielding and/or radiation devices as determined by individual module useage requirements.

(e) Support and/or attachment for electronic interfacing.

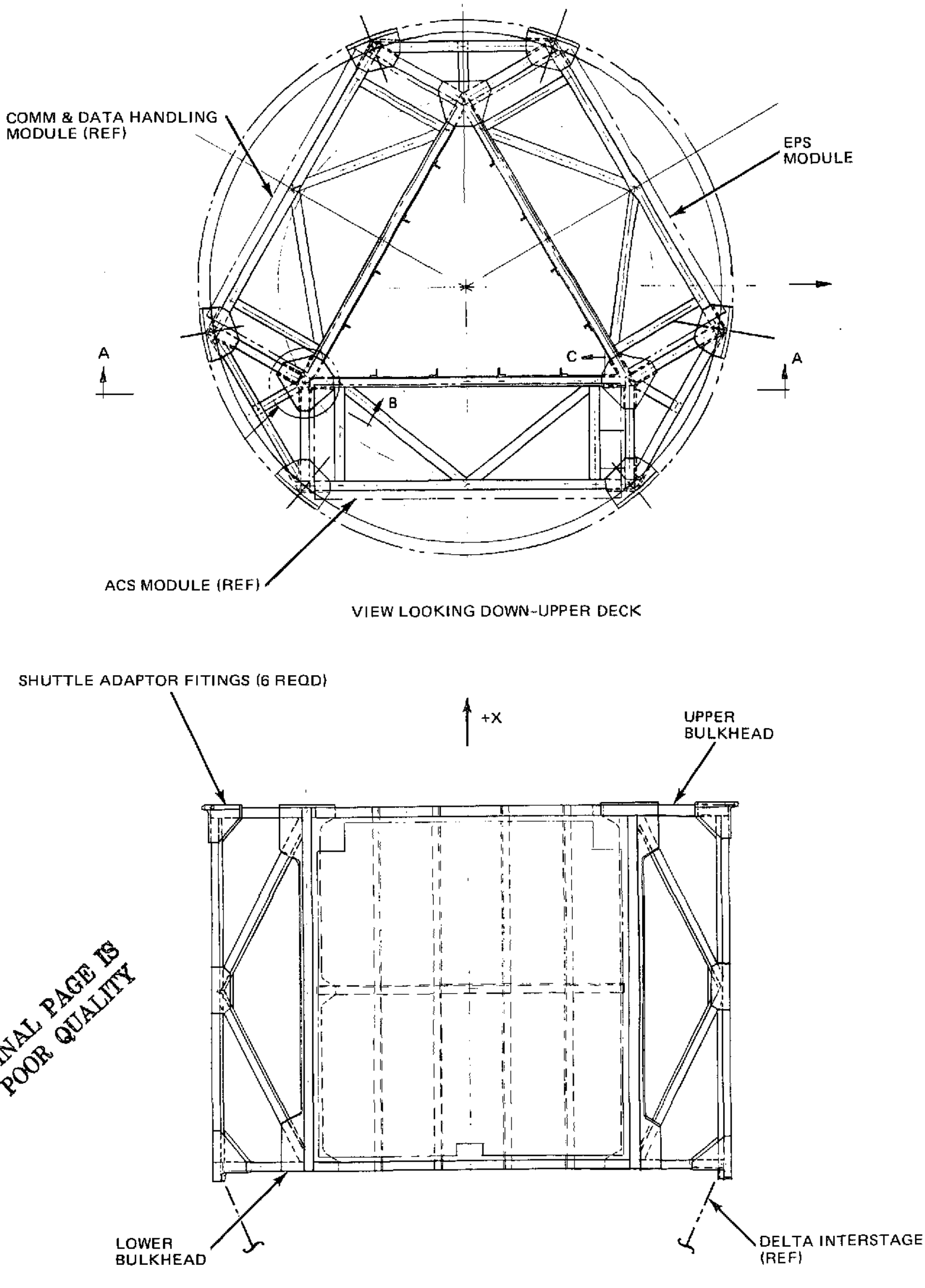

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Fig. 3-28 Basic Spacecraft Structural Arrangement

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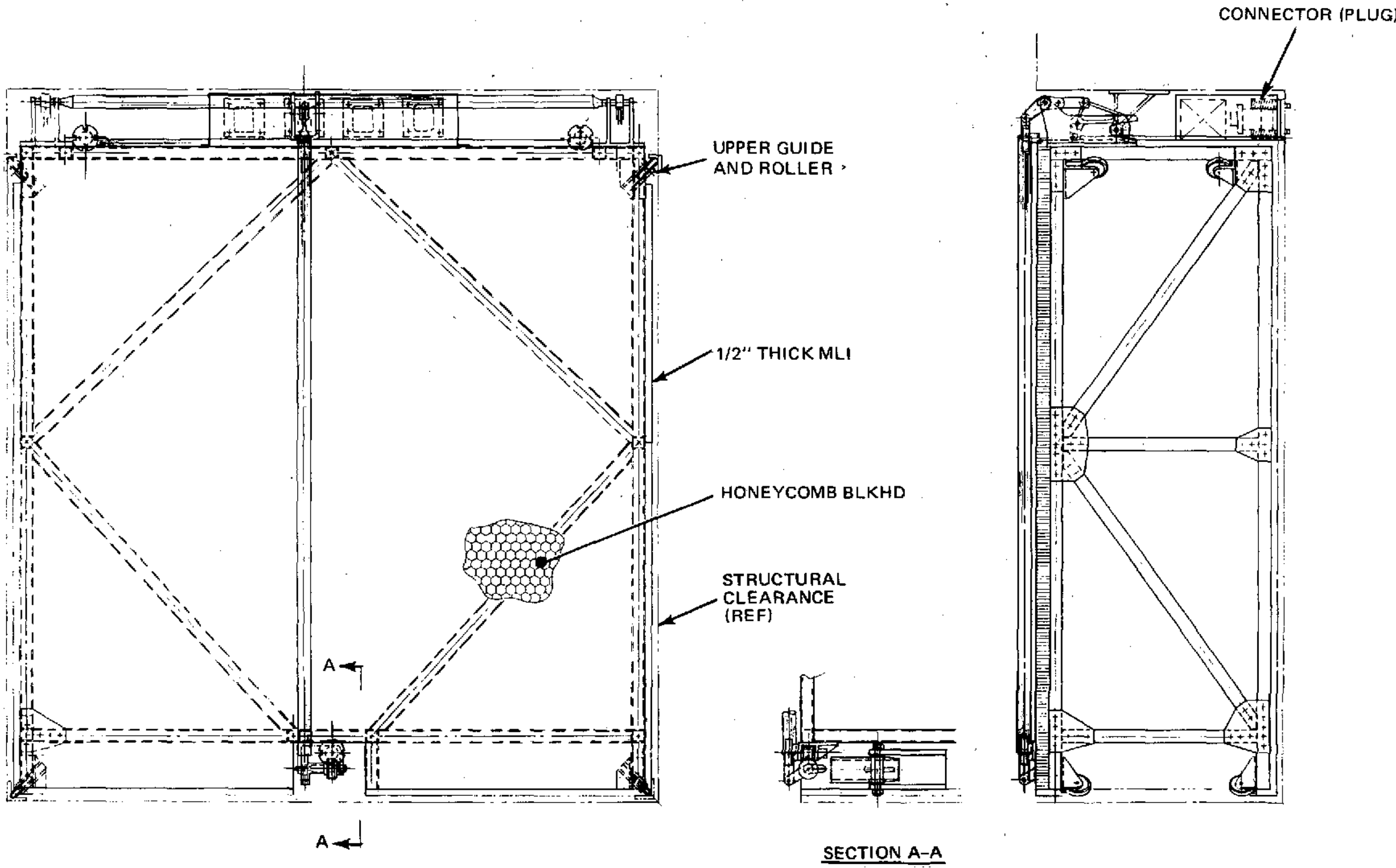

Fig. 3-29 Subsystem Module Structural Assembly

3.7.1.4.4.3 Orbit Adjust/Reaction Control System

Module shall be combined in a triangular structure demountable from the lower bulkhead of the Spacecraft core structure as shown in Fig. 3-30. It shall be capable of enclosure by the S/C to L/V adapter and shall contain:

(a) Four RCS thruster assemblies.

(b) Fuel tankage (2 units) to service the thrusters.

(c) Provisions for standardized on-orbit resupply latching system fittings.

3.7.1.5 THERMAL SUBSYSTEM

The thermal subsystem shall continuously maintain temperatures of the basic spacecraft within specification limits while under all potential combinations of external environment and equipment power.

3.7.1.5.1 General Requirements

3.7.1.5.1.1 Equipment and Structure Temperatures - Operating Mode

The minimum and maximum allowable operating temperatures for all equipment shall be determined from the reliability-life requirements of Paragraph 3.2.3.2. Equipment qualification test temperatures shall be determined by taking 25 percent of the difference between minimum and maximum allowable operating temperatures and appropriately subtracting and adding this value to the min/max allowable operating temperatures. The design goal operating temperatures shall be as follows:

(a) Communications and Data Handling Subsystem
Heat Sink $21^oC \pm 11^oC$

(b) Attitude Control Subsystem Heat Sink
$21^oC \pm 11^oC$
Gyro Operating Temperature TBD

(c) Electrical Power Subsystem Heat Sink
$21^oC \pm 11^oC$
Batteries - 1.1^oC (based on minimum power dissipation) to 10^oC (based on nominal power dissipation)

(d) Orbit Adjust/RCS Module
4.4^oC to 37.7^oC

(e) Structure Temperature

(1) Subsystems Structure - as specified in EOS-SS-240

(2) Orbit Adjust Structure - as specified in EOS-SS-240

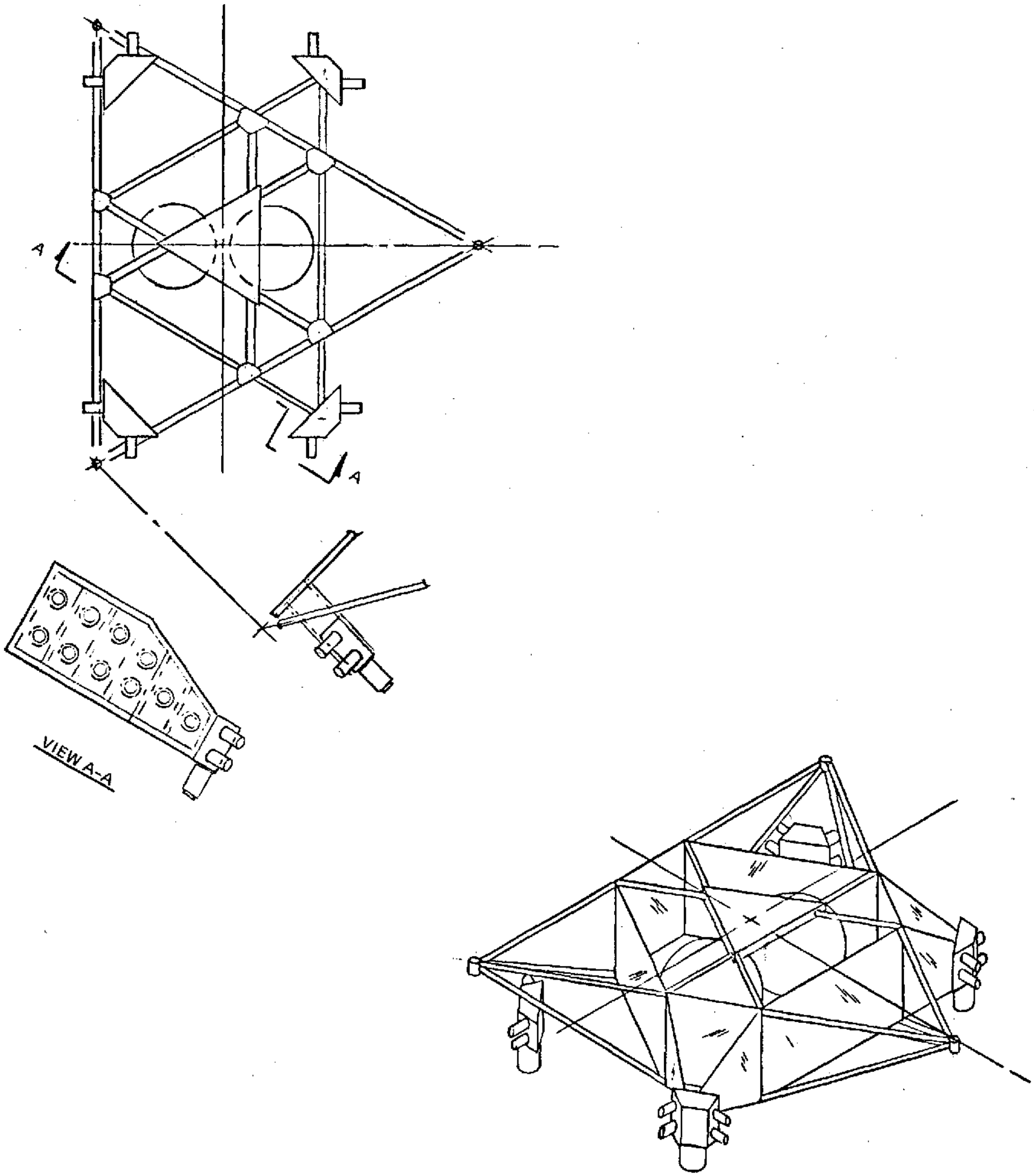

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Fig. 3-30 Orbit Adjust/RCS Module Configuration

In all cases design goal temperatures shall not exceed minimum and maximum allowable operating temperatures. The design goal temperatures shall be the minimum and maximum temperatures predicted for worst case cold and hot operation. The effects of time of year, operating duty cycles and surface property degradation shall be included in worst case predictions. Statistical variations, including external environment tolerances, surface properties, conductance and insulation effectiveness shall be included in the minimum and maximum predicted temperatures.

3.7.1.5.1.2 Equipment and Structure Temperatures - Survival Mode

The thermal design shall provide capability to maintain temperatures during a survival mode. The general temperature range requirement for the survival mode shall be $-40^{\circ}C$ to $66^{\circ}C$, unless otherwise specified in EOS-SS-240.

3.7.1.5.1.3 Control

The prime approach to achieve the Spacecraft thermal design shall use passive control. If it can be demonstrated that passive techniques cannot provide the required heat rejection capability or result in excessive temperature gradients or excessive heater power requirements, then active control shall be used. In all cases, preferences shall be given to qualified thermal control hardware and materials. Coating materials in critical areas shall have stability to provide the required radiant properties in the space environment to achieve the specified lifetime of the Spacecraft.

3.7.1.5.2 Subsystem Functions

The Thermal Subsystem shall maintain the Spacecraft equipment and structure temperatures within the limits specified for all mission phases.

3.7.1.5.3 Configuration

The prime approach for achieving temperature control shall be passive. Active control shall be implemented if required, as stipulated in Paragraph 3.7.1.5.1.3. The subsystem modules shall be designed to reject all equipment heat dissipation to space. The thermal design shall minimize module heat sink and structure temperature gradients. The modules and structure stages shall be made thermally independent of each other, using insulation and low conductance mounts.

3.7.1.5.3.1 Passive Control

The following thermal control hardware shall be considered passive:

(a) Thermal control skins and surface finishes

(b) Multilayer thermal insulation blankets

(c) Conductive path materials control

(d) Heater circuits, temperature controlled and/or ground commandable.

3.7.1.5.3.2 Active Control

The following thermal control hardware shall be considered active:

(a) Louvers

(b) Heat pipes - all types

3.7.1.5.4 Modes

The prime mission modes to be considered for the Spacecraft thermal design and corresponding temperature requirements are as follows:

(a) Prelaunch - design goal temperatures

(b) Launch and Boost - design goal temperatures

(c) Orbit-Operating (2 yrs) - design goal temperatures

(d) Orbit-Survival (3 yrs) - survival temperatures

(e) Shuttle-Retrieve - survival temperatures

3.7.1.5.5 Performance Requirements

The thermal subsystem shall continuously maintain all Spacecraft temperatures for all modes specified.

The analysis required to achieve the thermal design shall include:

(a) A detailed power load analysis, defining all equipment operating modes. This load analysis shall be updated periodically to incorporate measured value data and the specific requirements of each mission.

(b) An orbital heat flux analysis to determine the worst case heat fluxes for each critical spacecraft surface.

(c) Equipment thermal analysis for each unit to verify that the reliability requirements of Paragraph 3.7.1.5.1.1 are achieved.

(d) A detailed thermal nodal model for each common subsystem module and OA/RCS module.

(e) A detailed comprehensive thermal nodal model of the entire Observatory, including mission peculiar items as specified in Paragraph 3.7.3.1.5.5. In addition to the design function, this model shall be used to generate test and flight predictions and to correlate this data.

3.7.1.5.6 Interface Requirements

The thermal subsystem interface requirements shall be as follows:

(a) Power - as specified in Paragraph 3.7.1.2

(b) Remote unit interfaces for telemetry and commands as specified in Paragraph 3.7.1.1.5.2

(c) Mechanical - as specified in Paragraph 3.7.1.4.

3.7.1.5.7 Instrumentation Requirements

Location of equipment and structure temperature telemetry and heater command requirements shall be as specified in EOS-SS-240. Test instrumentation requirements shall be specified in each program test plan.

3.7.1.5.8 Ground Support Equipment

During all ground testing and checkout of the Spacecraft, provisions shall be made to monitor all Spacecraft temperature telemetry. A warning system to indicate out of limit conditions shall also be provided.

The launch facility shall provide ground cooling capability while the Spacecraft is at the launch complex. The cooling provisions shall meet all the environment requirements specified for the Spacecraft and shall be capable of maintaining Spacecraft temperatures within design goal limits during all operations at the launch complex.

3.7.1.6 ORBIT ADJUST/REACTION CONTROL SUBSYSTEM MODULE

The OA/RCS Module shall provide propulsive power for the translation and rotation maneuvers required to finalize and maintain the desired S/C orbit and to satisfy requirements of the ACS for desaturation of reaction wheels and three axis attitude control when wheels are inoperative. The OA/RCS design, within the module, shall be a monopropellant hydrazine, catalytic thruster design operating in a blowdown mode and shall be compatible with Shuttle operations (deployment/retrieval).

3.7.1.6.1 General Requirements

The OA/RCS provide reaction torques to control all internal and external disturbance torques. The OA/RCS module shall satisfy all position and attitude control requirements for translation and rotation of the S/C, as established by mission analysis and control system studies. The OA/RCS shall also satisfy the safety requirements for manned flight when contained within or operating in proximity to the Shuttle orbiter.

3.7.1.6.2 Operational Functions

The OA/RCS module shall perform the following propulsion functions:

(a) Three axis attitude control prior to, during and after the orbit injection error correction burn, up to and including solar array deployment.

(b) Correction of orbit injection errors in velocity and position caused by 3-sigma launch vehicle velocity dispersions.

(c) Orbit keeping

(d) Wheel unloading (>20% allocated for peak loads)

(e) Coarse attitude control (survival mode) in the event of a wheel/magnetic torquer failure.

The OA/RCS module shall perform the following safety functions:

(a) Fail-safe operation of thrusters

(b) Automatic pressure relief for all over-pressure conditions of propellant tanks.

(c) Venting of propellant tank pressure to a safe level (<20 psia).

(d) Retention of propellant within tank(s) under Shuttle crash loads.

3.7.1.6.3 Configuration

The OA/RCS shall consist of the following equipment arranged as shown in Fig. 3-31.

(a) Propellant Tank(s)

(b) Propellant Filter(s)

(c) Isolation Valve(s)

(d) RCS Thruster(s) (low level)

(e) RCS Thruster(s) (high level)

(f) OA Thruster(s)

(g) Fill/Drain Quick Disconnect(s)

(h) Fill/Vent Quick Disconnect(s)

(i) Pressure relief valve and burst disk assembly(s)

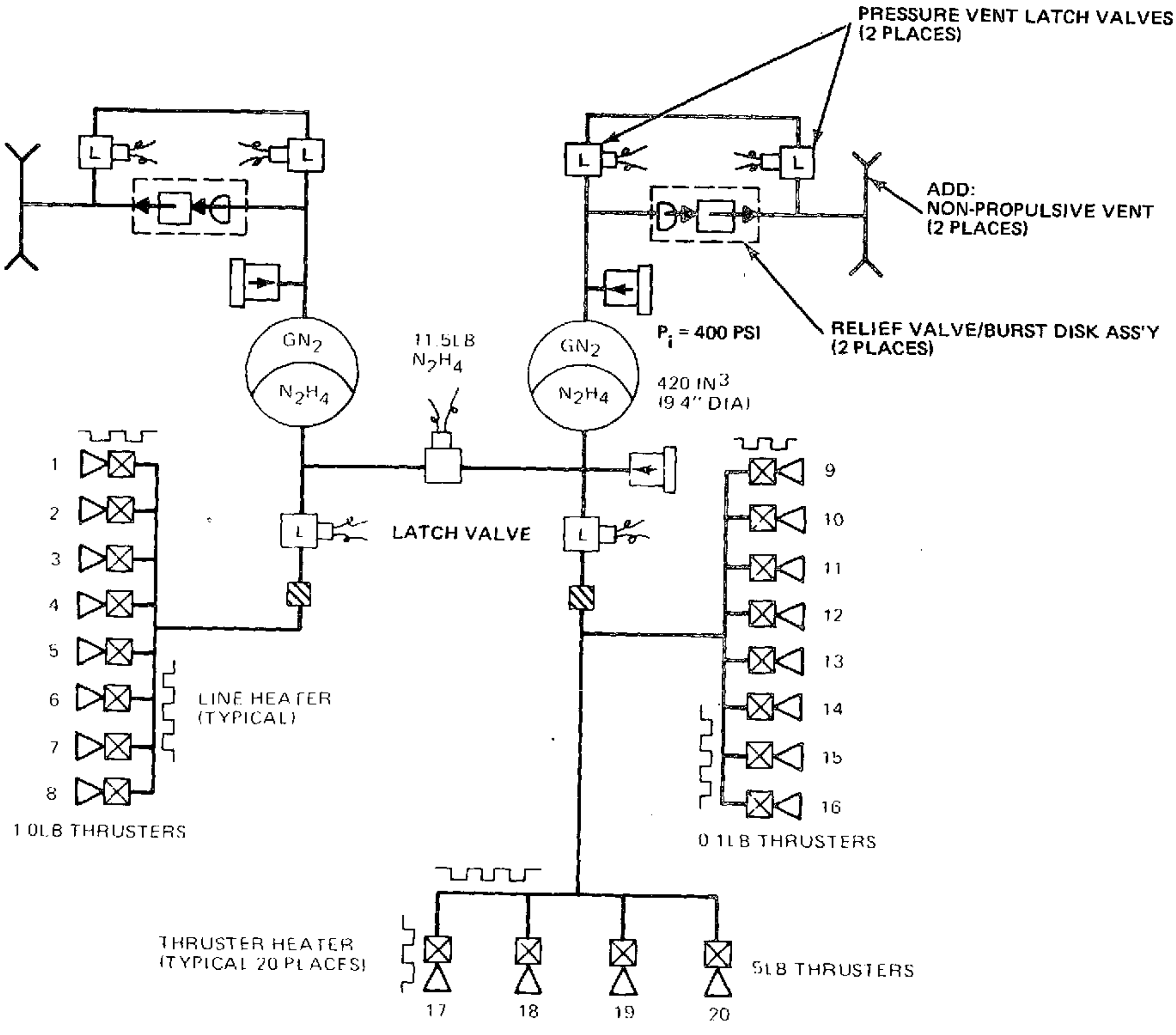

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Fig. 3-31 OA/RCS Schematic

(j) Vent Valve(s)

(k) Heaters

The following additional equipment shall also be mounted within the module:

(a) Bus Protection Assembly

(b) Spacecraft Interface

(c) Test Connectors

(d) Signal Conditioner

(e) Remote MUX/Decoder

(f) Wiring Harness

3.7.1.6.4 OA/RCS Modes

The OA/RCS shall be capable of operating in the following modes:

(a) Nominal

(b) Off-Nominal

(c) Survival

3.7.1.6.4.1 Nominal Mode

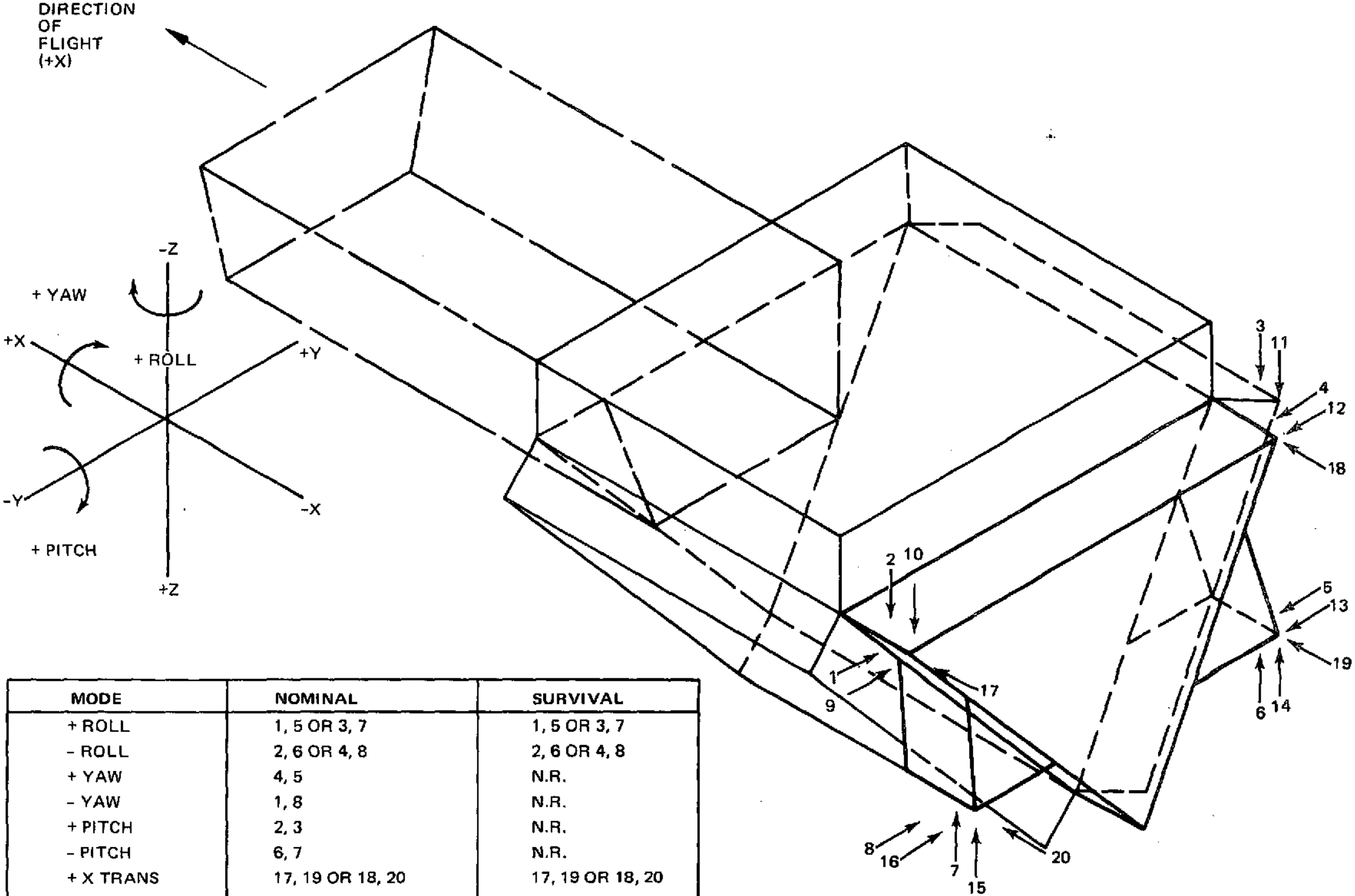
Nominal mode shall be defined by the isolation valving configuration shown in the schematic Fig. 3-31 and the nominal thruster firing logic shown in Fig. 3-32. In the nominal mode, all roll maneuvers shall be accomplished by coupled thruster firings to eliminate undesired torques and/or translations. It is a design goal to accomplish all rotational maneuvers by coupled thruster firings.

3.7.1.6.4.2 Off-Nominal Mode

Off-nominal or degraded operation of the OA/RCS shall be defined on the basis of a failure modes and effects analysis (FMEA).

3.7.1.6.4.3 Survival Mode

This mode shall be employed when a failure such as a failed open orbit adjust thruster or a jammed wheel occurs. The survival mode thruster firing logic of Fig. 3-32 is representative of the operation in this mode for the assumed failure of the roll wheel jamming. In the survival mode, the RCS shall be capable of performing a coarse attitude hold operation.



MODE	NOMINAL	SURVIVAL
+ ROLL	1, 5 OR 3, 7	1, 5 OR 3, 7
- ROLL	2, 6 OR 4, 8	2, 6 OR 4, 8
+ YAW	4, 5	N.R.
- YAW	1, 8	N.R.
+ PITCH	2, 3	N.R.
- PITCH	6, 7	N.R.
+ X TRANS	17, 19 OR 18, 20	17, 19 OR 18, 20
-X TRANS	ROTATION REQ.	N.R.

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Fig. 3-32 OA/RCS Thruster Firing Logic

3.7.1.6.5 Performance Requirements

3.7.1.6.5.1 Impulse

The OA/RCS shall deliver a minimum total impulse of 4980 lb-sec to perform orbit adjust maneuvers, attitude control and wheel unloading which shall be delivered with a maximum propellant weight of 22.9 lb of propellant. Five-lb_f thrusters shall be used for orbit adjust maneuvers, 1.0 lb_f thrusters for attitude control and 0.1 lb_f thrusters for wheel unloading. Up to double the total impulse/propellant weight can be provided with the addition of propellant tankage.

The selected propellant storage capacity shall allow for a 10 percent contingency growth in total impulse.

3.7.1.6.5.2 Propellants and Pressurant

Propellants shall be anhydrous hydrazine (N_2H_4) per MIL-P-26536 and pressurant shall be gaseous nitrogen (GN_2) per MIL-P-27401.

3.7.1.6.5.3 Operating Pressure

The OA/RCS operating pressure shall be 400 psia maximum at 120°F.

3.7.1.6.5.4 Leakage

The OA/RCS total leakage shall not exceed 100 cc/hr GN_2 when the OA/RCS is pressurized internally on both sides of the propellant tank diaphragm with GN_2 at 400 psia. Exclusive of thruster valve seat internal leakage, the OA/RCS system leakage shall not exceed 1 cc/hr GN_2.

3.7.1.6.5.5 Equipment Performance Requirements

The major equipment performance requirements shall be as summarized below.

3.7.1.6.5.5.1 Thrusters

The high-level RCS thrusters shall provide corrective torques in pitch, roll and yaw during initial stabilization, orbit injection error correction burns and restabilization following solar array deployment. The low-level RCS thrusters shall perform all other RCS functions. The OA thrusters shall provide translational delta velocity for orbit injection error correction and orbit keeping. Nominal and worst case thrust, impulse bit, and specific impulse characteristics and their repeatability shall be specified for beginning and end-of-life conditions.

Thruster orientation combined with the aft mounting arrangement of the OA/RCS module shall be designed to avoid the undesirable effects of heating, contamination and extraneous torques and translations due to plume impingement.

3.7.1.6.5.5.2 Propellant Tanks

The propellant tanks shall be designed to operate in a blowdown mode. An elastomeric diaphragm shall provide positive expulsion capability under all conditions of zero gravity and angular and translational accelerations. The pressurant (nitrogen gas) shall be supplied to one side of the diaphragm, with hydrazine monopropellant (N_2H_4) on the other. Diaphragm material shall be AF-E-332 or equivalent,

The propellant tanks shall be sized to accommodate propellant for all translation and attitude maneuvers required for two years of operation plus three years operation in the survival mode. Propellant quantities shall reflect the thruster performance characteristics projected for the impulse bit size, number of thruster activations, duty cycle and pressure as a function of time anticipated for the mission.

The propellant tanks shall be designed to retain the propellant under Shuttle crash loads.

3.7.1.6.5.5.3 Isolation Valves

The isolation valves shall be used to isolate failures in the OA/RCS. The latching isolation valve shall incorporate a position indicator switch.

3.7.1.6.5.5.4 Relief Valve and Burst Disk Assembly

The relief valve and burst disk assembly shall provide an automatic pressure relief capability for propellant tank over-pressure conditions while the Spacecraft is contained within the Shuttle Orbiter cargo bay.

3.7.1.6.5.5.5 Vent Valves

The vent valves shall be capable of reducing the propellant tank pressure to a safe level prior to Shuttle Orbiter retrieval/resupply or prior to a Shuttle Orbiter abort re-entry.

3.7.1.6.5.5.6 Heaters

Thermostatically controlled heaters shall be incorporated to maintain thruster temperatures as required to assure thruster life. If required, thermostatically controlled heaters shall be incorporated on thruster pod feed lines to prevent freezing of the hydrazine propellant.

3.7.1.6.6 Physical Requirements

3.7.1.6.6.1 Mass Properties

The mass properties of the OA/RCS module are given in Paragraph 3.2.2.1.1.

3.7.1.6.6.2 Dimensional and Volume Limitations

The dimensional and volume limitations shall be as indicated in Specification EOS-SS-250 and on subassembly drawings and assembly drawings.

3.7.1.6.6.3 Plume Impingement

The OA/RCS thruster location and arrangement shall minimize impingement of the plume on the S/C, solar array and payload sensors. The OA thruster nozzles shall point aft and the RCS thruster nozzles shall point radially to assure rapid dissipation of the thruster exhaust products from the payload sensors field of view.

3.7.1.6.6.4 Proof and Burst Pressure Factors

All pressure loaded components of the propulsion subsystem shall be designed for a minimum proof pressure of 2.0 times the maximum operating pressure and a minimum burst pressure of 3.0 time the maximum operating pressure, except for small diameter tubing and fittings which shall be designed for a minimum proof pressure of 2.0 times the maximum operating pressure and a minimum burst pressure of 4.0 times the maximum operating pressure.

3.7.1.6.6.5 Cleanliness

All RCS components shall conform to the cleanliness criteria specified in Specification EOS-SS-250. All lines and fittings shall be cleaned prior to assembly.

Assembly shall be performed in a clean room environment to class 10,000 of Federal Standard 209.

3.7.1.6.7 Interface Requirements

3.7.1.6.7.1 Electrical Interfaces

All OA/RCS Module electrical interfaces with the Spacecraft shall be via the Spacecraft Interface Connector(s). Umbilical Connector interfaces with prelaunch and resupply equipment shall also be via the Spacecraft Interface Connector(s). Electrical interfaces, for prelaunch test operations only, shall be made via the Test Connectors. All electrical interfaces between assemblies and interface connectors within the OA/RCS Module shall be made with the OA/RCS Module Harness.

3.7.1.6.7.1.1 Connectors

Spacecraft Interface Connector - The Spacecraft Interface Connector shall provide the electrical connect/disconnect between module and Spacecraft. The connector(s) shall be designed for and physically positioned to assure interchangeability of modules. Specific design requirements shall be: blind mate capability; anti-bind roll-off shell (angular disconnect capability); maximum axial movement of structure without affecting continuity, and highly reliable contacts with self-aligning capabilities.

Test Connectors - Test connectors shall be provided as applicable on major assemblies and at one side of the subsystem module. The test connectors shall provide the capability, to the maximum extent possible, to determine degradation or the flight worthiness of the assemblies and module without the need for demating connectors in flight circuits. All outputs to test connectors shall contain isolation circuitry.

3.7.1.6.7.1.2 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 subsystem electrical assemblies. It shall be possible to remove electrical assemblies without the harness. Harness and cable assembly practices shall meet the intent of MIL-W-5088 unless specified otherwise in this specification.

Wire types shall be lightweight, abrasion resistant, and space qualified. Coaxial cable shall conform to MIL-C-17. Wire size shall be determined by: circuit steady state current; voltage drop compatible with unit performance requirements; thermal environment; connector termination capabilities; minimum wire gauge 24 awg high strength copper alloy; bundle capacity, and minimum weight. Connectors shall be of the removal crimp contact type where feasible and shall meet environmental requirements for space application.

3.7.1.6.7.1.3 Power

The electrical power required to operate the OA/RCS module will be provided by the Power Module. The prime power shall be at a voltage of +28 ±7 VDC with detailed characteristics as defined in Specification EOS-SS-250. Any conditioning of the nominal +28 VDC power to other power types, voltage levels or regulation tolerance shall be provided within the OA/RCS module.

The OA/RCS module power distribution circuitry shall contain devices to protect the power busses from short circuits. The bus protection circuitry shall be provided for all loads except those which are nonredundant and critical to mission success. Protected loads and detailed bus protection requirements shall be in acoordance with Specification EOS-SS-250.

3.7.1.6.7.2 Command and Data Handling Interface

A remote multiplexer and decoder unit shall be dedicated to the OA/RCS module to provide a standard interface between module data and command signals and the multiplex data bus system which is controlled from within the C&DH module. The remote unit shall be capable of providing the signal input and output interface as defined in Paragraph 3.7.1.1.5.2.3.

3.7.1.6.7.2.1 Telemetry

The OA/RCS shall incorporate temperature sensors, pressure transducers, and isolation valve status/position indicators. The temperature sensors and position indicators outputs shall be signal conditioned. The pressure transducer output shall be directly proportional to propellant tank pressure and shall be signal conditioned within the transducer.

3.7.1.6.7.2.2 Commands

Commands to operate the OA/RCS latching isolation valves and heaters are listed in Table 3-12. Thruster firing commands are provided by the Attitude Control Subsystem.

Table 3-12 OA/RCS Commands

COMMAND/FUNCTION	SIGNAL TYPE
TANK ISOL VLVE #1 O/C	P
TANK ISOL VLVE #2 O/C	P
TANK XOVER ISOL VLVE O/C	P
0.1# THRUSTR CAT. BED HTRS ON/OFF	P
1.0# THRUSTR CAT. BED HTRS ON/OFF	P
5.0# THRUSTR CAT. BED HTRS ON/OFF	P

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3.7.1.6.7.3 Mechanical Interface

The mechanical interfaces between the OA/RCS module and the Spacecraft structure shall be in accordance with Paragraph 3.7.1.4.

3.7..1.6.7.4 Thermal Interfaces

The thermal design shall maintain all of the OA/RCS module equipment and structure within the limits specified for all mission phases. The OA/RCS module general operating temperature range shall be $4.4^{o}C$ to $37.7^{o}C$. Specific components requiring deviation from this value and temperatures for all modes shall be as specified in Paragraph 3.7.1.5.

The OA/RCS module shall be made thermally independent from the other subsystem modules and structure, using insulation and low conductance mounts. Catalyst bed heaters shall be incorporated on each thruster to assure thruster performance/life requirements are met.

3.7.1.6.7.5 Attitude Control Subsystem Interface

The OA/RCS thruster valves shall respond to driver signals provided by the Attitude Control Subsystem.

3.7.1.6.7.6 Subsystem/Ground Servicing Equipment Interfaces

Propellant and pressurant shall be loaded and drained from the system using a GSE cart.

Fluids entering the OA/RCS shall be filtered to 5 micron absolute. Fluid connections between the GSE and the OA/RCS shall be made at the three fill and drain valves of the OA/RCS. No dripping of propellant on external surfaces of the Spacecraft shall be permitted during servicing operations.

3.7.1.6.8 Instrumentation Requirements

The OA/RCS shall incorporate sufficient instrumentation to provide for OA/RCS control and failure analysis. As a minimum, the instrumentation listed in Table 3-13 shall be provided.

3.7.1.6.9 Ground Support Equipment

The OA/RCS ground support equipment is defined in Paragraph 3.1.3.1.

Table 3-13 OA/RCS Instrumentation Requirements

MEASUREMENT	NO./TYPE	RANGE	ERROR
PROPELLANT SUPPLY PRESSURE	2 ANALOG	0-400 PSIA (0-5.0 VDC)	± 2.0% FULL SCALE
PROPELLANT TANK TEMPERATURE	2 ANALOG	0 TO +120° F	± 3° F (0-120° F)
THRUSTER ASSEMBLY TEMPERATURE	20 ANALOG	-50 TO +250° F	± 5° F (25-200° F)
LATCHING VALVE POSITION	3 BI-LEVEL	OPEN/CLOSE	DNA

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3.7.1.7 ELECTRICAL INTEGRATION

3.7.1.7.1 General Requirements

Electrical Integration relates to all electrical subsystems and is required to unite them into an effective functioning system.

3.7.1.7.2 Functions

3.7.1.7.2.1 Power Distribution

Unregulated +28 VDC power shall be distributed to all Spacecraft/Observatory loads. The main power busses shall distribute power to each subsystem module and to the instruments. There shall be means of protecting the main bus/module interface from single point failure. The assemblies in each module and each instrument shall be redundantly fused to protect the Spacecraft bus. Power line voltage drop shall be compatible with subsystem requirements.

Power to all loads shall originate in and be returned to a distribution bus in the power subsystem module.

3.7.1.7.2.2 Signal Distribution

Distribution of command and telemetry data between the C&DH module and the other subsystem modules and the instruments shall be handled via remote decoders and multiplexers as described in Paragraph 3.7.1.1. This party line method of signal distribution shall be designed to minimize the need for numerous interface connections at the module/structure interfaces and provide significant immunity to noise.

Single conductor, twisted pair, single conductor shielded, multi-conductor shielded and coaxial cabling shall be used for the various types of signals to be distributed. Selection of the type used shall be based on the characteristics of the signals and the source and input impedances of the output/input circuitry. For short runs of high level, low frequency binary data and impendance analog data, unshielded wiring shall be used to minimize harness weight and simplify harness fabrication. Shielded or coaxial cable shall be used for most of the other categories of signal circuitry.

3.7.1.7.3 Configurations

The main structure harnesses shall be capable of being installed as integral assemblies where feasible. The main harness shall be divided into three major segments: Spacecraft; pyrotechnic/actuator, and instrument.

3.7.1.7.3.1 Spacecraft Harnesses

The Spacecraft harnesses shall supply all electrical interfaces between modules, launch instrument harness, umbilical, and peripheral equipment necessary for total system function. As a result of this function, there shall be some differences due to mission peculiar equipment.

3.7.1.7.3.2 Pyrotechnic/Actuator Harnesses

Harnessing for the pyrotechnic and/or actuator citcuitry shall be similar in design and configuration to the Spacecraft harnesses. The wiring and circuitry shall be in accordance with SAMTECH Range Safety Manual 127-1 and EMC Specifications, Paragraph 3.3.2. The mission to mission uniqueness, dictates this harness to be mission peculiar.

3.7.1.7.3.3 Instrument Harness

The instrument harness shall be designed as a replaceable assembly interfacing with the Spacecraft harness at one interface. The actual configuration of the harness shall vary according to the types of instruments used and their overall requirements and is considered mission peculiar.

3.7.1.7.4 Requirements

3.7.1.7.4.1 Electromagnetic Compatibility

Requirements for electromagnetic compatibility, including grounding are defined in Paragraph 3.3.2.

3.7.1.7.4.2 Redundancy

Redundant wiring shall be provided primarily in power, control, and primary data circuits to assure that critical Observatory functions are maintained for the planned life of the Observatory. Connector contact redundancy shall be provided at all connector interfaces containing circuits critical to proper Observatory operation.

3.7.1.4.3. Spacecraft Interface Connector

The Spacecraft Interface Connector shall provide the electrical connect disconnect between module and Spacecraft. The connector (s) shall be designed for, and physically positioned to assure interchangeability of modules. Specific design requirements shall be: blind mate capability; anti-bind roll-off shell (angular disconnect capability) maximum axial movement of structure without affecting continuity, and highly reliable contacts with self aligning capabilities. The interface connectors shall be compatible with Shuttle Resupply operation requirements for growth potential.

3.7.1.7.4.4 Harness Components

Harness and cable assembly practices shall meet the intent of MIL-W-5088 unless specified otherwise in this specification.

Wire types shall be lightweight, abrasion resistant, and space qualified. Coaxial cable shall conform to MIL-C-17. Wire size shall be determined by: circuit steady state current; voltage drop compatible with unit performance requirements; thermal environment; connector termination capabilities; minimum wire gauge 24 awg high strength copper alloy; bundle capacity, and minimum weight. Connectors shall be of the removable crimp contact type where feasible and shall meet environmental requirements for space application.

3.7.1.8 OBSERVATORY SOFTWARE

The Observatory software shall be prepared in modules which may be assembled and verified independently before linking to provide the software package for a specific spacecraft. Three classes of module are distinguished: basic software: adaptable software, and mission peculiar software.

3.7.1.8.1 Basic Software

Basic software consists of those software modules which, other than linkage addresses and status words, may be used in any mission without modification.

3.7.1.8.1.1 Executive Software Module

The Executive Software Module shall schedule the running of other modules of the Observatory software, provide for their initialization and start-up, and provide support subroutines for necessary mathematical functions.

3.7.1.8.1.2 Self-Test Software Module

The Self-Test Software Module shall verify the propter continuing operation of the Observatory software by monitoring running times, memory write protect changes and the Spacecraft clock, issuing warnings for computer action and ground monitoring as appropriate.

3.7.1.8.1.3 Program Change Software Module

The Program Change Module shall implement ground commanded program changes, with appropriate memory write protect clearing and resetting, with command validity checks, and before-and-after memory content reports.

3.7.1.8.1.4 Command Handling Software Module

The Command Handling Module shall accept and implement ground commands, providing the following functions:

(a) Command verification

(b) Command storage

(c) Command scheduling by time

(d) Command scheduling by location

(e) Command output to the Spacecraft systems

3.7.1.8.1.5 Mode Control Software Module

The Mode Control Software Module shall select the appropriate Spacecraft operating mode based on inputs describing the Spacecraft status and on ground commands.

3.7.1.8.1.6 Operations Scheduling Software Module

The Operations Scheduling Software Module shall choose the timing and type of experiment operations based on input criteria and time, location, sun angle and instrument status.

3.7.1.8.1.7 Data Compression Software Module

The Data Compression Software Module shall prepare statistical abstracts of selected spacecraft variables and store them for onboard use or for downlink reporting. Statistical terms provided shall include:

(a) Minimum value since reset

(b) Maximum value since reset

(c) Running mean

(d) Running variance.

3.7.1.8.1.8 History Software Module

The History Software Module shall store and report by downlink, occurrance of events of interest in terms of an event code number and a time of occurrance. Events recorded shall include:

(a) Spacecraft mode changes

(b) Thruster burn start and stop

(c) Experiment start and stop

(d) Data transmission periods

(e) Equipment failures.

3.7.1.8.1.9 Situation Assessment Software Module

The Situation Assessment Software Module shall examine the performance indicators of all Spacecraft functions, compare them to the performance required for the current mode, and set mode change or status change indicators for use by other software modules.

3.7.1.8.1.10 Computer Dump Software Module

The Computer Dump Software Module shall, on command, provide downlink outputs describing the contents of specific areas of the computer memory.

3.7.1.8.1.11 Stabilization Software Module

The Stabilization Software Module shall accept error signals from the Spacecraft sensing and guidance software modules and provide wheel torque, magnetic torque and thruster torque commands to maintain the Spacecraft in the desired stable attitude.

3.7.1.8.1.12 Position Computation Software Module

The Position Computation Software Module shall accept uplinked Spacecraft ephemeris data and spacecraft time inputs, and compute the current Spacecraft geocentric latitude, longitude and altitude. Interpolation terms for latitude and longitude shall be computed to permit determination of up-to-date values between computations.

3.7.1.8.1.13 Subsystem Service Software Module

The Subsystem Service Software Module shall provide worker routines to monitor and operate Spacecraft functions such as:

(a) Temperature controls

(b) Solar array

(c) Power system

(d) Propulsion system.

3.7.1.8.2 Adaptable Basic Software

Adaptable Basic Software consists of those software modules which, although they are required for all Observatory missions, require some internal changes for applicability to specific missions.

3.7.1.8.2.1 Downlink Software Module

The Downlink Software Module shall format data for downlink telemetry transmission and initiate reset and clearing of the data compression and history files when transmission is complete.

3.7.1.8.2.2 Guidance Software Module

The Guidance Software Module shall accept sensor inputs and attitude command inputs, and generate guidance commands to guide the Spacecraft to the desired attitude.

Limiting of guidance rates to acceptable levels will be required during experiment operation.

3.7.1.8.2.3 Sensing Software Module

The Sensing Software Module shall provide worker routines to monitor and operate the Spacecraft sensors required for Spacecraft operation such as:

(a) Magnetometer

(b) Rate integrating tyro

(c) Star tracker

(d) Digital sun sensor.

3.7.1.8.2.4 Prelaunch Test Software Module

The Prelaunch Test Software Module shall be designed to occupy memory areas which are overlayed after launch with stored command data, and shall verify the proper operation of all equipment which interfaces with the Observatory computer or the multiplex data bus.

3.7.1.8.2.5 Pre-Maneuver Test Software Module

The Pre-maneuver Test Software Module shall verify the proper status of all maneuver subsystems prior to orbit maneuvers, and provide readiness indicators for downlink to the control station.

3.7.1.8.2.6 System Monitor Software Module

The System Monitor Software Module shall perform tests of all subsystems during their idle times and output subsystem status indicators. Subsystems without appreciable idle time, such as the rate integrating gyros will not be tested.

3.7.1.8.2.7 System Troubleshoot Software Module

The System Troubleshoot Software Module, when initiated by failure indicators or by ground command, shall provide test procedures for subsystems not tested by the System Monitor Software Module, replacing the output of the subsystem under test with appropriate signals from other sources or from statistical data stored from previous orbits.

3.7.2 INSTRUMENT FUNCTIONAL CHARACTERISTICS

The instruments for the LRM mission A consist of two sensors, Multi-spectral Scanner and the Thematic Mapper.

3.7.2.1 Multi-Spectral Scanner (MSS)

This instrument is an adaption of the sensor previously used on the R&D ERTS satellite program. It is considered the operational sensor on LRM mission A. The instrument is used to map a 185 Km swath of the earth's surface directly below the spacecraft to a system resolution of 80 meters in four spectral bands in the visual region and to 300 meters in a band in the 10 micron region. The unit is an electromechanical scanner of the object plane type providing output data in digital form of 6-bit accuracy in a digital bit stream of approximately 16 megabits per second. For further details, see the MSS user handbooks available from the ERTS A/O EOS program office.

3.7.2.2 Thematic Mapper (TM)

This instrument shall provide a ground resolution of 30 meters and shall provide seven spectral bands (four visual, two IR and one thermal). The instrument shall employ a row of closely spaced sensors which scan the earth in panoramic style generating strips of data perpendicular to the satellite flights vector. The instrument shall have a swath width of 185 Km. The signal to noise ratio shall be selected to enhance system performance in the winter at high lattitudes where light levels are marginal. The output of the instrument shall be in digital form of seven bit accuracy at a total data rate of approximately 86 megabits per second. The data from each spectral band shall be available separately at the output of the sensor in order to provide system flexibility in data processing and increased system reliability. Detail requirements of this instrument, are presented in the Instrument Specification (Report No. 2 of the EOS System Definition Study) or the Instrument Interface Control Document 314-ICD-002.

3.7.3 MISSION PECULIAR EQUIPMENT

3.7.3.1 LAND RESOURECES MANAGEMENT MISSION A

3.7.3.1.1 Communications and Data Handling (C&DH)

3.7.3.1.1.1 Communications Group

The communications group shall be capable of transmitting additional telemetry rates through the TDRS. These shall include the following:

- Narrow Band Data Rate: Selectable, 32 Kbps, 16 Kbps in addition to 8 Kbps, 4 Kbps, 2 Kbps, or 1 Kbps.
- Medium Band Data Rate: 128 Kbps.

3.7.3.1.1.1.1 Configuration Impact

The delta change to the C&DH subsystem shall be the addition of a high gain TDRS S-Band steerable antenna diplexer, and a switch. The S-Band steerable antenna shall be combined with the Ku-Band high gain antenna with a dual S/Ku band feed.

3.7.3.1.1.1.2 Modes of Operation

The modes of operation that shall be expanded is for telemetry through the TDRS as follows:

(a) Real time low data rate, selectable

(b) Memory dump

3.7.3.1.1.1.3 Performance Requirements

3.7.3.1.1.1.3.1 Telemetry Link Considerations

The telemetry link considerations shall be the same as described in Paragraph 3.7.1.1.5.1.2.1.2 except that the required Observatory EIRP shall be 20.1 dBW.

3.7.3.1.1.1.3.2 Command Link Considerations

The command link considerations shall be the same as described in Paragraph 3.7.1.1.5.1.2.1.2.

3.7.3.1.1.1.3.3 Telemetry/Command Antenna

The telemetry/command antenna shall be a dual feeu S/Ku band steerable antenna as shown in Fig. 3-33. Characteristics shall be as listed in Table 3-14.

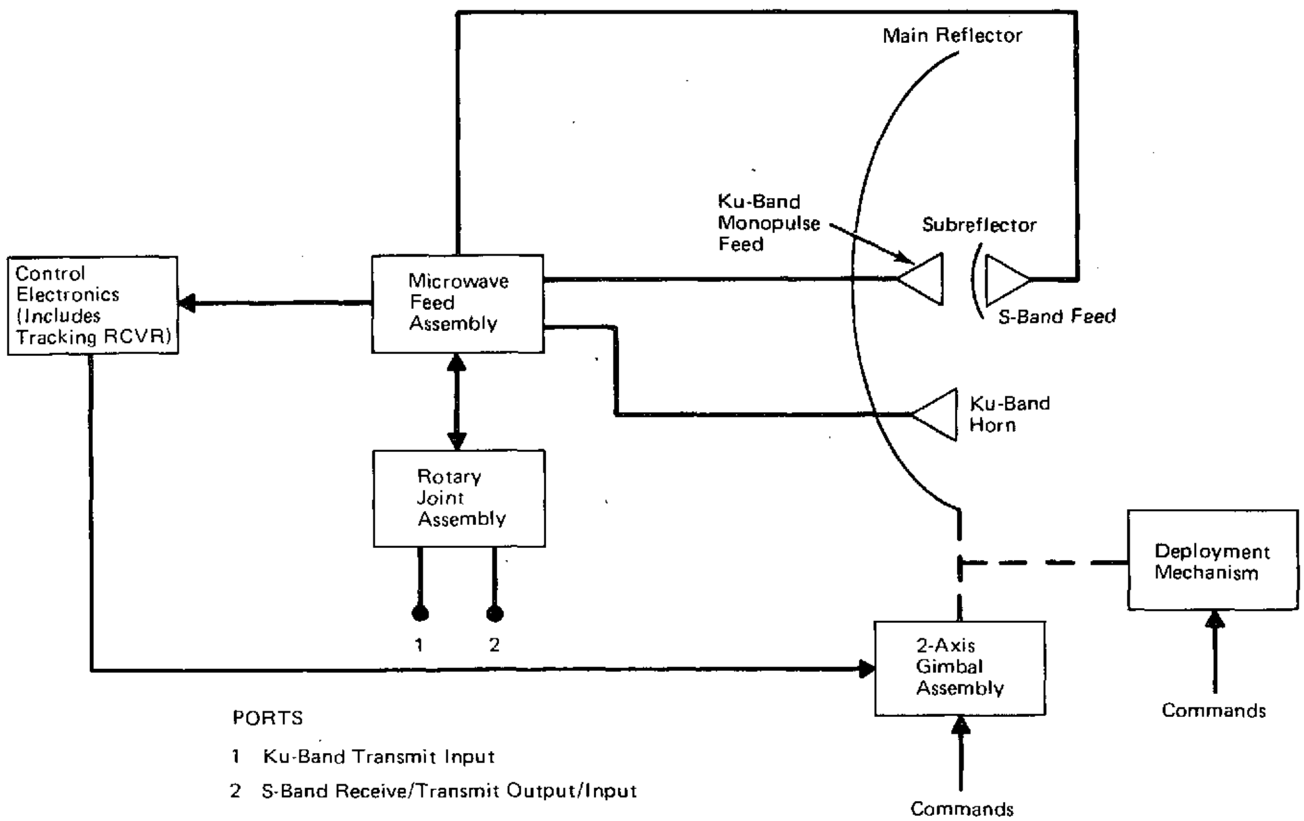

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Fig. 3-33 Dual Feed S/Ku-Band Steerable Antenna System – Block Diagram

Table 3-14 Dual Feed - S/Ku-Band Steerable Antenna Design Requirements

PARAMETER	S-BAND	KU-BAND HIGH GAIN	KU-BAND LOW GAIN
1. FREQUENCY, GHZ TRANSMIT RECEIVE	 2.025 TO 2.120 2.200 TO 2.300	 14.6 TO 15.2 13.6 TO 14.0	 14.6 TO 15.2
2. ANTENNA TYPE	PARABOLIC DISH	PARABOLIC DISH	OPEN ENDED WAVEGUIDE
3. FEED TYPE	PRIME FOCAL POINT	CASSEGRAIN	N/A
4. POLARIZATION	RHCP	RHCP	RHCP
5. AXIAL RATIO, DB, MAX	1.5	1.5	1.5
6. INPUT VSWR AT ROTARY JOINT OUTPUT	1.4:1	1.5:1	1.5:1
7. SIDE AND BACK LOBE LEVELS, DB	⩽ 17.0	⩽ 17.0	N/A
8. ANTENNA DISH SIZE (FT)	12.5 FREQ. (2.25 GHZ)	12.5 FREQ. (14.6 GHZ)	N/A FREQ. (14.6 GHZ)
9. NET ANTENNA GAIN (DB) MEASURED AT THE ROTARY JOINT INPUT (INCLUDES ALL FEED ILLUMINATION AND TRANSMISSION LINE COMPONENT LOSSES)	35	51	0 (MINIMUM WITH-IN 60° HPEW)
10. TRACKING CONFIGURATION	OPEN	CLOSED (PSEUDO MONOPULSE)	–
11. TRACKING ACCURACY, 3α	–	0.17 DEGREES	–
12. POINTING ACCURACY, 3α	–	0.05 DEGREES	–
13. GIMBAL STEP SIZE	–	0.02 DEGREES	–
14. SLEW RATE, VELOCITY ACCELERATION	 – –	 20 DEG/SEC MAXIMUM 60 DEG/SEC² MAXIMUM	 – –
15. SCAN ANGLE OFF-BORESIGHT, 2 AXIS (XY GIMBAL)	X (INNER) GIMBAL ± Y (OUTER) GIMBAL ±	90 DEGREES 110 DEGREES	
16. TOTAL WEIGHT, LBS INCLUDING FEED/MICROWAVE SYSTEM, ROTARY JOINTS, REFLECTOR, FEED SUPPORT, GIMBAL ASSEMBLY, CONTROL ELECTRONICS AND DEPLOYMENT HARDWARE		TBD	
17. TOTAL POWER (WATTS) • DEPLOYMENT • PEAK OPERATING POWER (INCLUDES MOTOR DRIVE AND MOTOR CLAMPING PLUS CONTROL ELECTRONICS) • AVERAGE POWER (INCLUDES MOTOR POWER SLEW, MOTOR POWER - TRACK AND CONTROL ELECTRONICS		 TBD TBD TBD TBD	
18. RELIABILITY	DESIGN FOR A 2 YEAR OPERATIONAL LIFE.		

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3.7.3.1.1.1.3.4 RF Coaxial Switch

The RF coaxial switch shall be the same as described in Paragraph 3.7.1.1.5.1.2.5.

3.7.3.1.1.2 Data Handling Group

The On Board Computer (computer memory, processor, input-output, power regulator, etc.) shall be designed to be capable of accommodating optional memory expansion to 24K words in 8K word segments.

3.7.3.1.2 Electrical Power

The major EPS mission peculiar equipment will consist of a Solar Cell Array and associated mechanisms and, if required, additional Power Module energy storage capability above the basic 40 Ampere-Hours.

The EPS shall be capable of providing to the spacecraft subsystems and payloads, an orbital average power of 525 W for the two-year operational life and 250 W for the subsequent three-year survival period. During the operational phase, a minimum of 200 W, orbital average power shall be available for the payloads.

3.7.3.1.2.1 Solar Cell Array

The Solar Cell Array shall be capable of providing, for the full duration of the mission, sufficient power to the total spacecraft/observatory average sunlight load plus recharge the Power Module batteries.

The array shall consist of auxiliary and main electrical power sections that are compatible with the overall EPS requirements defined in Paragraph 3.7.1.2.3. Mechanically, the array shall consist of multiple, rigid panels that are stowed in a folded configuration during launch and deployed after achieving orbit. Following deployment, the array shall be coplanar when mounted on the end of a Y (pitch) axis shaft that extends out of the -Y (sun) side of the Spacecraft. The mounting of the array at the end of the shaft shall include a pre-flight (ground only) adjustment capability that allows optimizing the angle formed by the shaft and the solar array plane so to account for the apparent out-of-orbital plane movement of the sun. The apparent movement of the sun in-the-orbit plane during each orbit shall be tracked by continuously rotating the array about the Y (pitch) axis. The solar array shall be designed for retraction during Shuttle Orbiter retrieval.

Additional requirements of the solar cell array are defined in Paragraphs 3.7.1.2 and 3.7.3.1.4. Detailed requirements shall be as specified in Specification EOS-SS-210 and Interface Control Drawing 314-ICD-001.

3.7.3.1.2.2 Energy Storage Capacity

The energy storage capacity of the Power Module for the LRM A shall be sufficient to satisfy the overall EPS performance requirements for the full duration of the mission. A minimum of two batteries shall be utilized with individual battery depths-of-discharge limited to values that are consistent with accepted design practices.

Detailed energy storage requirements shall be in accordance with Specification EOS-SS-210.

3.7.3.1.3 Attitude Control

The Size 1 reaction wheels and magnetic torquers are considered elements of the basic ACS. Size 2 and 3 reaction wheels and magnetic torquers are mission peculiar items, as are algorithms for the computer to be used during transfer orbit maneuvers. For the case in which one LRM A Observatory is placed in orbit, neither heavier reaction wheels and magnetic torquers nor transfer orbit maneuvers are required. In this case, then there are no ACS mission peculiars. For the case in which two LRM A observatories are placed in orbit, transfer orbit maneuvers are performed. In this case, mission peculiar algorithms are required for the transfer orbit maneuvers.

3.7.3.1.4 Structure (Instrument Support)

3.7.3.1.4.1 General Requirements

Requirements of Paragraph 3.7.1.4.1 apply.

3.7.3.1.4.2 Functions

The structure subsystem shall provide:

(a) All primary and secondary structures required to provide adequate support and protection to all instruments such that they will withstand all natural and induced environmental forces to which the Observatory shall be exposed during all ground and flight phases of the mission.

(b) Adequate rigidity in those areas where instruments and equipment items requiring critical alignment are mounted and whose geometry is critical to achieving mission objectives.

(c) For all required mechanical interfaces between the instrument section and:

(1) The Spacecraft

(2) Ground Support Equipment

(3) Launch pad handling equipment

(d) For stowage and deployment of the X-Band and S/Ku Band (TDRS), antennas, and EPS solar arrays.

(e) Sufficient space for adequate access to permit efficient preflight and on-orbit servicing, maintenance, and replacement of instruments and equipment items.

(f) For the prevention of structural deformations of a magnitude sufficient to:

(1) Cause structural failure

(2) Jeopardize the proper functioning of equipment items

(3) Endanger the functional characteristics of the Observatory at any time during all ground and flight phases of the mission.

3.7.3.1.4.3 Configuration

The structure subsystem shall consist of the instrument section structure as shown in Figure 3-34. The Instrument section for LRM-A shall contain:

(a) A lower equipment deck

(b) Vertical support panels

(c) An upper equipment deck

(d) Supports and/or attachments for

(1) Thematic Mapper (TM)

(2) Multi Spectral Scanner (MSS)

(3) Solar array stowage/erection supports

(4) Solar array drive motor

(5) S/Ku-Band (TDRS) steerable antenna

(6) X-Band steerable antennas (2)

(7) X-Band fixed antenna

(8) Instrument mission peculiar module

(9) Thermal control insulation blankets

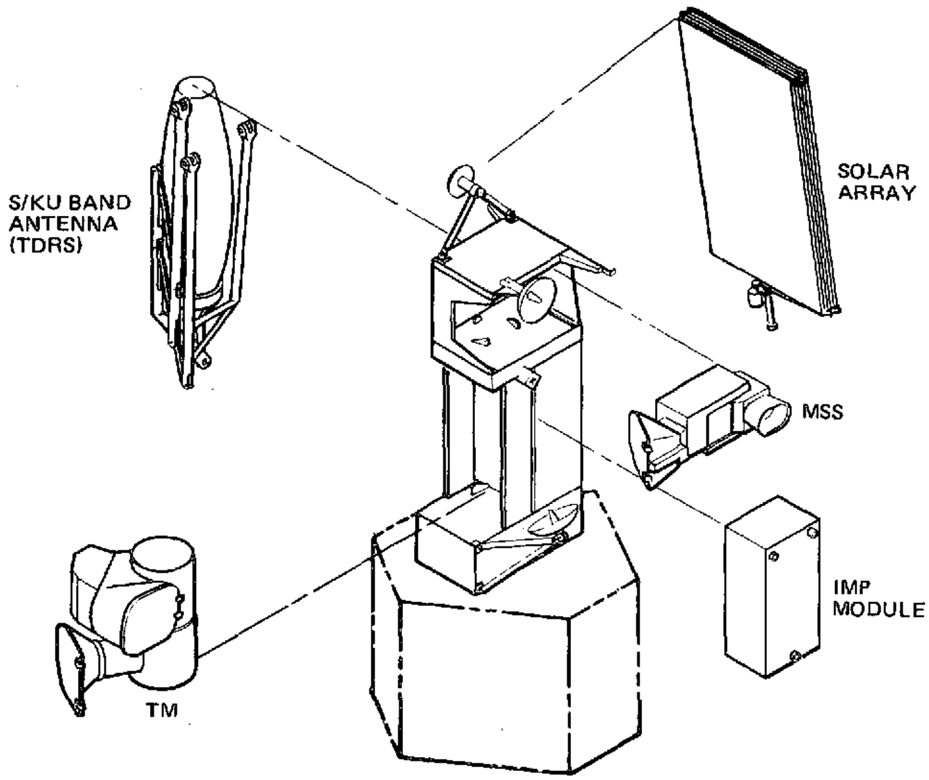

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Fig. 3-34 Instrument and Mission Peculiar Equipment

3.7.3.1.4.4 Solar Array Installation

3.7.3.1.4.4.1 Solar Array

(a) The solar array shall be designed to meet the requirements of Moving Mechanical Assemblies.

(b) The plane of the deployed array shall be perpendicular to the XZ plane of the Spacecraft and be capable of rotating about the Spacecraft Y axis.

3.7.3.1.4.4.2 Stowage Tiedown and Release Mechanism

(a) The loads induced in the tie-down mechanism shall not be transmitted to the solar array drive motor/torque shaft assembly.

(b) The tie-down mechanism shall be compatible with the selected solar panel substrate design.

(c) The mechanism shall apply the required preload to each side of each of the panel substrate interfaces to support the loads induced during the launch environment.

(d) The release mechanism shall perform in as short a time that is consistent with the strength capabilities of the solar panels.

3.7.3.1.4.4.3 Deployment and Lock Mechanism

Deployment time of the array shall be in as short a time possible, consistent with the strength capabilities of the structure, solar panels, and deployment mechanism.

3.7.3.1.4.4.4 Solar Array Motor Drive Assembly

(a) The motor assembly shall contain redundant bearing assemblies designed for the launch environments imposed on the motor/shaft assembly masses.

(b) The motor housing shall have provisions for vehicle mounting.

3.7.3.1.4.5 Spacecraft to Delta Launch Vehicle Adapter

This structure shall provide structural continuity between the existing adapter support ring fitting on the Delta L/V and the bottom of the S/C core structure as described in Fig. 3-35. Its structural characteristics are:

(a) A lower ring fitting which bolts to an existing Delta L/V fitting.

(b) A conic stiffened sheet metal load redistribution section approximately 24 inches long enclosing the OA/RCS module.

(c) An upper ring fitting attaching to the lower bulkhead of the S/C core structure at six discrete points. The Spacecraft shall be separable from the adapter by pyrotechnic devices at this interface.

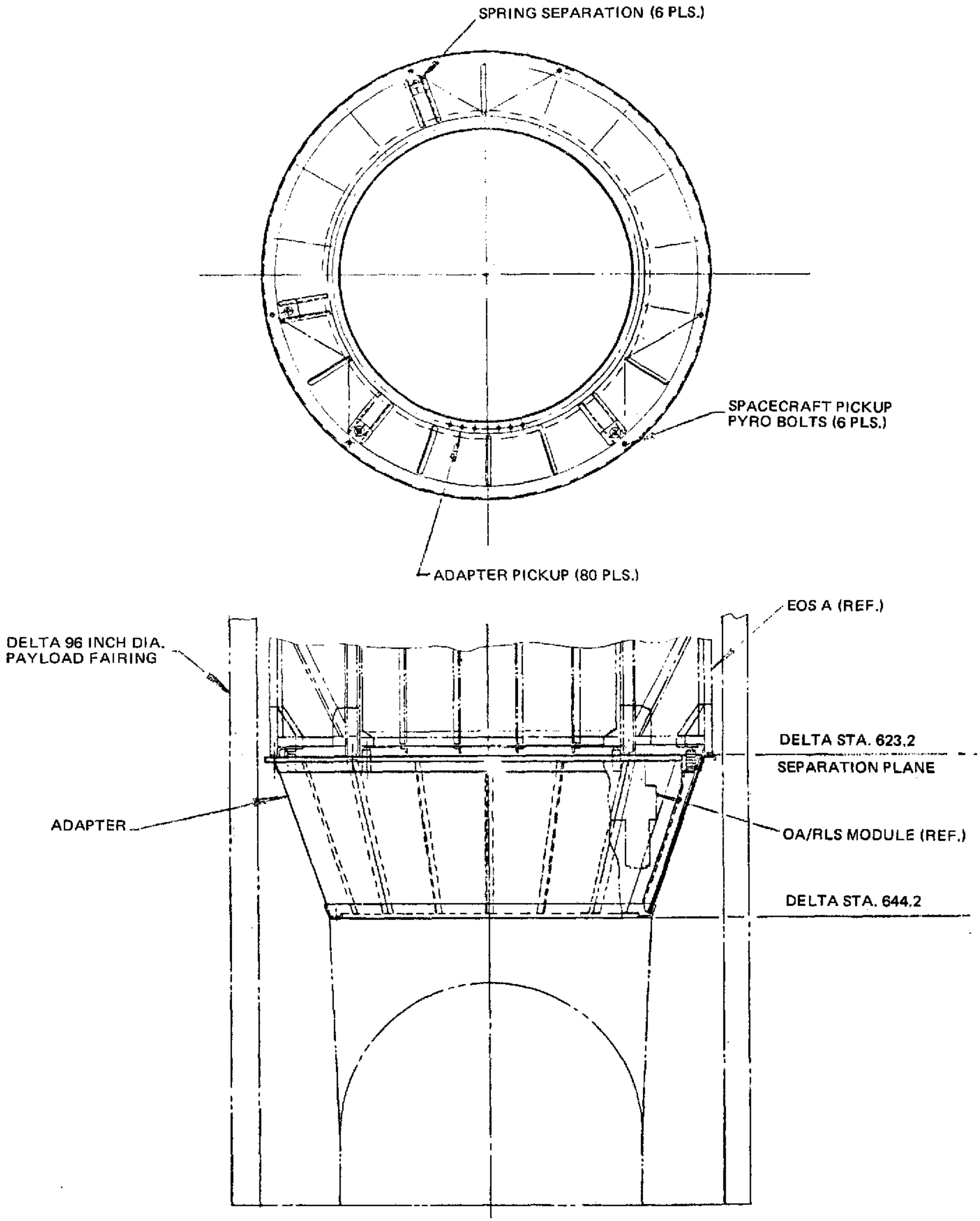

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Fig. 3-35 Spacecraft/Launch Vehicle-Adapter

3.7.3.1.5 Thermal Subsystem

The thermal subsystem shall continuously maintain temperatures of all mission peculiar items on the Observatory within specification limits while under all potential combinations of external environment and equipment power. The mission peculiar items are instruments, instrument structure, instrument mission peculiars and solar array.

3.7.3.1.5.1 General Requirements

3.7.3.1.5.1.1 Equipment Operating Temperatures

The minimum and maximum allowable operating temperatures for all mission peculiar equipment shall be determined from the reliability-life requirements of Paragraph 3.2.3.2.

3.7.3.1.5.1.2 Instrument Temperatures

The minimum/maximum allowable operating temperature, qualification temperatures, design goal operating temperatures and survival temperatures, shall be as specified in 314-ICD-002.

3.7.3.1.5.1.3 Instrument Structure Temperatures

The minimum/maximum allowable temperatures, qualification temperatures, design goal temperatures and survival temperatures shall be as specified in 314-ICD-002.

3.7.3.1.5.1.4 Instrument Mission Peculiar Temperatures

The minimum/maximum allowable operating temperatures, qualification temperatures, design goal operating temperatures, and survival temperatures shall be as specified in 314-ICD-001.

3.7.3.1.5.1.5 Solar Array Temperatures

The minimum/maximum allowable operating temperatures, qualification temperatures, design goal operating temperatures and survival temperatures shall be as specified in EOS-SS-240.

3.7.3.1.5.1.6 Design Requirements

In all cases design goal temperatures shall not exceed minimum and maximum allowable operating temperatures. The design goal temperatures shall be the minimum and maximum temperatures predicted for worst case cold and hot operation. The effects of time of year, operating duty cycles and surface property degradation shall be included in worst case predictions. Statistical variations, including external environment tolerances, surface properties, conductance and insulation effectiveness shall be included in the minimum and maximum predicted temperatures.

3.7.3.1.5.1.7 Control

The prime approach to achieve the thermal design for mission peculiar items shall be passive control. If it can be demonstrated that passive techniques cannot provide the required head rejection capability or result in excessive temperature gradients or excessive heater power requirements, then active control shall be used. In all cases, preference shall be given to qualified thermal control hardware and materials. Coating materials in critical areas shall have stability to provide the required radiant properties in the space environment to achieve the specified lifetime of the mission peculiar equipment.

3.7.3.1.5.2 Function

The thermal subsystem shall maintain all mission peculiar equipment and structure temperatures within the limits specified for all mission phases.

3.7.3.1.5.3 Configuration

The prime approach for achieving temperature control for mission peculiar items shall be passive. Active control shall be implemented if required as stipulated in Paragraph 3.7.3.1.5.1.7. The instruments and instrument mission peculiar equipment shall be designed to reject all heat dissipation to space. The thermal design shall minimize equipment heat sink and instrument structure temperature gradients. The equipment and structure shall be made thermally independent of each other, using insulation and low conductance mounts.

3.7.3.1.5.3.1 Passive Control

The following thermal control hardware shall be considered passive:

(a) Thermal control skins and surface finishes

(b) Multilayer thermal insulation blankets

(c) Conductive path materials control

(d) Heater circuits, temperature controlled and/or ground commandable.

3.7.3.1.5.3.2 Active Control

The following thermal control hardware shall be considered active:

(a) Louvers

(b) Heat pipes - all types

3.7.3.1.5.4 Modes

The prime mission modes to be considered for the thermal design of mission peculiar items and corresponding temperature requirements are as follows:

(a)	Prelaunch-	design goal temperatures
(b)	Launch and Boost -	design goal temperatures
(c)	Orbit-Operating (2 yrs) -	design goal temperatures
(d)	Orbit-Survival (3 yrs) -	survival temperatures
(e)	Shuttle-Retrieve -	survival temperatures

3.7.3.1.5.5 Performance Requirements

The thermal subsystem shall continuously maintain all mission peculiar items within specified temperatures for all modes specified. The analysis required to achieve the thermal design shall include:

(a) A detailed power load analysis, defining all equipment operating modes. This load analysis shall be updated periodically to incorporate measured value data and the specific requirements of each mission.

(b) An orbital heat flux analysis to determine the worst case heat fluxes for each critical surface.

(c) Equipment thermal analysis for each mission peculiar equipment unit to verify that the reliability requirements of Paragraph 3.7.1.5.1.1 are achieved.

(d) A detailed thermal nodal model for each instrument mission peculiar module.

(e) A detailed thermal analysis of the solar array.

(f) A detailed comprehensive thermal nodal model of the entire Observatory, including mission peculiar items and the Basic Spacecraft. In addition to the design function, this model shall be used to generate test and flight predictions and to correlate this data.

The requirements and responsibilities to achieve the above analysis shall be established in 314-ICD-001 for the instrument mission peculiars.

3.7.3.1.6 Orbit Adjust/Reaction Control Subsystem

The OA/RCS module shall provide the capability for the Observatory to be compatible with the Shuttle Orbiter for retrieval.

3.7.3.1.6.1 Shuttle Retrieval

The OA/RCS Module shall provide the following capability for Shuttle retrieval:

(a) Propellant tank pressure relief

(b) Fail-safe operation when operating in close proximity to the orbiter

(c) Propellant tank retention of fluids under crash loads.

3.7.3.1.7 Electrical Integration

3.7.3.1.7.1 Harness

The main structure harnesses shall be as specified in Paragraph 3.7.1.7.3. Since their function is to integrate all subsystems and instruments into one effective functioning system, the harnesses shall reflect wiring differences congruent to specific mission requirements.

The basic requirements, such as hardware, redundancy, and EMC shall not be mission peculiar. However, additional mission peculiar interface requirements shall be specified for Shuttle resupply umbilical provisions.

3.7.3.1.7.1.1 Shuttle Umbilical Provision

Minimum shuttle umbilical requirements shall be: supply 28 VDC heater power for critical components; hardline to C&DH module for monitor and control capability; caution and warning hardlines for capability of monitoring conditions potentially hazardous to the shuttle; and power arm/disarm command capability for idle Power Module.

3.7.3.1.7.2 Pyrotechnic/Actuator Control

The basic commandable circuitry for the pyrotechnic/actuator control function shall contain redundant isolated safe, arm, and fire functions. However, the number of circuits shall vary according to specific deployment, ignitor, and mechanism requirements. The design of the control unit shall be in accordance with SAMTECH Range Safety Manual 127-1 and EMC Specifications Paragraph 3.3.2.

Ground status of each circuit shall be required via the ground umbilical. Likewise, shuttle compatibility operations shall require monitor of the applicable circuits as a caution and warning against potential hazard to the Shuttle Orbiter. The C&DH module shall initiate the pulse and time delayed control signals required to activate the respective busses and outputs of the pyrotechnic/actuator control unit.

This unit shall be attached to structure and shall not be considered a candidate for Shuttle resupply. Spare circuits and external wiring shall be included in the unit, if feasible, for additional functions anticipated for future Shuttle resupply operation requirements.

3.7.3.1.7.3 Solar Array Drive

3.7.3.1.7.3.1 Function

The function of the solar array drive shall be to rotate the solar array at a rate which tracks the sun for optimum solar incidence and provide the interface for power output and signal transmission from solar array to Spacecraft.

3.7.3.1.7.3.2 Description

The solar array drive shall consist of a motor drive, control electronics, and slip ring assembly. The motor drive shall be a direct coupled, brushless, permanent magnet rotor, synchronous motor. The control electronics shall be designed to receive computer direction for drive operation of the motor drive. The slip ring assembly shall transmit signal and power across the rotary joint with line loss as low as possible.

3.7.3.1.7.3.3 Modes

As a minimum requirement, the solar array drive modes shall be: power off; power on; normal track; fast slew, and reverse rotation. Solar array power shall be transmitted via slip rings in all modes when available. During the orbit eclipse or dark periods, the drive shall continue to rotate at normal track speed. The on board computer shall periodically update the drive for position accuracy.

3.7.3.1.7.3.4 Performance Requirements

Performance requirements for the solar array drive are basically mission peculiar and shall be detailed by EOS-SS-260 Specification for the Electrical Integration Subsystem. Relative parameter/requirements are denoted in Table 3-15.

Table 3-15 Solar Array Drive Requirements

PARAMETER	REQUIREMENT
OPERATION	CONTINUOUS, BI-DIRECTIONAL
OPERATING VOLTAGE	28±7 VDC
TRACK RATE(1)	ORBIT DEPENDENT (3.8°/MIN NOMINAL
TRACK ACCURACY(1)	SPECIFIED IN EOS-SS-260(2)
FAST SLEW(1)	15°/MIN., NOMINAL
POSITION INDICATION	±1°
TORQUE(1)	2 TIMES TOTAL REFLECTED TORQUE AT OUTPUT SHAFT DUE TO FRICTION IN BEARINGS & SLIP RINGS MINIMUM
POWER TRANSMISSION(1)	50 A MAX; 125 VDC MAX
SIGNAL TRANSMISSION(1)	LIGHT/DARK SENSOR; TEMPERATURE & VOLTAGE FOR EACH SOLAR PANEL

NOTES

(1) REQUIREMENTS ARE MISSION PECULIAR – ORBIT AND/OR INSTRUMENT DEPENDENT.

(2) ACCURACY BASED ONLY ON TRACKING THE SUN WITHIN THE SPACECRAFT ORBIT PLANE & DOES NOT INCLUDE THE INCIDENT ANGLE VARIATIONS CAUSED BY OUT OR ORBIT PLANE MOVEMENT OF THE SUN.

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3.7.3.1.7.3.5 Physical Requirements

3.7.3.1.7.3.5.1 Configuration

The size and weight of the solar array drive shall be a minimum consistent with good design. The weight shall not exceed 26 pounds. The size/profile shall be as specified in EOS-SS-260 Specification for the Electrical Integration Subsystem.

3.7.3.1.7.3.5.2 Shuttle

The configurations for Shuttle deploy/retrieve requirements shall be packaged with a spacecraft interface connector assembly as specified in Paragraph 3.7.1.7.4.3. The size/profile and weight shall be as specified in EOS-SS-260.

3.7.3.1.7.3.5.3 Slip Rings

Slip rings for array power transmission shall be redundant and current carrying rated according to mission requirements. Requirements shall be as specified in EOS-SS-260.

3.7.3.1.7.3.6 Interface Requirements

The solar array drive interface requirements shall include:

- Electronics - Power Module-28 VDC and return; C&DH Module - command, telemetry (temperature, shaft position)

- Slip Rings - Power Module-Array power output, array signals (light/dark, voltage); C&DH Module - array temperature; Solar Array - array power output. array signals (light/dark, voltage temperature)

3.7.3.1.8 Instrument Data Handling and Wide Band Communications

3.7.3.1.8.1 Wide Band Data Handling and Compaction (WBDHC)

The WBDHC equipment shall provide the interface between the LRM mission peculiar instruments and the communication equipment.

3.7.3.1.8.1.1 Functions

The functions of the WBDHC equipment shall be:

(a) Format Thematic Mapper (TM) data for transmission over the wideband data link

(b) Reduce and format TM data (by data elimination) for transmission over the compacted data (low cost user) link

(c) Accept formatted Multi-Spectral Scanner data for transmission over the wideband link and, under on-board computer control, over the local user data link in place of the compacted TM data

(d) Accept formatted tape recorder inputs in lieu of realtime data for output to the communication equipments

(e) Accept on-board computer overhead data for inclusion into the TM and low cost user formatted data stream.

3.7.3.1.8.1.2 Configuration

The major components of the WBDHC equipment shall be:

(a) Wideband Combiner

(b) Data Compactor

(c) Compaction Buffer

A block diagram of the WBDHC equipment and the equipment interfaces are shown in Fig. 3-36. Included in the WBDHC are a timing source and power supply as well as the TM Data Handling Unit and Compacted Data Selection (see Fig. 3-37.

3.7.3.1.8.1.3 Modes of Operation

The WBDHC equipment shall have the following modes of operation, selectable by the on-board computer:

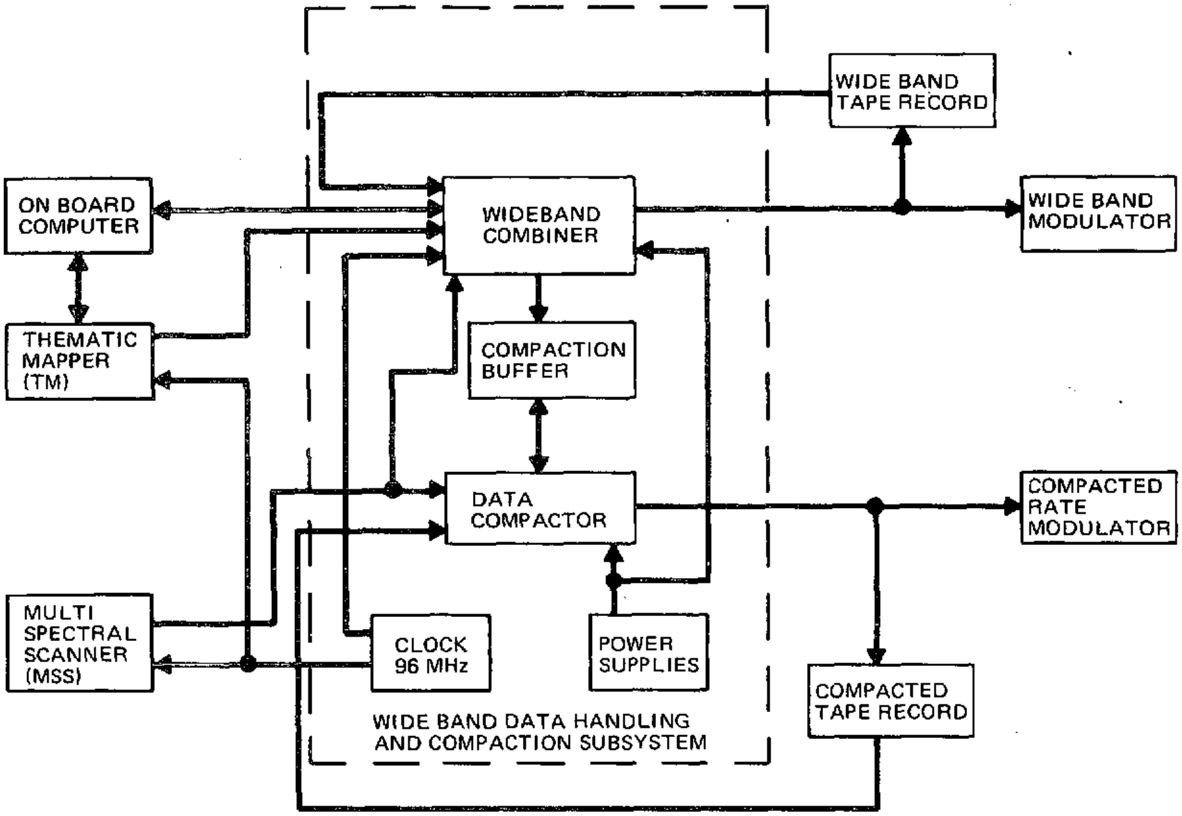

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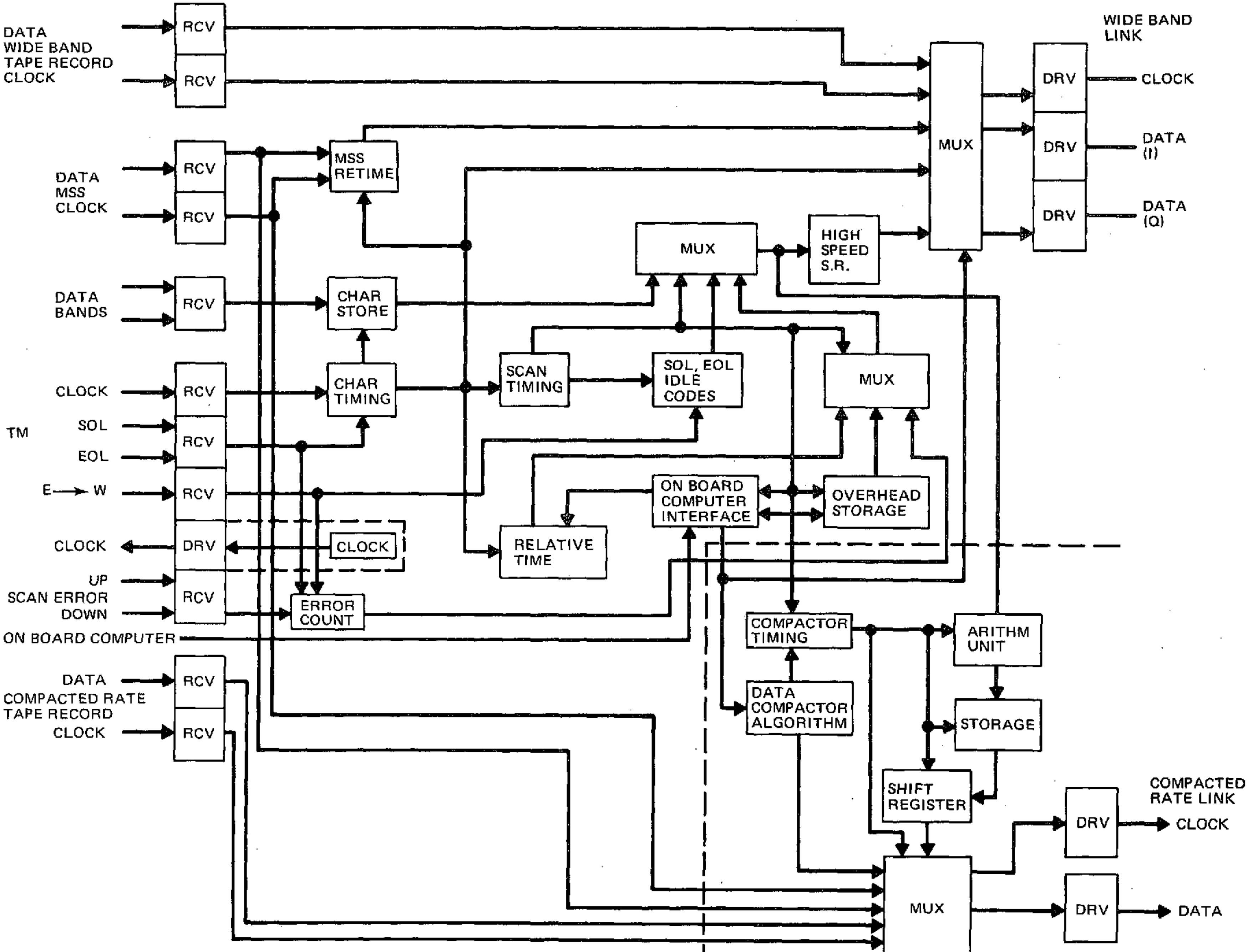
Fig. 3-36 C&DH Wide-Band Data Handling and Compaction Subsystem Interface



(1)5-48

Fig. 3-37 TM Data Handling Unit and Compaction Data Selection

(a) Wide band data output of either real time or play back (tape recorder) data for both TM and MSS.

(b) Data output of either compacted TM or MSS, real time or play back.

(c) Compacted TM data shall be of the following forms:

(1) All bands at 1/16 the resolution (2 resolution elements (RE) in direction of scan by 3 RE in direction of S/C motion).

(2) One high resolution band at full resolution plus low resolution (IR) band.

(3) All bands at full resolution for an 18 mile swath.

(4) Three bands at full resolution for 36 mile swath.

(5) Four high resolution bands at 1/4 resolution (2 RE in each direction) plus IR band.

3.7.3.1.8.1.4 Interface Requirements

The WBDHC equipment shall interface with the following equipment signal lines:

(a) TM

(1) Data, 7 lines: Each data line, with the exception of the seventh, contains a seven bit sample from eighteen detectors multiplexed together. The seventh line contains a seven bit sample only once every twelve sample intervals of the other six. No valid data exists in band 7 during the other 11/12ths of the time.

(2) Control, 5 lines: The five control signals are start of line (SOL), end of line (EOL), east to west scan indicator (E-W), and two scan error signals, positive increment (up) and negative increment (down). SOL indicates when the active portion of a scan is started. EOL indicates when the active portion of the scan is completed. E-W indicates the direction in which the instrument is scanning. The two scan error signals are incremental change signals that are required to remove scan non linearities during ground processing of the data. A zero error exists on the east to west scan at the SOL time. The error at any other time in a two scan cycle can be determined by accumulating the incremental error signals.

(3) Data Clock, 1 line: A clock signal is transmitted from the instrument to define the transition times of the other twelve signals. It may be used to strobe the data and control lines.

(4) Master Clock, 1 line: A master clock signal at 96 MHz is supplied by the WBDHC equipment to the TM from which all instrument timing is derived.

(b) MSS

(1) Data, 1 line: A single data line carrying the formatted MSS data.

(2) Data Clock, 1 line: A clock signal transmitted from the MSS at 16 MHz to define the transition times of the data.

(3) Master Clock, 1 line: A master clock at 16 MHz is supplied by the WBDHC equipment to the MSS from which all instrument timing is derived.

(c) On-Board Computer

(1) Input Data to WBDHC, 1 line: A single data line from Remote Decoder supplies data in 16 bit serial words.

(2) Input Data Gate, 1 line: This signal indicates when data is present on data line.

(3) Input Data Clock, 1 line: A 20 KHz clock to define data transitions.

(4) Output Data, 16 lines: Sixteen status outputs are supplied to the processor via the remote multiplexer. These outputs shall include power status, timing status, etc.

(d) Wideband Modulator

(1) Data, 2 lines: Two data signals are supplied by the WBDHC, one for the I channel (TM data), and one for the Q channel (MSS data).

(2) Clock, 1 line: The timing signal for clocking the data is supplied by the WBDHC at 120 MHz.

(e) Compacted Data Modulator

(1) Data, 1 line: Compacted data signal line is supplied by WBDHC.

(2) Clock, 1 line: A timing signal for compacted data is supplied by WBDHC at 20 MHz.

(f) Wideband Tape Recorder

(1) Output Data, 2 lines: Two data signals are supplied by the WBDHC - the preformatted TM data and the MSS data.

(2) Output Data Clock, 1 line: A timing signal is supplied by the WBDHC for clocking the data at 96 MHz.

(3) Input Data, 2 lines: Two data signals are supplied to the WBDHC-preformatted TM data and MSS data.

(4) Input Data Clock, 1 line: A timing signal is supplied to the WBDHC at 96 MHz for clocking the input data.

(g) Compacted Data Tape Recorder

(1) Output Data, 1 line: The compacted data output is supplied by the WBDHC.

(2) Output Data Clock, 1 line: A timing signal for compacted data is supplied by WBDHC.

(3) Input Data, 1 line: Stored compacted data is supplied to the WBDHC.

(4) Input Data Clock, 1 line: A timing signal for input data is supplied to the WBDHC.

(h) Prime Power System - Two independent connections shall be made to this 28 volt system to provide totally redundant power sources for the data handling equipment. Total power drawn from the prime power bus shall not exceed 120 W.

3.7.3.1.8.1.5 Performance Requirements

The following features shall be provided in the WBDHC equipment:

(a) Wideband Combiner

(1) Generate a data frame for each scan.

(2) Generate a start of line and end of line code in the frame at the receipt of SOL and EOL signals from TM.

(3) Combine up to 700 bits of computer supplied overhead, time at the start of a frame and frame error throughout the frame with the data.

(4) Generate an idle code against which the SOL code can be detected.

(5) Synchronize the MSS data to the TM data (the rates are 6 to 1).

(6) Supply data at 96 Mbps, from either real time or play back TM and at 16 Mbps (synchronized to 96 Mbps) real time or play back MSS.

(7) Output data rates shall be capable of being increased to 120 and 20 Mbps by only changing the master clock.

(b) Data Computer

(1) Generate a data frame for each scan.

(2) Include in the data frame all the parameters overhead appearing in the wideband frame.

(3) All the operating modes in Paragraph 3.7.3.1.8.1.3 shall be provided.

(4) Output rate shall be 16 Mbps.

(5) Output shall be capable of operation at 20 Mbps.

(c) Compaction Buffer

The compaction buffer shall be designed to handle the worst case mode of operation described in Paragraph 3.7.3.1.8.1.3 in both size and speed.

(d) WBDHC Equipment General

(1) Weight - 36 pounds max

(2) Power - 120 watts max

(3) MTTF - 4 years (calculated)

(4) Volume - 1.5 ft^3 max.

3.7.3.1.8.2 Primary Relay (TDRS) Wideband Communications Subsystem

The primary relay (TDRS) wideband communications subsystem shall satisfy the requirements for transmitting wideband experiment data to the ground via the TDRS. It shall be compatible with the NASA TDRS system as specified in GSFC document X-80574176.

3.7.3.1.8.2.1 Functions

The primary relay (TDRS) wideband communications subsystem shall:

(a) Provide simultaneous transmission of two wideband data channels to the ground via TDRS.

(b) Provide telemetry points for monitoring of critical functions.

(c) Provide command capability for controlling subsystem modes of operation.

3.7.3.1.8.2.2 Configuration

The major components of the primary relay (TDRSS) wideband communications subsystem shall be:

(a) QPSK Modulator/Exciter

(b) Ku-Band RF Power Amplifier

(c) Omni/Directional Antennas

The relay wideband communications subsystem shall be configured as shown in Fig. 3-38.

3.7.3.1.8.2.3 Modes of Operation

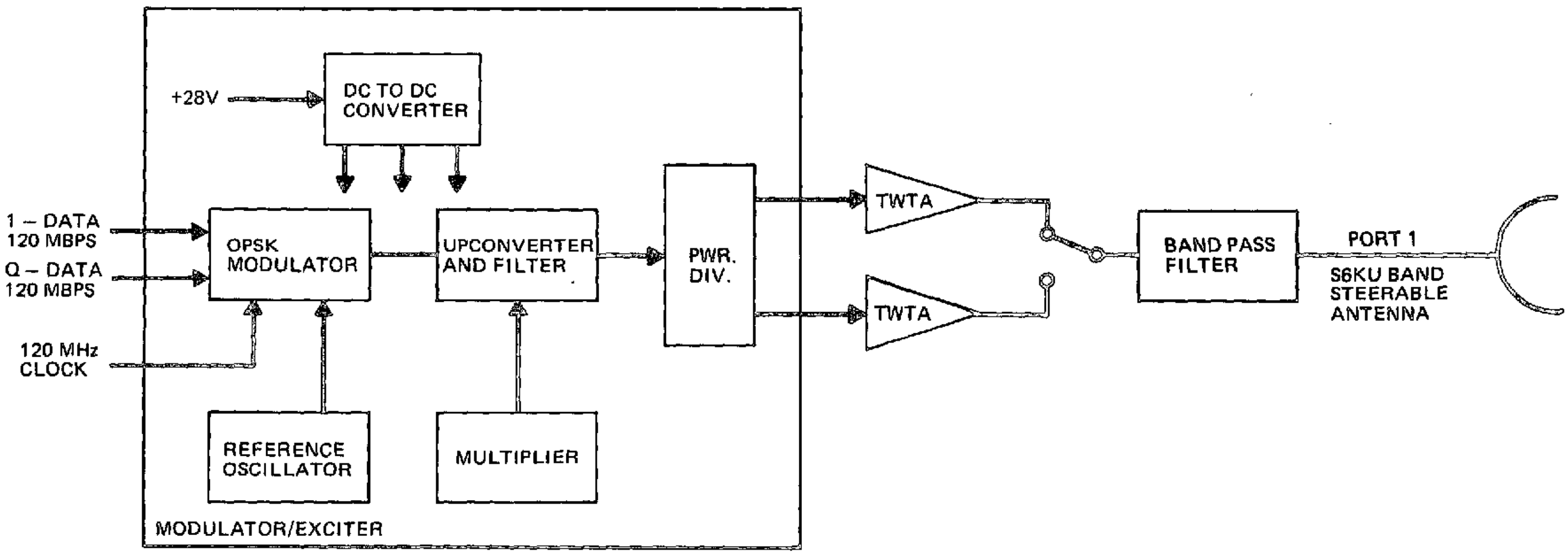


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Fig. 3-38 Block Diagram of the Primary Relay (TDRS) Wideband Communications (Ku Band)

The relay wideband communications subsystem modes of operation shall be selected by execution of ground or stored commands. The modes of operation shall be as follows:

(a) TDRS acquisition (unmodulated carrier)

(b) Data Transmission (modulated carrier)

3.7.3.1.8.2.4 Performance Requirements

3.7.3.1.8.2.4.1 Link Considerations

The primary relay (TDRSS) link shall transmit OPSK encoded data to the ground via TDRS with a 3 dB system margin above the signal level required for a 5×10^{-6} bit error rate under the following conditions:

(a) Frequency: 15.0085 MHz

(b) TDRS Antenna Gain or axis: 52.6 dB

(c) Polarization Loss: 0.5 dB

(d) Data Rate: 120 Mbps/channel
240Mbps/2 channels (quadriphase)

(e) Modulation: QPSK

(f) Data Coding: Differential, for ambiguity resolution only

(g) Maximum range (LDS): 42,000 Km

(h) E/N. required: 12.5 dB

(i) Transponder Loss: 20 dB

(j) Demodulation Loss: 1.5 dB

(k) TDRS Ts: 710^{o}K

(l) Pointing Loss: 0.5 dB

(m) Residual Carrier Loss: 1.0 dB

(n) EOS EIRP Required:
Acquisition: 8 dBW
Data Transmission: 61.3 dBW

3.7.3.1.8.2.4.2 Low Gain/High Gain Directional Antenna

The omni/directional antenna shall be an integrated assembly having the characteristics listed in Table 3-16 and illustrated in Fig. 3-39.

Table 3-16 Dual Feed - S/Ku-Band Steerable Antenna Design Requirements

PARAMETER	S-BAND	KU-BAND HIGH GAIN	KU-BAND LOW GAIN
1. FREQUENCY, GHZ TRANSMIT RECEIVE	 2.025 TO 2.120 2.200 TO 2.300	 14.6 TO 15.2 13.6 TO 14.0	 14.6 TO 15.2
2. ANTENNA TYPE	PARABOLIC DISH	PARABOLIC DISH	OPEN ENDED WAVEGUIDE
3. FEED TYPE	PRIME FOCAL POINT	CASSEGRAIN	N/A
4. POLARIZATION	RHCP	RHCP	RHCP
5. AXIAL RATIO, DB, MAX	1.5	1.5	1.5
6. INPUT VSWR AT ROTARY JOINT OUTPUT	1.4:1	1.5:1	1.5:1
7. SIDE AND BACK LOBE LEVELS, DB	≤ 17.0	≤ 17.0	N/A
8. ANTENNA DISH SIZE (FT)	12.5	12.5	N/A
	FREQ. (2.25 GHZ)	FREQ. (14.6 GHZ)	FREQ. (14.6 GHZ)
9. NET ANTENNA GAIN (DB) MEASURED AT THE ROTARY JOINT INPUT (INCLUDES ALL FEED ILLUMINATION AND TRANSMISSION LINE COMPONENT LOSSES)	35	51	0 (MINIMUM WITH-IN 60° HPEW)
10. TRACKING CONFIGURATION	OPEN	CLOSED (PSEUDO MONOPULSE)	–
11. TRACKING ACCURACY, 3σ	–	0.17 DEGREES	–
12. POINTING ACCURACY, 3σ	–	0.05 DEGREES	–
13. GIMBAL STEP SIZE	–	0.02 DEGREES	–
14. SLEW RATE,			
VELOCITY	–	20 DEG/SEC MAXIMUM	–
ACCELERATION	–	60 DEG/SEC^2 MAXIMUM	–
15. SCAN ANGLE OFF-BORESIGHT, 2 AXIS (XY GIMBAL)	X (INNER) GIMBAL ± Y (OUTER) GIMBAL ±	90 DEGREES 110 DEGREES	
16. TOTAL WEIGHT, LBS INCLUDING FEED/MICROWAVE SYSTEM, ROTARY JOINTS, REFLECTOR, FEED SUPPORT, GIMBAL ASSEMBLY, CONTROL ELECTRONICS AND DEPLOYMENT HARDWARE		TBD	
17. TOTAL POWER (WATTS)			
◦ DEPLOYMENT		TBD	
◦ PEAK OPERATING POWER (INCLUDES MOTOR DRIVE AND MOTOR CLAMPING PLUS CONTROL ELECTRONICS)		TBD TBD	
◦ AVERAGE POWER (INCLUDES MOTOR POWER SLEW, MOTOR POWER - TRACK AND CONTROL ELECTRONICS		TBD	
18. RELIABILITY -	DESIGN FOR A 2 YEAR OPERATIONAL LIFE.		

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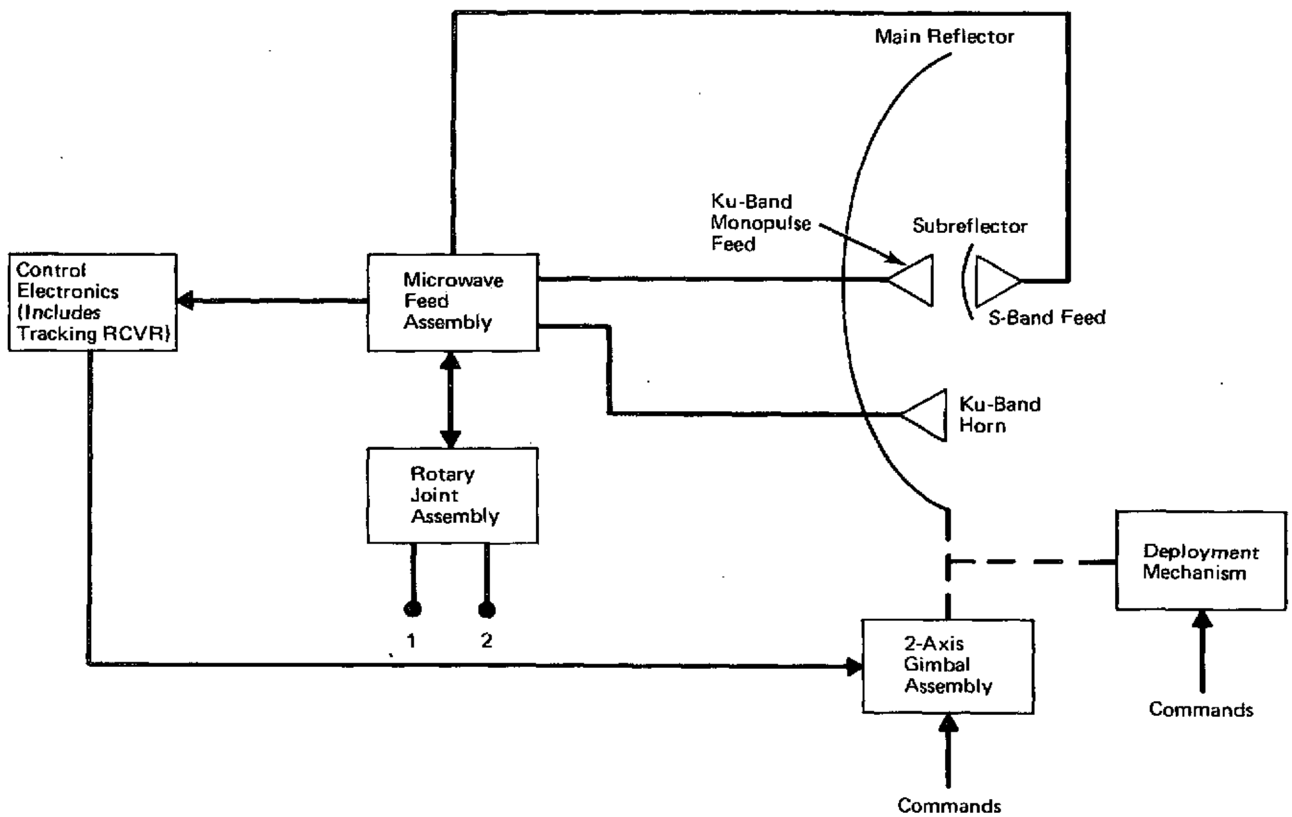

PORTS

1 Ku-Band Transmit Input

2 S-Band Receive/Transmit Output/Input

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Fig. 3-39 Dual Feed S/Ku-Band Steerable Antenna System – Block Diagram

3.7.3.1.8.2.4.3 Modulator/Exciter

(a) Frequency - The output frequency shall be fixed in the 14.896 to 15.121 GHz frequency range. The frequency shall remain within $\pm$ 0.0001 TBD percent of the assigned frequency and shall include tolerance, stability and environmental effects.

(b) RF Power Output - Minimum power output under worst-case specified environment and at 24 VDC input voltage shall be 5 milliwatts minimum. Rated power shall be provided with a load of 50 ohms at a maximum VSWR of 1.8:1 at any phase angle. No damage to the modulator/driver shall occur if the load is open or shorted.

(c) Modulation Type-QPSK modulation shall be employed. The modulator shall be capable of accepting two 120 Mbps data streams with the capability of reclocking the data pulses. Channel encoding snall be differential QPSK to resolve carrier phase ambiguities. Output filtering shall provide minimum overall transmission loss and detection loss at a BER of 10^{-6}.

3.7.3.1.8.2.4.4 TWT Amplifier

(a) Frequency Range - The operational frequency range of the TWTA shall be the same as the driver.

(b) Power Output - The TWTA output power shall not degrade below the minimum power necessary to achieve the specified EIRP under all orbital operations.

(c) Power Output Variation with Frequency - With a constant level swept input signal equal to that level required to produce saturation of the TWTA at mid-band, the maximum RF power output variation over the 14.8 to 15.2 GHz frequency range shall not exceed $\pm$ 0.2 dB.

(d) Gain - The saturated gain of the TWTA with a constant signal level input (which produces saturated power output) at the center of the frequency range (15.0085 GHz) shall not be less than 43 dB nor more than 45 dB.

3.7.3.1.8.2.4.5 Power Divider

A power divider shall be used at the output of the Modulator/Exciter and connected to the input of each TWTA.

(a) VSWR - 1.8:1 maximum referenced to 50 ohms at any port with other ports terminated.

(b) Insertion loss - 3.8 dB maximum at $f_t \pm$ MHz (includes 3 dB).

(c) Isolation - 15 dB minimum between two output ports.

3.7.3.1.8.2.4.6 RF Ferrite Switches

Two ferrite switches shall be latching type switches with a position indicator circuit. The switches are used to select a TWTA output and an antenna system.

(a) VSWR - 1.15:1 maximum referenced to 50 ohms.

(b) Insertion loss - 0.3 dB maximum at $f_t \pm 70$ MHz.

(c) Isolation - 25 dB minimum between two output ports.

(d) Switch time - TBD.

3.7.3.1.8.2.4.7 Bandpass Filter

A bandpass filter shall be used in the transmission of the steerable antenna.

(a) VSWR - 1.5:1 maximum

(b) Insertion loss - 1.0 dB maximum at center frequency.

(c) Bandwidth - The minimum 3 dB bandwidth shall be 300 MHz.

3.7.3.1.8.2.4.8 Telemetry Monitoring Points

The TDRS Communications Subsystem shall include diagnostic instrumentation for measuring a variety of functions, and converting the measurements to voltage and impedance levels suitable for interface with the Spacecraft telemetry system. The functions shall include but not be limited to the following:

(a) Modulator/exciter

(1) Output level

(2) Module temperature

(b) TWTA's

(1) Helix current

(2) Cathode current

(3) Converter reference volts

(4) Converter temperature

(5) Collector temperature

(c) Switch position

3.7.3.1.8.2.5 Physical

The TDRS subsystem major components shall be housed in a standard module as specified in Paragraph 3.7.4.6. Component physical requirements shall be specified in Specification EOS-SS-200. The weight allocation for the TDRS Subsystem which is part of the C&DH subsystem, is included in the overall C&DH subsystem weight as specified in Paragraph 3.2.2.1.1. The maximum weights and volumes for the TDRS subsystem equipments shall be as follows:

Equipment	Weight-lb	Volume-in^3
Modulator/Exciter	7.0	300
TWTA	10.0	350
Power Divider	0.3	15
Switch	0.3	15
Filter	0.3	25

3.7.3.1.8.3 Primary Direct Wideband Communications Subsystem

The primary direct wideband communications subsystem shall satisfy the requirements for transmitting wideband experiment data to the NASA ground stations (ETC, GDS, ULS) and the DOI ground station (Sioux Falls). It shall be compatible with the operational requirements defined in the NASA/GSFC Users Guide No. 101.1.

3.7.3.1.8.3.1 Functions

The primary direct wideband communications subsystem shall:

(a) Provide simultaneous transmission of two wideband data channels to the ground directly.

(b) Provide telemetry points for monitoring of critical functions.

(c) Provide command capability for controlling subsystem modes of operation.

3.7.3.1.8.3.2 Configuration

The major components of the primary direct wideband communications subsystem shall be:

(a) QPSK Modulator/Exciter

(b) X Band RF Power Amplifier

(c) Directional Antenna

The direct wideband communications subsystem shall be configured as shown in Fig. 3-40.

3.7.3.1.8.3.3 Modes of Operation

The direct wideband communications subsystem modes of operation shall be selected by execution of ground or stored commands.

(a) Antenna acquisition

(b) Data transmission

(c) Antenna selection

3.7.3.1.8.3.4 Performance Requirements

3.7.3.1.8.3.4.1 Link Considerations

The primary direct wideband communications link shall transmit QPSK encoded data to the ground with a 6 dB system margin above the signal level required for a 10^{-5} bit error rate under the following conditions:

(a) Frequency: TBD MHz in the 8.025 to 8.4 GHz range

(b) Ground G/T: 31 dB/^{o}K

(c) Minimum Elevation Angle to Ground Antenna: 2^{o}

(d) CCIR Power Flux Density Limits:

Elevation of User $< 5^{o}$, -154 dBW/4KHz/M^2

Elevation of User $> 5^{o}$, $< 25^{o}$, $-154 + \frac{\theta - 5^{o}}{2}$, θ = elevation angle

Elevation of User $> 25^{o}$, -144 dBW/KHz/M^2

(e) Atmosphere Loss (rain, cloud, 0_2): 7.1 dB

(f) Polarization: RHCP

(g) Maximum Slant Range:

(h) Pointing Loss: 0.5 dB

(i) Data Rate: 120 Mbps/Channel
240 Mbps/ 2 Channels (Quadriphase)

(j) Modulation: QPSK

(k) E/No Required: 13 dB @ Pe = 10^{-6}

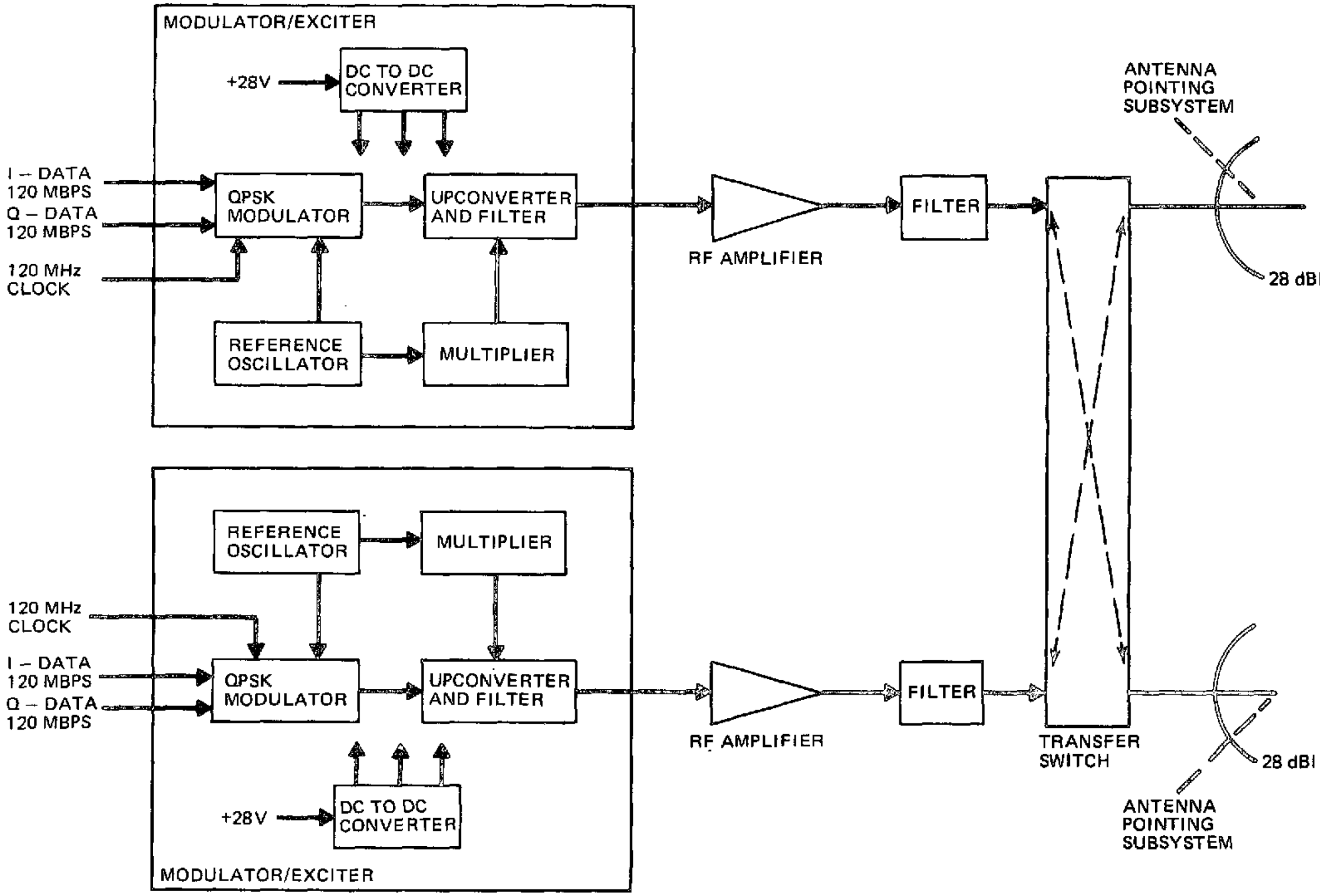

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Fig. 3-40 Block Diagram of the Primary Direct Wideband Communications Subsystem (X-Band)

(l) Demodulation Loss: N/A

(m) EOS EIRP Required: 30 dBW

3.7.3.1.8.3.4.2 High Gain Directional Antenna

The high gain directional antenna shall provide a capability of pointing towards any point on the earth disc visible from the Spacecraft, upon ground command, and to continue to redirect its direction (i.e., track) as the look angles change during the pass. The nominal gain shall be 28 dBi measured at the rotary joint (or other input port), but the exact gain will be determined by the EIRP requirement. The antenna will transmit with right hand circular polarization, and will have an axial ratio not to exceed 1 dB. The VSWR at the input port shall not exceed 1.5 to 1. The sidelobes shall be down 15 dB or more relative to the boresight level.

3.7.3.1.8.3.4.3 Modulator/Exciter

(a) Frequency - The output frequency shall be fixed in the 8.025 to 8.4 GHz frequency range. The frequency shall remain within $\pm$0.000 TBD percent of the assigned frequency and shall include tolerance stability and environmental effects.

(b) RF Power Output - Minimum power output under worst-case specified environment and at 24 VDC input voltage shall be 1 milliwatt minimum. Rated power shall be provided with a load of 50 ohms at a maximum VSWR of 1.8:1 at any phase angle. No damage to the modulator/driver shall occur if the load is open or shorted.

(c) Modulation Type - Inphase/Quadrature PSK modulation shall be employed. The modulator shall be capable of accepting two 120 Mbps data streams with the capability of re-clocking the data pulses. Channel encoding shall be differential to resolve carrier phase ambiguities. Output filtering shall provide minimum overall transmission loss and detection loss at a BER of 10^{-6}.

3.7.3.1.8.3.4.4 RF Power Amplifier (PA)

(a) Frequency Range - The operational frequency range of the RF PA shall be 8.025 to 8.4 GHz.

(b) Power Output - The output power shall not degrade below the level required to achieve the 30 dB EIRP under all orbital operations.

(c) Power Output Variation with Frequency - With a constant level swept input signal equal to that level required to produce rated power output at mid-band, the maximum power output variation over the 8.025 to 8.4 GHz frequency range shall not exceed $\pm$ 0.2 dB.

C-3

(d) Gain - The saturated gain of the RF PA with a constant signal level input at the center of the frequency range (8.2125 GHz) shall not be less than 43 dB nor more than 45 dB.

3.7.3.1.8.3.4.5 RF Transfer Switch

An RF transfer switch shall be a latching type switch with a position indicator circuit. The switch is used to connect either RF PA output to either directional antenna.

(a) VSWR - 1.15:1 maximum referenced to 50 ohms.

(b) Insertion Loss - 0.2 dB maximum at $f_t \pm 70$ MHz.

(c) Isolation - 40 dB minimum between two output ports.

(d) Switch time - 1.0 sec maximum.

3.7.3.1.8.3.4.6 Bandpass Filter

A bandpass filter shall be used in the transmission of each channel.

(a) VSWR - 1.5:1 maximum

(b) Insertion Loss - 1.0 dB maximum at center frequency.

(c) Bandwidth - The minimum 3 dB bandwidth shall be 300 MHz.

3.7.3.1.8.3.4.7 Telemetry Monitoring

The Primary Direct Wideband Communications Subsystem shall include diagnostic instrumentation for measuring a variety of functions, and converting the measurements to voltage and impedance levels suitable for interfacing with the Spacecraft telemetry system. The functions shall include but not be limited to the following:

(a) Modulators/Exciters

(1) Output level

(2) Module temperature

(b) RF Power Amplifiers

(1) Helix, Collector, and/or other DC power inputs

(2) Converter reference volts

(3) Converter temperature

(4) Tube or amplifier device temperature

(c) Switch Position

3.7.3.1.8.3.4.8 Physical

The wideband communication 240 Mbps subsystem major components shall be housed in a standard module as specified in Paragraph 3.7.4.6. Component physical requirements shall be specified in specification EOS-SS-200. The weight allocation for the wideband communications 240 MBPS subsystem, which is part of the C&DH subsystem, is included in the overall C&DH weight as specified in Paragraph 3.2.2.1.1.

The maximum weights and volumes for the wideband communication 240 Mbps subsystem equipments shall be as follows:

Equipment	Weight-lb	Volume-in^3
• Modulator/Exciter	7.0	330
• RF PA	8.5	325
• Filter	0.3	25
• Transfer Switch	0.8	50

3.7.3.1.8.4 Local User Wideband Communications Subsystem

The local user wideband communications subsystem shall satisfy the requirements for transmitting wideband experiment data to selected ground stations. It shall be compatible with the operational requirements defined in TBD document.

3.7.3.1.8.4.1 Functions

The local user wideband communications subsystem shall:

(a) Provide transmission of one wideband data channel to the ground.

(b) Provide telemetry points for monitoring of critical functions.

(c) Provide command capability for controlling subsystems modes of operation.

3.7.3.1.8.4.2 Configuration

The major components of the local user wideband communications subsystem shall be:

(a) Modulator/Exciter

(b) X-Band RF Power Amplifier

(c) Low gain antenna

The local user wideband communications subsystem shall be configured as shown in Fig. 3-41.

3.7.3.1.8.4.3 Modes of Operation

The local user wideband communications subsystem modes of operation shall be selected by execution of ground or stored commands.

3.7.3.1.8.4.4 Performance Requirements

3.7.3.1.8.4.4.1 Link Considerations

The local user wideband communications link shall transmit DPSK encoded data to the ground with a 3 dB system margin above the signal level required for a 10^{-5} bit error rate under the following conditions:

(a) Frequency: TBD MHz in the 8.025 to 8.4 GHz range

(b) Ground G/T: 11 dB/^{o}K

(c) Minimum Elevation Angle of Ground Antenna: 30^{o}

(d) CCIR Power Flux Density Limits:

Elevation of User $<5^{o}$, -154 dBW/4KHZ/M^2

Elevation of User $>5^{o}$, $<25^{o}$, $-154 + \frac{\theta - 5^{o}}{2}$, θ = elevation angle

Elevation of User 25^{o}, -144 dBW/4KHz/M^2

(e) Atmosphere Loss:

(f) Polarization: RHCP

(g) Maximum Slant Range:

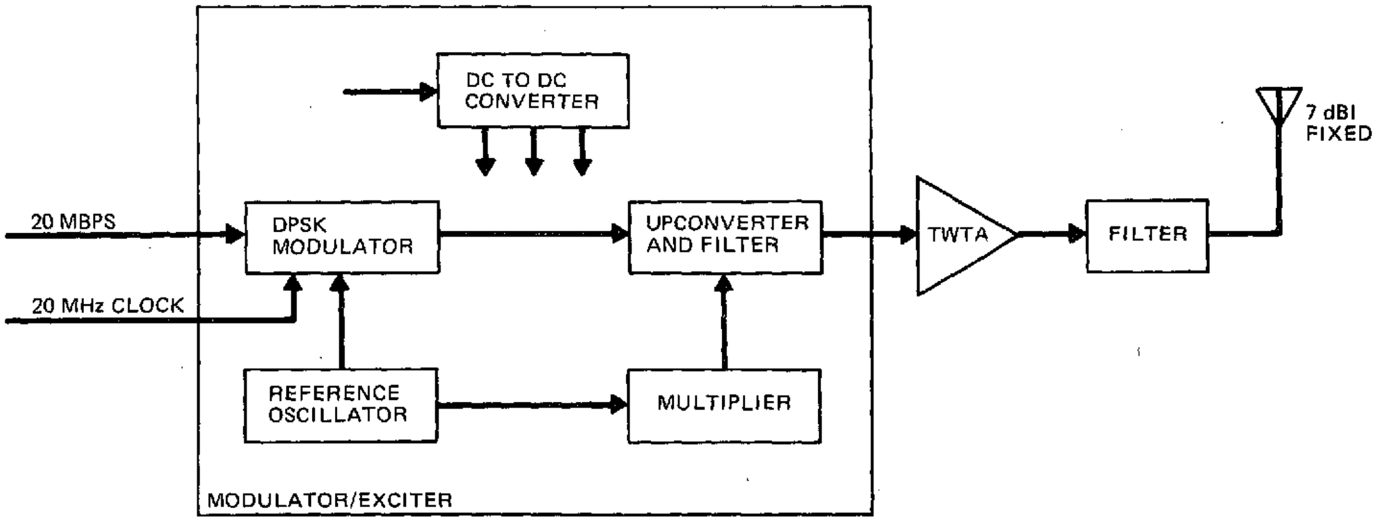
(h) Data Rate: 20 Mbps

(i) Modulation: DPSK

(j) E/N^{o} Required: 12 dB for Pe = 10^{-5}

(k) Demodulation Loss: N/A

(l) EOS EIRP Required: 22 dBW



(1) 5-51 3.7 Fig. 3-41 Block Diagram of the Local User Wideband Communications Subsystem (X-Band)

3.7.3.1.8.4.4.2 Low Gain Fixed Antenna

The low gain antenna shall be installed in a location and possess a pattern so as to illuminate local users within 500 Km of nadir. The antenna pattern shall be shaped so as to yield as constant as possible a signal (flux density) within the specified ground coverage area. Such shaping should not be carried to such an extent that users close to nadir actually receive a weaker signal than those further out, nor should it achieve uniformity at the expense of minimum gain level. The reference antenna gain, for planning purposes, is 7 dBi. The antenna pattern shall exhibit an axial ratio of 3dB within its defined coverage area. The VSWR at the input port shall not exceed 1.5 to 1.

3.7.3.1.8.4.4.3 Modulator/Exciter

(a) Frequency - The output frequency shall be fixed in the 8.025 to 8.4 GHz frequency range. The frequency shall remain within ±0.000 TBD percent of the assigned frequency and shall include tolerance, stability and environmental effects.

(b) RF Power Output - Minimum power output under worst-case specified environment and at 24 VDC input voltage shall be 30 milliwatts minimum. Rated power shall be provided with a load of 50 ohms at a maximum VSWR of 1.8: 1 at any phase angle. No damage to the modulator/driver shall occur if the load is open or shorted.

(c) Modulation Type - Differential PSK modulation shall be employed.

3.7.3.1.8.4.4.4 TWT Amplifier

(a) Frequency Range - The operational frequency range of the TWTA shall be 8.025 to 8.4 GHz.

(b) Power Output - The TWTA output power shall not degrade below the minimum required to meet the specified EIRP under all orbital operations.

(c) Power Output Variation with Frequency - With a constant level swept input signal equal to that level required to produce saturation of the TWTA at mid-band, the maximum RF power output variation over the 8.025 to 8.4 GHz frequency range shall not exceed ±0.2 dB.

(d) Gain - The saturated gain of the TWTA with a constant signal level input (which produces saturated power output) at the center of the frequency range (8.2125 GHz) shall not be less than 33 dB nor more than 35 dB.

3.7.3.1.8.4.4.5 Bandpass Filter

A bandpass filter shall be used in the output transmission line of the TWTA.

(a) VSWR - 1.5: 1 maximum

(b) Insertion Loss: 1.0 dB maximum at center frequency

(c) Bandwidth - The minimum 3 dB bandwidth shall be 80 MHz

(d) Passband - TBD

3.7.3.1.8.4.4.6 Telemetry Monitoring Points

The Local User Wideband Communication Subsystem shall include diagnostic instrumentation for measuring a variety of functions, and converting the measurements to voltage and impedance levels suitable for interfacing with the spacecraft telemetry system. The functions shall include but not be limited to the following:

(a) Modulator/Exciter

(1) Output level

(2) Module temperature

(b) TWTA

(1) Helix current

(2) Cathode current

(3) Converter reference volts

(4) Converter temperature

(5) Collector temperature

3.7.3.1.8.4.5 Physical

The wideband communication 20 Mbps subsystem major components shall be housed in a standard module as specified in Paragraph 3.7.4.6. Component physical requirements shall be specified in specification EOS-SS-200. The weight allocation for the C&DH subsystem weight is specified in Paragraph 3.2.2.1.1 which includes the wideband communication 20 Mbps subsystem. The maximum weights and volumes for the wideband communication 20 Mbps subsystem equipments shall be as follows:

	Equipment	Weight-lb	Volume-in^3
•	Modulator/Exciter	6.5	280
•	TWTA	10.0	350
•	Filter	0.3	25

3.7.3.1.9 Mission Peculiar Software

Mission Peculiar Software consists of those software modules which are prepared for use with the instruments peculiar to specific missions.

3.7.3.1.9.1 Experiment Software Module

The Experiment Software Module shall provide accumulation, storage and downlink of Spacecraft data required for engineering or space physics experiments using the Spacecraft and its sensors as a data source.

3.7.3.1.9.2 Experiment Control & Maintenance Software Module

The Experiment Control and Maintenance Software Module shall provide continuous monitoring and control of each mission peculiar sensor assigned to the Spacecraft, fulfilling its test, calibration and operating requirements.

3.7.3.1.9.3 Antenna Steering Software Module

The Antenna Steering Software Module shall provide steering signals to drive the gimbals of the downlink data antennas, based on current and predicted Spacecraft position and on the tabulated position of the receiving station.

3.7.3.1.9.4 Experiment Data Software Module

The Experiment Data Software Module shall monitor the flow of experiment data by sampling, to verify the operation of the experiment and to permit recognition of gross satellite characteristics.

3.7.3.2 FOLLOW-ON MISSION DRIVER REQUIREMENTS

3.7.3.2.1 Communication & Data Handling

(a) The C&DH module must be able to acquire and route housekeeping and scientific data up to a data rate of 32 Kbps. This requirement is established by the SEASAT mission.

(b) The C&DH module must be capable of including a tape recorder which will record at the data rate given of 32 Kbps and dump two orbits' worth of data in one ground station contact of 10 minutes maximum duration. This requirement is established by the SEASAT mission.

(c) The C&DH module must be capable of interfacing with up to 32 remote units (multiplexes and decoders) each of which can have 64 input and 64 output channels. This requirement is established by the TBD mission.

(d) The memory of the on-board computer must be expandable to 65 K words. This requirement is established by the TBD mission.

3.7.3.2.2 Electrical Power

(a) The Electrical Power System shall be capable of being expanded to provide at least 600 W of orbital average power to a non-sun synchronous retrograde low earth orbit payload. This requirement is established by the SEASAT mission.

(b) The Electrical Power System shall be capable of the addition of enough capacity to handle a 2 KW peak load with a 25% duty cycle. This requirement is imposed by the SEASAT mission.

(c) The Electrical Power System shall be capable of the addition of a two axis gimballed array drive. This requirement is imposed by the SEASAT mission.

3.7.3.2.3 Attitude Control

(a) The ACS shall be capable of pointing to array spot on the earth on demand to within 5.04 sec (0.0014^o, 24.4 rad, or about 1 n mi). This requirement is imposed by the SEOS mission.

(b) The ACS shall be capable of holding a spot on tne earth with a stability of $.46 \times 10^{-6}$ degrees/sec. (0.0017 sec/sec) for periods of 20 minutes with at most one update during this period. This requirement is imposed by the SEOS mission.

(c) The ACS shall be capable of pointing to any spot on the solar disc or corona on demand to within 5.76 sec (0.0016^o, 27.9 rad, or about 5000 Km). This requirement is imposed by the Solar Maximum Mission.

(d) The ACS shall be capable of holding a spot on tne solar disc or corona to within 0.93×10^{-6} degrees/sec (0.0033 sec/sec) for periods of at least 3 hours using error signals from payload mounted solar sensor for update. This requirement is imposed by the Solar Maximum Mission.

(e) The ACS shall be capable of slewing the spacecraft and payload to any spot up to 16 sec, (solar radius) within 8 sec. This requirement is imposed by the Solar Maximum Mission.

3.7.3.2.4 Structure

(a) The structure of the basic spacecraft shall be capable of supporting at least 2500 pounds of payload throughout the launch and most severe boost environment of the conventional launch vehicles considered for launch of EOS follow-on missions (e.g., Delta 2910, Delta 3910, Titan IIIB/SSB/NUS, Titan III-C7) with a minimum of modification. This requirement is imposed by the LRM-C mission.

(b) The structure must be capable of supporting at least 2500 pounds of payload throughout the launch and boost and normal landing environment of the Shuttle without damage to the payload, with a minimum of modification. This requirement is imposed by the LRM-C mission.

(c) The structure must be capable of supporting at least 2500 pounds throughout the crash landing environment of the Shuttle Orbiter without danger to the crew. This requirement is imposed by the LRM-C mission.

3.7.3.2.5 Thermal

(a) The Basic Spacecraft shall have the capability to reject at least 150 W in an operating temperature range 70°F ∓ 20°F using only additional passive cooling devices such as variable conductance heat pipes and optical solar detectors. This requirement is generated by the SEOS, SEASAT and SMM missions.

(b) The Basic Spacecraft shall have the capability to supply TBD watts of additional heater power, and the structural arrangement shall be flexible enough to allow for incorporation of these additional heaters. This requirement is imposed by the LRM-C mission.

3.7.3.2.6 RCS/Orbit Adjust/Orbit Transfer Module

(a) The OA/RCS shall provide for the capability of adding additional propellant storage capacity to perform 100% of the wheel unloading. This requirement is imposed by the SEOS mission.

(b) The orbit adjust thrusters shall have the capability of providing thrust rector control during kick motor burns. This requirement is imposed by the TIROS-N and LRM-C missions.

(c) The OA/RCS shall provide for the capability of adding additional propellant storage capacity for the thrust rector control propellant. This requirement is imposed by the TIROS-N and LRM-C missions.

(d) The orbit transfer module shall provide for the addition of kick motors and the associated support structure. This kick stage will be used for:

- Circularization of mission orbit
- Transfer from Shuttle Orbiter parking orbit to operational mission orbit and return to Shuttle Orbiter orbit.

These requirements are imposed by the TIROS-N mission.

3.7.3.2.7 Instrument Data Handling & Wideband Communications

(a) The instrument data handling and wideband communications shall be capable of routing and transmitting to the ground directly or TDRS with TBD BER using a TBD watt transmitter up to 430 Mbps. This requirement is imposed by the LRM-C mission.

(b) The instrument data handling and wideband communications system shall have the capability of interfacing with a single or several tape recorders capable of recording at the rate given and a capability of storing two orbits worth of data and dumping this data in one ground contact of no more than 10 minutes duration.

3.7.4 SUPPORT EQUIPMENT FUNCTIONAL CHARACTERISTICS

The contractor shall consider Table 3-17 as an initial identification of support equipment and its utilization from factory manufacturing through Observatory launch. Support equipment shall not be limited to this list, which may be modified by the contractor to conform to his recommended approach.

3.7.4.1 ELECTRICAL EQUIPMENT

3.7.4.1.1 Test and Integration Station (T&I)

The functional test and evaluation of the EOS shall be performed with real time command, control and monitoring provided by a T&I Station. The station shall consist of: a general purpose mini-computer with a TBD K core memory, a TBD bit word and a TBD micro sec cycle time; S-Band and Ku-Band RF front end for receipt and transmittal of downlink housekeeping data and uplink commands; a command and control display console; peripheral equipment consisting of disc storage, magnetic tape storage, hard copy printer and a paper tape reader/punch.

Also included shall be the capability to check an instrument test signal through a X-Band Receiver, Decom & Bit Sync and word comparitor.

Software for the T&I shall consist of two main categories, the real time operating system and the EOS test routines:

Table 3-17 Ground Support Equipment Utilization

DEVELOPMENT & QUAL

Item	MODULE FAB & ASSEMBLY	MODULE MASS REP. & INSTALLATION	MODULE STRUCT. DEV & ACOUSTIC QUAL	MODULE THERMAL VACUUM DEV. TEST	S/C STRUCTURE FAB & ASSEMBLY	MODULE & INSTR MASS REP. INSTALL.	STRUCTURE LOADS ACCEL TEST	ACOUSTIC TEST (LRM CONFIG)	MECHANICAL VIBRATION & SHOCK	MODEL SURVEY	RECONFIG INSTR MASS REPS	MODEL SURVEY	MECHANICAL VIBRATION & SHOCK	ACOUSTIC TEST	SEPARATION & DEPLOYMENT QUAL TEST	MODULE WIRING / COMPONENT INTEGR	INST INTEGR & OBS FUNCT TEST	ACOUSTIC VIBRATION / PYRO SHOCK	THERMAL VACUUM, INCLUDING EMC	POST T/V SEP TESTS
ELECTRICAL																				
1. TEST & INTEGRATION STATION																o	o	o	o	o
2. BREAKOUT BOX SET																o	o	o	o	o
3. BATTERY CONDITIONER																				
4. TEST BATTERY SET																	o	o	o	o
5. S/C POWER SET & CABLES																	o	o	o	o
6. RANGING TEST ASSEMBLY																o				
7. PYRO TEST SET																		o		
8. INTERFACE CABLE SET																	o			
9. SOLAR SIMULATOR																	o			
10. INSTRUMENT INTERFACE SIMULATOR																	o			
11. UMBILICAL SIMULATOR																	o			
12. DITMICO PROGRAM & CABLE SET																o				
13. POWER MODULE C/O BENCH																o				
14. C & DH MODULE C/O BENCH																o				
15. ACS MODULE C/O BENCH																o				
16. PROPULSION C/O BENCH (RCS)																o				
17. DCS SIMULATOR																	o	o	o	o
18. S/C MONITOR & CONTROL																	o	o	o	o
19. IMP MODULE C/O BENCH																o	o			
MECHANICAL																				
1. INTERFACE ADAPTER SET							o	o	o	o	o	o	o	o	o			o	o	o
2. HOIST BAR & SLING SET	●	●			o	o	o	o	o	o	o	o	o	o	o		o			
3. SUPPORT DOLLY VERTICAL S/C					o	o				o		o			o		o			o
4. SUPPORT DOLLY, HORIZONTAL							o	o	o				o	o				o	o	
5. ACCESS WORK STAND - S/C					o	o											o			o
6. SKIN STORAGE RACK																				
7. WT & CG FIXTURE - S/C						o														
8. MASS SIMULATOR SET		●				o														
9. SUPPORT DOLLY - S/C MODULES			o	o												o				
10. SHIPPING CONTAINERS S/C MODULES																				
11. S/C COVER SET																				
12. HUMIDITY CONTROL KIT																				
13. SHIPPING CONTAINER - SOLAR ARRAY																				
14. SOLAR ARRAY INST & DEPLOY. FIX															o					
15. TRANSPORTERS																				
16. INDICATING ACCELEROMETER KIT																				
17. PYRO INST. TOOL KIT																	o			
18. STAGE MOTOR INST FIXTURE																				
19. EOS/SHUTTLE SIMULATOR ADAPTER															o					
20. TIE DOWN KIT SET																				
21. BATTERY SHIPPING CONTAINER																				
22. BATTERY INSTALLATION TOOL																				
FLUID																				
1. GN_2 CONDITIONING UNIT																	o	o	o	o
2. GN_2 REGULATION UNIT																	o	o	o	o
3. VOLUMETRIC LEAK DETECTOR																	o	o		o
4. RCS VACUUM TEST CART																	o	o		
5. FLUID DIST SYS, GRUMMAN																	o	o		
6. PRESSURE MAINTENANCE UNIT																				
7. GN_2 STORAGE SYS (TRANSPORTER)																				
8. GN_2 MANIFOLD & SUPPLY PLATFORM																				
9. FLUID DIST SYS, LAUNCH SITE																				
10. PROPELLANT TRANS ASSEMBLY																o			o	
11. MASS SPECTROMETER LEAK DETECTOR																	o			

FLIGHT S/C EGR

Item	MODULE FAB ASSY & WIRING	COMPONENT INSTALLATION & INTEGR	S/S FUNCTIONAL C/O	MODULE ENVIRONMENTAL ACCEPT TEST	POST ENVIRONMENTAL TEST MODULE C/O	S/C STRUCTURAL FAB & ASSY	S/C WIRING & C/O	S/C MODULE INTEGR & FUNCT C/O	INSTR INTEG & C/O, IMP & OAS INTEGR	OBS SYS TEST & ENC	SEPARATION & DEPLOYMENT MECH C/O	OBS SPECIAL TEST	WORKMANSHIP ACOUSTIC, PACK & SHIP	PRELAUNCH OPERATIONS	LAUNCH
ELECTRICAL															
1. TEST & INTEGRATION STATION		o	o					o	o	o	o	o	o	o	o
2. BREAKOUT BOX SET		o	o	o				o	o	o		●		o	
3. BATTERY CONDITIONER										o				o	
4. TEST BATTERY SET								o	o	o	o			o	
5. S/C POWER SET & CABLES								o	o	o	o	o	o	o	
6. RANGING TEST ASSEMBLY		o													
7. PYRO TEST SET										o					
8. INTERFACE CABLE SET								o	o						
9. SOLAR SIMULATOR									o			o			
10. INSTRUMENT INTERFACE SIMULATOR								o	o						
11. UMBILICAL SIMULATOR										o				o	
12. DITMICO PROGRAM & CABLE SET							o								
13. POWER MODULE C/O BENCH		o	o	o	o										
14. C & DH MODULE C/O BENCH		o	o	o	o										
15. ACS MODULE C/O BENCH		o	o	o	o										
16. PROPULSION C/O BENCH (RCS)		o	o	o	o										
17. DCS SIMULATOR								o	o	o				o	
18. S/C MONITOR & CONTROL								o	o	o	o	o	o	o	
19. IMP MODULE C/O BENCH		o	o	o	o				o						
MECHANICAL															
1. INTERFACE ADAPTER SET										o	o		o	o	o
2. HOIST BAR & SLING SET				o				o		o	o		o	o	o
3. SUPPORT DOLLY VERTICAL S/C						o	o	o	o	o	o	o		o	
4. SUPPORT DOLLY, HORIZONTAL										o			o		
5. ACCESS WORK STAND - S/C						o	o	o	o	o		o			
6. SKIN STORAGE RACK						o	o	o	o						
7. WT & CG FIXTURE - S/C										o					
8. MASS SIMULATOR SET															
9. SUPPORT DOLLY - S/C MODULES	o	o	o	o	o			o							
10. SHIPPING CONTAINERS S/C MODULES													o		
11. S/C COVER SET													o		
12. HUMIDITY CONTROL KIT													o		
13. SHIPPING CONTAINER - SOLAR ARRAY													o		
14. SOLAR ARRAY INST & DEPLOY. FIX											o				
15. TRANSPORTERS													o		
16. INDICATING ACCELEROMETER KIT													o		
17. PYRO INST. TOOL KIT									o						
18. STAGE MOTOR INST FIXTURE								o							
19. EOS/SHUTTLE SIMULATOR ADAPTER											o				
20. TIE DOWN KIT SET													o		
21. BATTERY SHIPPING CONTAINER		o													
22. BATTERY INSTALLATION TOOL		o													
FLUID															
1. GN_2 CONDITIONING UNIT								o	o	o		o		o	o
2. GN_2 REGULATION UNIT								o	o	o		o		o	o
3. VOLUMETRIC LEAK DETECTOR								o	o	o					
4. RCS VACUUM TEST CART								o	o	o					
5. FLUID DIST SYS, GRUMMAN								o	o	o					
6. PRESSURE MAINTENANCE UNIT													o		
7. GN_2 STORAGE SYS (TRANSPORTER)													o		
8. GN_2 MANIFOLD & SUPPLY PLATFORM													o		
9. FLUID DIST SYS, LAUNCH SITE														o	o
10. PROPELLANT TRANS ASSEMBLY															o
11. MASS SPECTROMETER LEAK DETECTOR								o	o	o					

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- Real Time Operating System - This is a multitask real time operating system which shall provide memory file management, concurrent foreground/background processing capability, program overlays and I/O operations. Operations shall be scheduled and run by a real time executive in response to a real time clock, operator requests, external interrupts or by other tasks being executed. Also included in the Real Time Operating System library shall be the PCM data handling routine, data processing routines, display routines, printout routines, utility routines, and station diagnostics. The contractor shall determine and develop other programs that may be required.

- Observatory Test Routines - The contractor shall determine and develop test programs required exercising the Observatory by uplink command and comparing the downlink housekeeping to predetermined responses and evaluating, in real time, the proper operation of the Observatory.

The station shall be designed for easy transportability for use at the launch site as the prelaunch and launch checkout station.

3.7.4.1.2 Breakout Box Set

The breakout box set shall permit entry to interface points between S/S modules for the purpose of monitoring these points while permitting S/S operation. They may also be used during S/S module integration and build up. The boxes shall be universal and shall have common connectors.

3.7.4.1.3 Battery Conditioner

This equipment shall be used to condition the flight battery in a charge and discharge cycle.

3.7.4.1.4 Test Battery Set

These are a set of batteries which shall be used in place of the flight batteries for Observatory power during test in order to conserve the flight batteries.

3.7.4.1.5 S/C Power Set and Cables

This console shall consist of a ground power supply and regulator which shall be used to keep the test battery set in a charged condition and is floated across them during Observatory "power on" condition. It also shall consist of power cables between the console and the Observatory ground power connector.

3.7.4.1.6 Ranging Test Assembly

This console shall provide for a range test of the ranging channel in the C&DH transponder.

3.7.4.1.7 Pyro Test Set

This portable console shall simulate pyro device bridge wire prefiring, firing and post-firing characteristics, measure resistance and verify no voltage on pyros prior to their installation. It shall be used to simulate pyro device action during system test.

3.7.4.1.8 Interface Cable Set

This set of cables shall be designed for use between the Observatory harness and the S/S modules as well as interfacing between the modules and breakout boxes. The number and type required shall be determined by the contractor.

3.7.4.1.9 Solar Simulator

This shall be a light source capable of providing stimulation of the solar array cells to the extent necessary for checkout of the solar array when installed in the Observatory.

3.7.4.1.10 Instrument Interface Simulation

This device shall simulate the Instrument Module interface during Observatory integration without the instrument module.

3.7.4.1.11 Umbilical Simulator

This device shall simulate the launch vehicle interface to the Observatory.

3.7.4.1.12 DITMICO - Program & Cable Set

This program and cable set shall provide for the verification of the Observatory harness. The contractor shall perform a trade of manual checkout vs DITMICO to determine the more cost effective approach to be used.

3.7.4.1.13 Power Module C/O Bench

This C/O bench shall be capable of providing power, loads and variation of loads simulating the mission profile necessary for the integration and verification test of the power S/S to the level of failed assembly identification within the module. Its design shall permit the use of the bench for the power module maintenance and be compatible for use as a checkout and maintenance bench in the NASA LPS during Shuttle operations.

3.7.4.1.14 C & DH Module C/O Bench

This C/O bench shall be capable of providing power, loads and signals necessary for the integration and verification test of the C&DH to the level of the failed assembly within the module. Its design shall permit the use of the bench for C&DH module maintenance and be compatible for use as a checkout and maintenance bench in the NASA LPS during Shuttle operations.

3.7.4.1.15 ACS Module C/O Bench

This C/O bench shall be capable of providing power, loads and signals necessary for the integration and verification test of the ACS to the level of failed assembly identification within the module.

The computer requirements of the bench may be met by utilization of an existing computer external to the bench. Its design shall permit the use of the bench for ACS maintenance and be compatible for use as a checkout and maintenance bench in the NASA LPS during Shuttle operations.

3.7.4.1.16 Propulsion C/O Bench (RCS)

This C/O bench shall provide propellant transfer, pressure, control, and measurement necessary for the integration and verification test of the RCS to the level of failed assembly identification within the module. Fluid storage will be external to this unit which shall provide the interconnect hose between fluid source and the bench. The bench shall be designed to maintain the required fluid cleanliness levels. Its design shall also permit the use of the bench for propulsion module maintenance and as a checkout and maintenance bench in the NASA LPS during Shuttle operations.

3.7.4.1.17 S/C Monitor and Control

This unit shall provide displays and controls for monitoring those test points on the Observatory that are only used during S/C build up integration and verification testing. It shall be compatible with the interface cable set which may be used to provide the interface connection between the GSE connectors and this unit.

3.7.4.1.18 IMP Module C/O Bench

This C/O bench shall be capable of providing power, loads and simulated sensor inputs necessary for the integration and verification testing of the IMP to the level of failed instrument or assembly identification within the module. The bench shall be designed with sufficient flexibility to permit its modification to accommodate follow-on instruments, and to be rapidly reconfigured to any instrument checkout it has been modified for, including the original configuration.

3.7.4.2 MECHANICAL EQUIPMENT

3.7.4.2.1 Interface Adapter Set

The interface adapter set shall mate with the Observatory providing pick-up points for a number of different pieces of handling equipment. It avoids need for multiple attachment points on the Observatory itself. It shall provide for:

- Mounting to the vertical and horizontal support dollies
- Mounting to the hoist bar and sling set
- Mounting for shipping and rotation
- Attachment of the Observatory cover
- Interface points for chock and vibration tests

3.7.4.2.2 Hoist Bar and Sling Set

The hoist bar and sling set shall provide for hoisting the Observatory in a vertical not horizontal position, for transition from the vertical to horizontal position, hoisting the Observatory cover and horizontal dolly with Observatory and cover combination.

3.7.4.2.3 Support Dolly-Vertical

This dolly shall provide a platform to hold the Observatory in vertical position during build up and test. It shall permit local movement for shop operations.

3.7.4.2.4 Support Dolly Horizontal

This dolly shall provide a platform to hold the Observatory in a horizontal position for movement in inter and intra plant operations, as well as for long distance transportation. It shall permit the attachment of the Observatory cover which in combination shall provide a protective enclosed structure.

3.7.4.2.5 Access Work Stand

The stand shall be designed to provide access to the S/C during vertical assembly, build up and test. The stand shall be easily removable and provide coverage around the Observatory.

3.7.4.2.6 Skin Storage Rack

The skin storage rack shall be designed to provide a means to safely store skin panels and fragile skin components temporarily removed from the EOS during checkout and test operations.

3.7.4.2.7 Weight and CG Fixture

The weight and cg fixture shall be designed to be capable of determining the weight and cg location of the Observatory. It shall be used in conjunction with the vertical support dolly, interface adapter, load cells and digital read out equipment.

3.7.4.2.8 Mass Simulator Set

This set shall be designed to simulate the mass and cg of each replaceable component of the Observatory. Attachment points shall be provided to permit the simulators to be installed in the Observatory in their respective positions.

3.7.4.2.9 Support Dolly-Module

The module dollies shall be designed to provide for the support, local shop movement and transportation of the S/S modules. Attachment points shall permit the securing of the modules to the dolly and sufficient clearance shall be provided to permit attachment of sling and hoist bar.

3.7.4.2.10 Shipping Containers - Modules

These containers shall be designed to provide an enclosed cover together with the support dolly for protection during shipment for each module. The cover shall permit the introduction of GN_2 for the purpose of cleanliness maintenance during the shipment.

3.7.4.2.11 Observatory Cover Set

This Observatory cover shall be designed to provide an enclosure for the Observatory in conjunction with the horizontal dolly for protection during shipment. It shall permit the introduction of GN_2 for the purpose of cleanliness maintenance during the shipment.

3.7.4.2.12 Humidity Control Kit

A humidity control kit shall be provided for the Observatory during shipment. Humidity control shall be maintained by the pressure maintenance unit which will supply dry nitrogen to the control kit. A humidity indicator shall be located on the Observatory cover. Desiccators shall also be utilized to aid in humidity control.

3.7.4.2.13 Shipping Container-Solar Array

The shipping container shall be designed to provide protection for the solar array during shipment.

3.7.4.2.14 Solar Array Installation and Deployment Fixture

The fixture shall be designed to support the solar array during installation and check-out of the functioning of the deployment mechanism. It shall provide support to overcome gravity loads in earth environment while the solar array is attached in an undeployed and deployed position on the Observatory. It shall also be capable of providing the support as the solar array is deployed.

3.7.4.2.15 Transporter

The transporter shall be designed as the undercarriage support during EOS transportation for the Pressure Maintenance Unit, GN_2 Storage System, and GN_2 Manifold and Supply Platform. It shall also be designed to be secured as a unit with the Observatory cover and horizontal dolly during transportation.

3.7.4.2.16 Indicating Accelerometer Kit

An indicating accelerometer kit shall provide a permanent, direct reading record of the dynamic environment to which the Observatory has been subjected during transportation.

3.7.4.2.17 Pyro Installation Tool Kit

The pyro tool kit shall provide all the tools necessary for the installation and removal of pyro hardware on the Observatory.

3.7.4.2.18 Storage Motor Installation Fixture

This is a fixture required only when the EOS contains an orbit kick stage (Follow-on Mission C&E). The fixture shall be designed to support the stage motor and to lift and guide it during installation into and removal from the Observatory.

3.7.2.4.19 Observatory/Shuttle Simulator Fixture

This fixture shall provide a simulation of the Shuttle attachment points for verification of the correct placement and fit of Observatory corresponding attachment points and fittings.

3.7.4.2.20 Tie Down Kit

This kit shall provide all the tie down hardware required by the Observatory and S/S modules during transportation such as steel rope and turnbuckles and attachment hardware. The hardware shall be compatible with the securing points of the transportation vehicle.

3.7.4.2.21 Battery Shipping Container

This container shall be designed to provide protection for the flight batteries during shipment from the battery vendor and the contractor.

3.7.4.2.22 Battery Installation Tool

The battery installation tool shall be designed to support and guide the flight and test batteries during installation into and removal from the Power S/S Module.

3.7.4.3 FLUID EQUIPMENT

3.7.4.3.1 GN_2 Conditioning Unit

The GN_2 Conditioning Unit shall be a mobile unit with a manifold containing a hose connection for attaching the unit to an external source of GN_2, filter, pressure regulator, heater with controls, flow meter, flow control valve, shut-off valve, relief valve and associated plumbing. The unit shall be capable of receiving GN_2 from an external source delivering controlled GN_2 at a maximum temperature of 120 degrees and a maximum pressure of 421 Kg/cm^2 with a controlled flow rate up to TBD m^3/min. This unit is used for purging and drying of propulsion module components after test.

3.7.4.3.2 GN_2 Regulation Unit

The GN_2 Regulation Unit shall consist of gages, manual shut-off valves and hand loading regulators. The unit shall be capable of accepting inlet pressures of up to 253 Kg/cm^2 and regulating it over a 7 to 28 Kg/cm^2 range. This unit is used for testing of propulsion module components.

3.7.4.3.3 Volumetric Leak Detector

This device shall be designed to provide a volumetric displacement method to measure leaks during propulsion module checkout.

3.7.4.3.4 RCS Vacuum Test Cart

The RCS vacuum test cart shall be a portable unit containing a vacuum pump, valves, discharge, trap, vacuum gage and shall have provision for remote or local control. The test cart shall be capable of creating a vacuum of TBD mm Hg maximum absolute pressure in the propulsion module feed lines and tanks at a rate of TBD m^3/min.

3.7.4.3.5 Fluid Distribution System - Grumman

This system shall consist of hoses and pipes at the contractor's facility necessary for Observatory fluid operations.

3.7.4.3.6 Pressure Maintenance Unit

A pressure maintenance unit shall be designed to provide accurately regulated, clean, dry GN_2 at the required pressure to maintain the EOS and such items as propellant tanks and feed lines in a pressurized state during transportation. The unit shall be attached to the transporter and to the Observatory shipping container (Dolly & Cover) during EOS transportation.

3.7.4.3.7 GN_2 Storage System (Transporter)

The GN_2 Storage System shall provide a source of high pressure GN_2 to the Pressure Maintenance Unit when the Observatory is being transported. It shall be secured to the Transporter and consist of GN_2 storage cylinders connected to a manifold which shall control flow by means of valves and pressure gages.

3.7.4.3.8 GN_2 Manifold and Supply Platform

The GN_2 Manifold and Supply Platform shall provide replacement of GN_2 leakage loss from the EOS shipping cover during transportation. Flow shall be controlled by a shut-off valve, a pressure gage and connecting manifolds. It shall be secured to the transporter and connected to GN_2 storage cylinders.

3.7.4.3.9 Fluid Distribution System - Launch Site

The system shall consist of hoses and pipes for fluid distribution to the Observatory, peculiar to the launch site.

3.7.4.3.10 Propellant Transfer Assembly

The propellant transfer assembly shall be a pressure-fed system capable of controlling and transferring a specified quantity of fuel (or simulated propellant) to the OT/RCS Module Tanks.

3.7.4.3.11 Mass Spectrometer Leak Detector

The Mass Spectrometer shall provide the capability to indicate and detect a leak in a OT/RCS Module component or line.

3.8 PRECEDENCE

The order of precedence of specifications places the EOS Program Specification first, then this specification, and then the mission configuration specification, and the subsystem configuration last.

QUALITY
ASSURE. PROVISIONS

4 - QUALITY ASSURANCE PROVISIONS

4.1 GENERAL

This section defines the test/verification activities required to verify that the LRM Observatory complies with the performance, design and workmanship requirements specified in Section 3 of this document in accordance with the following documents:

- S-320-G-1, General Environmental Test Specification For The Spacecraft and Components.
- Report 4 - EOS System Definition Study, Grumman Management Approach Recommendations dated 15 July, 1974.

(a) Methods of verification shall be those defined below:

(1) Inspection - Verifies conformance of physical characteristics to related requirements without the aid of special laboratory equipments.

(2) Demonstration - Verifies the required operability of hardware and computer programs, without the aid of test devices.

(3) Similiarity - Verifies that the Observatory components satisfy their performance and design requirements, based upon the certified qualification of similiar components under identical or similar operating conditions.

(4) Analysis - Verifies conformance to requirements, based upon studies, calculations and modeling.

(5) Test - Verifies conformance to required performance/physical characteristics and design/construction by instrumented functional operation and evaluation techniques.

(b) Test Levels applicable to the EOS Observatory are defined below:

(1) Component Level - The tests conducted at the black box level of assembly.

(2) Module Level - The tests performed on the subsystem module, mission peculiar module, instrument, or structure element level of assembly.

(3) Spacecraft Level - The tests performed with the Spacecraft subsystem modules installed and integrated into the Basic Spacecraft, and structural test performed with mass simulation of Observatory elements.

(4) Observatory System Level - The Observatory System level tests performed with the instruments and Mission Peculiar Modules installed in the Spacecraft.

(c) Test Types - The following test types have been identified as applicable to the EOS Observatory.

(1) Development Tests - Model testing, compatibility demonstrations and computer simulations to support the engineering analysis of Space Segment performance .

(2) Qualification Tests - Component, module, Spacecraft and Observatory level tests of flight type hardware performance and design when subjected to greater than mission type environments.

(3) Integration Tests - Observatory segment mechanical and electrical tests which verify interface compatibility and functional performance using applicable support equipment and procedures.

(4) Acceptance Tests - Component, module, Spacecraft and Observatory systems and environmental tests which verify flight hardware workmanship and flight readiness.

(d) Location of Testing - The location of the Observatory Segments tests shall be in accordance with the EOS System Test Plan and the EOS Master Program Schedule.

4.1.1 RESPONSIBILITY FOR INSPECTIONS AND TEST

The Observatory prime contractor shall be responsible for conducting all Observatory Testing specified herein. Integration of the instruments and Observatory level tests shall be supported by the Instrument Contractor. Instrument performance when undergoing observatory level tests shall be the responsibility of the Instrument Contractor.

4.1.2 SPECIAL TESTS AND EXAMINATIONS

4.1.2.1 SPACECRAFT SPECIAL DEVELOPMENT TESTS AND EXAMINATIONS

This paragraph contains the ground rules for performance of component development tests for new EOS components and a description of subsystem, module, Spacecraft and Observatory level development tests.

4.1.2.1.1 Components

4.1.2.1.1.1 Component Development Tests

For those components to be new designs for the Spacecraft the contractor shall conduct specific development tests including step-stress and overstress tests to validate design assumptions, optimize design, effect corrective action early in the program and provide the basis for estimating reliability growth. In selecting development tests, the following shall be considered:

(a) Component breadboard tests to verify feasibility and aid in the selection of materials and parts.

(b) Component mock-up tests where thermal or loads analyses are complex or unfeasible.

(c) Two types of testing shall be considered;

(1) Thermal - To be performed on components to assess the temperature characteristics of the components and to determine the need for added cooling provisions.

(2) Mechanical - To be performed on an actual component chassis, using simulated mass loading of all internal parts in order to determine response data and the need for isolation, damping, structural stiffening, or weight reduction.

4.1.2.2 STRUCTURE SUBSYSTEM

4.1.2.2.1 Cantilevered Mode Survey

A modal survey of the structure subsystem, with its installed equipment, shall be performed to determine all mode shapes, modal damping coefficients, and modal resonance frequencies below 60 Hz. The test article shall be of flight quality including the adapter interfaces in every detail except for mass/inertia simulators for nonstructural components. This shall be a cantilever test with the test article attached to the adapter and the adapter shall be cantilevered at the launch vehicle interface.

4.1.2.2.2 Module Structural Tests

4.1.2.2.2.1 Vibration and Acoustic Tests

Acoustic and mechanical vibration tests shall be performed to define the acceptance test environments for adequate screening or component workmanship at the module level.

Test articles shall be of flight quality and complete in every detail except for mass/inertia simulators for nonstructural components. The modules shall be attached to the test fixtures so that no amplification or attenuation of the module environmental inputs is induced by the fixture which are not representative of inputs from the Spacecraft structure. Acoustic and vibration test levels shall be as specified in the subsystem module end item specifications.

4.1.2.3 COMMUNICATIONS AND DATA HANDLING SUBSYSTEM

4.1.2.3.1 Antenna Pattern Tests

A 1/5 scale model antenna pattern test shall be performed to verify antenna coverage and gain performance requirements specified in Paragraph 3.7.1.1.

4.1.2.4 WIDE BAND COMMUNICATIONS SUBSYSTEM

4.1.2.4.1 Antenna Pattern Tests

A full scale antenna pattern test shall be performed to verify antenna coverage, and gain performance specified in Paragraphs 3.7.3.1.1 and 3.7.3.1.8.

4.1.2.4.2 Isolation Tests

Isolation of X/Ku and S-Band antennas specified in Paragraph 3.7.3.1.8 shall be verified during full scale tests if isolation cannot be verified by analysis.

4.1.2.5 THERMAL SUBSYSTEM DEVELOPMENT TESTS

4.1.2.5.1 Module Thermal Model Development Tests

Module level thermal vacuum, thermal balance test shall be performed to verify the module thermal analysis model. The test articles shall consist of the module structure, thermal simulation of installed equipment and a test set of external skins.

4.1.2.6 FLIGHT, GROUND AND LAUNCH ELEMENT COMPATIBILITY TEST

4.1.2.6.1 Observatory Command and Data Link/STDN

The compatibility between the flight observatory S-Band command and data link STDN shall be demonstrated prior to launch. The flight observatory S-Band link shall be demonstrated with the Observatory in flight configuration at WTR via radiation to the WTR ground station, with the PCC and remote sites on line and recurring and displaying down link S-Band data. Prior to delivery of the Observatory to WTR demonstration of STDN compatability shall be performed via hardline.

4.1.2.6.2 Instrument Communication and Data Handling/TDRS and STDN

The compatability of the Instrument Communication and Data Handling hardware and software with the TDRS and STDN shall be demonstrated prior to launch. The flight Instrument, Instrument Communication and Data Handling Module and X/Ku-Band antennas shall be tested at GSFC with the primary ground station prior to integration into the flight spacecraft.

4.1.2.6.3 Launch System

Range and launch vehicle RFI compatibility shall be verified prior to launch. RFI tests shall be performed between the range, launch vehicle and Spacecraft by simulation of launch RF environment during flight observatory integration and test. Demonstration of Observatory and launch system RFI compatibility shall also be performed during prelaunch operations with the flight observatory at WTR.

4.1.2.6.4 Launch Vehicle Interface

4.1.2.6.4.1 Mechanical

A match mate between the S/C adapter and the launch vehicle mechanical interface shall be performed. The flight S/C adapter shall be shipped to the launch vehicle manufacturer and match mated including shimming and machine of the interfaces to meet the flatness and indexing requirements defined in the S/C - Launch Vehicle ICD.

4.1.2.6.5 Observatory Service Tower Interfaces

4.1.2.6.5.1 Electrical Umbilical

Prior to mating of the Observatory to the service tower launch and prelaunch checkout umbilicals validation of the functional interfaces shall be performed. Pin-to-pin isolation of greater than 2.0 megohms, and end to end ringout of all umbilical cables shall be verified. Functional interface shall be verified from the launch complex GSE through to the Spacecraft umbilical interface using a breakout box and standard test equipment.

4.1.2.6.5.2 Air Conditioning and Fluid Interfaces

Prior to mating service tower air conditioning and fluid lines to the Observatory, conformance to the cleanliness, flow rates, temperature and dew point requirements as specified in the Launch Complex ICD must be verified.

4.2 QUALITY CONFORMANCE INSPECTION

Formal qualification test, analyses and inspections shall be conducted to validate that the Observatory, its modules and components satisfy the design, performance and workmanship requirements specified in Section 3 of this document. Test specimens shall be identical to those of the flight articles. Where environmental conditions cannot be properly or conservatively simulated in test, allowance for material properties, combined loading and other missing effects shall be provided for in test procedure and applied loads and tests supplemented by analyses. Where prior loading histories affect the structural adequacy of a test article, these shall be included in all test requirements. Instrumentation shall be provided in order to evaluate test results. All qualifications testing shall be completed prior to first flight.

4.2.1 COMPONENT QUALIFICATION TESTS

All components shall be qualified to the requirements of the GSFC General Environmental Test Specification S-320-G-1 and MIL STD 810B, with the exceptions identified in Paragraph 4.3 herein. Component environmental qualification test levels shall be as identified in the individual component and subsystem and Item Specification. Components with existing designs may be qualified by similiarity to previous applications and previous test history.

4.2.2 MODULE QUALIFICATION TESTS

All modules and Instruments shall be qualified to the component requirements of the GSFC General Environmental Test Specification S-320-G-1. Module structural qualification shall be conducted in conjunction with observatory level structural qualification tests. Module thermal design qualification shall be conducted in as part of observatory level qualification tests. Module design environmental levels to be used in establishing test levels shall be as defined in the subsystem end item specification.

4.2.3 OBSERVATORY QUALIFICATION TESTS

A observatory level qualification test program shall be performed in accordance with the requirements of the GSFC General Environmental Test Specification S-320-G-1. Static load tests, acoustic tests, mechanical vibration, shock and separation and deployment system qualification tests shall be performed on a fully representitive Spacecraft, with modules and instrument structure and mass, representation for non structural components. Measurement of acoustic, and vibration and shock loads shall be used to verify component design environments prior to component qualification. After completion of the separation and deployment qualification tests, the qualification structure shall be wired, qualification components integrated and Observatory Systems Acoustic, Thermal Vacuum, Thermal balance, EMC and Systems Performance qualification tests performed.

4.2.3.1 OBSERVATORY QUALIFICATION TEST ENVIRONMENTS

The Observatory qualification test article shall be subjected to the environments specified below and in accordance with the requirements of NASA GSFC S-320-G-1 except as noted. During the Observatory system qualification tests the Observatory shall be examined and functionally tested before and after each environmental exposure. During the test, the Observatory shall be operated in the appropriate mission phase duty cycle.

4.2.3.1.1 Static Load

The Observatory structural model shall be subjected to a static load test. The test levels to be applied shall be determined from a combined Observatory/Launch Vehicle dynamic loads analysis, Observatory structural loads and stress analyses for the worst case conditions of Tables 4-1, 4-2 and 4-3. Static tests shall include a demonstration of secondary structure survival of Shuttle crash loads defined in Table 4-3.

4.2.3.1.2 Acoustic Field

The Observatory shall be exposed to a broadband random sound field with an overall sound pressure level of 149 dB (Re: 20 Newton/m^2). The octave band sound pressure levels shall be as specified in Table 4-4. The Observatory shall be mounted on a flight-type adapter during the test.

4.2.3.1.3 Sinusoidal Vibration

The Observatory shall be attached to a vibration fixture using a flight-type adapter and flight-type clamp. Sinusoidal vibration excitation shall be applied at the base of the adapter along each of the three orthogonal axes. Shuttle mounting points shall be evaluated for additional dynamics test input locations. The test levels and logarithmic frequency sweep rate shall be as shown in Table 4-5. The reduction of the sinusoidal vibration test levels, in the Spacecraft's resonant frequency band, will be required in order to prevent the application of unrealistic loads. This "notching" of the input levels shall be determined by dynamic analysis of the Spacecraft in combination with the Launch Vehicle.

4.2.3.1.4 Mechanical Shock

The Observatory shall be subjected to a mechanically applied shock transient to the Spacecraft/Launch Vehicle interface twice along each of the three orthogonal axes. The test level, using shock spectral analysis with a Q =10, shall be defined in terms of shock response spectrum and in accordance with Fig. 4-1.

4.2.3.1.5 Pyrotechnic Shock

The Observatory shall be subjected to two pyrotechnic separation tests. In addition to the Spacecraft, the test shall include the flight-type adapter, flight-type clamp and pyrotechnic devices. The Observatory shall also be subjected to additional pyrotechnic shocks dependent on the type and quantity of release devices used for solar arrays, antennas, etc.

Table 4-1 Ultimate Load Factors Delta Launch Vehicle

CONDITION	LONGITUDINAL X	LATERAL Y OR Z
LIFT - OFF	+ 4.35 - 1.5	3.0
MAIN ENGINE CUT OFF	+18.45	1.0

(1)T5-52

Table 4-2 Ultimate Load Factors Titan III B/NUS Launch Vehicles

CONDITION	LONGITUDINAL X	LATERAL Y OR Z
LIFT - OFF	+ 3.45 - 1.2	3.0
STAGE I SHUTDOWN (DEPLETION)	+12.3 - 3.75	2.25
STAGE II SHUTDOWN (COMMAND)	+16.2 - 3.0	2.25

NOTES:

1. LIMIT LOAD FACTOR TIMES 1.5
2. LOAD FACTOR CARRIES THE SIGN OF THE EXTERNALLY APPLIED LOAD.
3. INCLUDES BOTH STEADY STATE AND DYNAMIC CONDITIONS.

(1)T5-53

Table 4-3 Ultimate Load Factors Shuttle

CONDITION	DIRECTIONS		
	X	Y	Z
LIFT - OFF (4)	+2.55 ± 0.9	± 0.45	+1.20 +0.30
HIGH Q BOOST	+2.85	± 0.30	-0.30 +0.75
BOOSTER END BURN	+4.5 ± 0.45	± 0.30	+0.60
ORBITER END BURN	+4.5 ± 0.45	± 0.30	+0.75
SPACE OPERATIONS	+0.30 -0.15	± 0.15	± 0.15
ENTRY	± 0.38	± 0.75	-4.5 +1.5
SUBSONIC MANEUVERING	± 0.38	± 0.75	-3.75 +1.5
LANDING AND BRAKING	± 2.25	± 2.25	-3.75
CRASH	-9.5 +1.5	± 1.5	-4.5 +2.0

NOTES:

1. LIMIT LOAD FACTOR TIMES 1.5
2. LOAD FACTOR CARRIES THE SIGN OF THE EXTERNALLY APPLIED LOAD. POSITIVE X, Y, Z, DIRECTIONS EQUAL FORWARD, RIGHT AND DOWN.
3. CRASH LOAD FACTORS FOR THE NOMINAL PAYLOAD OF 65,000 LB AND ONLY USED TO DESIGN PAYLOAD SUPPORT FITTINGS.
4. THESE FACTORS INCLUDE DYNAMIC TRANSIENT LOAD FACTORS.

(1) T5-54

Table 4-4 Acoustic Noise Spacecraft Qualification Test Levels

OCTAVE BAND	
CENTER FREQUENCY (HZ)	SOUND PRESSURE LEVEL (DB*)
31.5	131
63	137
125	142
250	144
500	143
1000	141
2000	137.5
4000	135
8000	133
OVERALL	149.5
DURATION: 2 MINUTES	

*DB RE: 20 μ NEWTONS/M^2

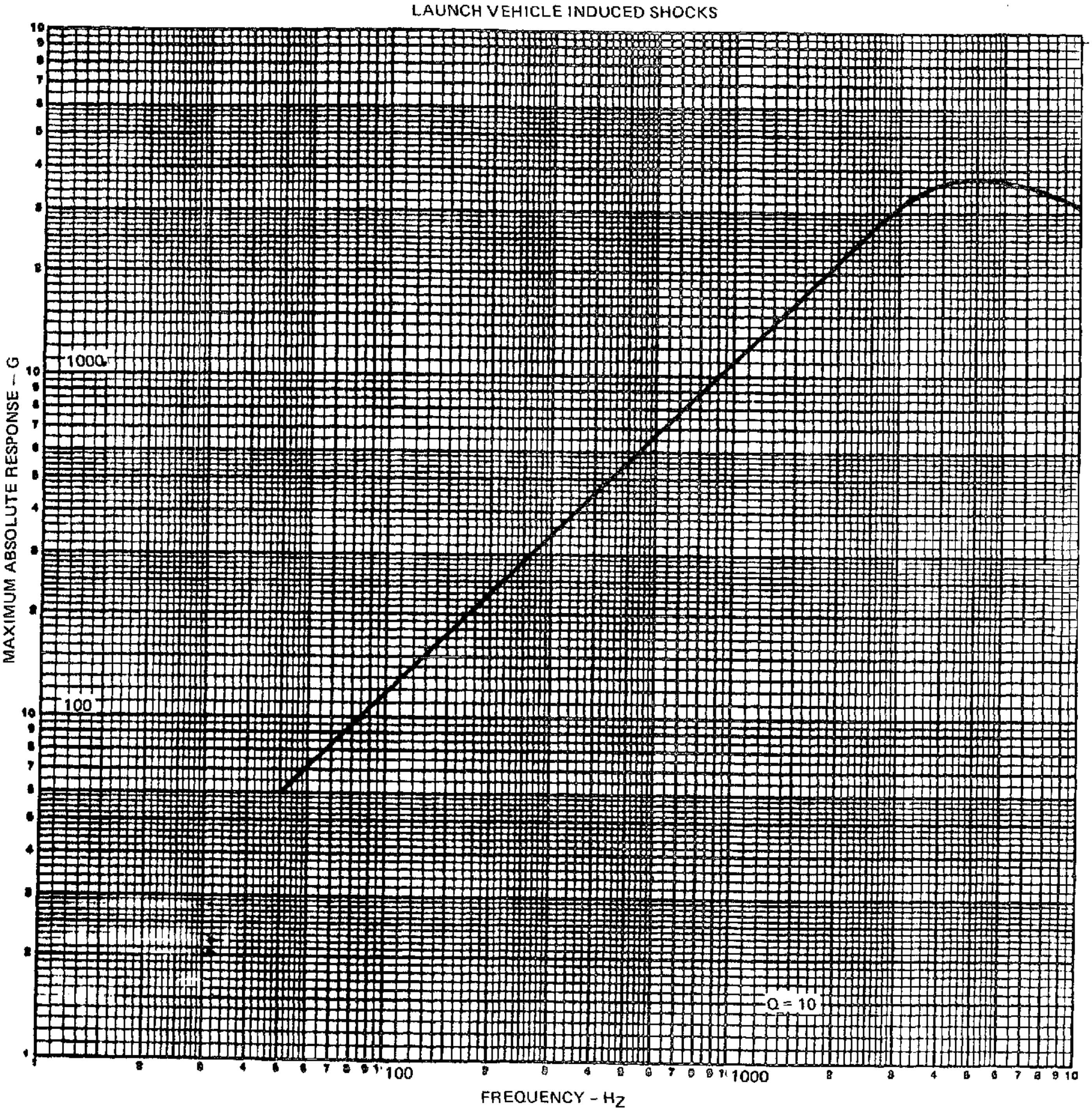
(1)T5-55

Table 4-5 Sinusoidal Vibration Spacecraft Qualification Test Levels

AXIS OF EXCITATION	FREQUENCY RANGE (HZ)	ACCELERATION ZERO-TO-PEAK ± (g)
LONGITUDINAL (X-X)	5 - 9.5	12.7 MM D.A.
	9.5 - 15	2.3
	15 - 21	6.0
	21 - 50	3.0
	50 - 200	2.3
LATERAL (Y-Y) & (Z-Z)	5 - 7.1	19.0 MM D.A.
	7.1 - 22	2.0
	22 - 200	1.5
SWEEP RATE: 2 OCTAVES/MINUTE		

(1)T5-56

NOTE: QUALIFICATION TEST LEVELS



(1) 5-29

Fig. 4-1 Shock Response Spectrum at Observatory/Launch Vehicle Interface

4.2.3.1.6 Thermal Vacuum

The Observatory shall be subjected to a thermal vacuum, thermal balance qualification test to verify the Observatory Thermal Design and Observatory Systems performance when subjected to nominal and off nominal mission thermal environments. The test environments shall be in accordance with the requirements of Paragraph 3.7.1.5.1. For observatory environments other than equipment the qualification temperature shall be $10^{o}C$ above and below the design goal temperature. The test article shall be representative of the flight observatory with the exception of the Solar Arrays which are not required to be installed and the use of heater skins, in place of flight skins, to provide thermal stimulation.

4.2.4 SUPPORT EQUIPMENT QUALITY CONFORMANCE

All support equipment shall be visually inspected and functionally tested to approved engineering prepared tests procedures to assure quality conformance. Functional tests shall consist of equipment operation demonstrating its ability to perform the required EOS test or function within tolerances.

4.2.5 SOFTWARE QUALITY ASSURANCE

The Observatory software shall be verified in a cascade of evaluations, beginning at a level suitable for the design stage of the element of software being tested. Specifications for performance at each level shall be prepared in parallel with the development of the software to be evaluated.

4.2.5.1 ALGORITHM LEVEL

Testing at the algorithm level shall provide assurance that the element of software being tested meets the following criteria:

(a) Accepts the full range of all input values

(b) Provides the specified function within an acceptable time

(c) Produces the full range of output values

(d) Performs the proper input and output linkages.

4.2.5.2 MODULE LEVEL

Testing at the module level shall provide assurance that each module provides the desired functions in the presence of software driving programs which simulate the inputs from other portions of the observatory software. Test criteria shall be:

(a) Proper control flow

(b) Proper stability

(c) Appropriate numerical results.

4.2.5.3 SYSTEM LEVEL

Testing at the system level shall provide assurance that the system is compatible with the hardware system of the Spacecraft and the simulated inputs of the operating environment. Test criteria shall be:

(a) Lack of internal interference

(b) Proper data flow

(c) Proper hardware stability and function

(d) Expected systems reactions to simulated operating environment.

4.2.5.4 OPERATIONAL LEVEL

Operational Level testing shall provide assurance that all software functions are compatible with safe and proper spacecraft operation. Operational testing will begin with all but essential software functions inhibited at the output point. As the test progresses, the outputs will be sequentially enabled and the Observatory Performance monitored for appropriate operation. The test criteria shall be:

(a) Stability

(b) Acceptable error levels

(c) Duplication of results found in the system level tests

(d) Suitable mission performance.

4.3 ACCEPTANCE TESTS

Acceptance tests shall be performed to verify that the workmanship of the Observatory, and its modules and components satisfy the Observatory requirements specified in Section 3 of this specification. Acceptance tests shall consist of functional and environmental tests of the EOS Observatory and/or its modules and components.

4.3.1 COMPONENT ACCEPTANCE TESTS

Component level acceptance tests shall consist of reliability (burn in) tests and functional acceptance tests conducted at the component manufacturer's plant. Specific test requirements shall be defined in the component and item specification. Components environmental acceptance tests shall be conducted at the module level of Observatory assembly.

4.3.2 MODULE ACCEPTANCE TESTS

Module level acceptance tests shall consist of harness continuity and dialectric strength, acoustic or mechanical vibration, thermal cycling and thermal vacuum tests. Selection of acoustic or mechanical vibration for module acceptance tests shall be based on the module development tests, which will define the environments seen by the components.

4.3.3 OBSERVATORY ACCEPTANCE TESTS

Observatory level acceptance tests shall consist of spacecraft functional tests, systems EMC/RFI, weight and CG, Ambient Environment Mission Profile System Test, a Workmanship Acoustic Test, and Separation and Development mechanical systems functional tests. Observatory system performance as well as subsystem trend data shall be evaluated throughout the Observatory acceptance test program to verify Observatory flight readiness.

4.3.3.1 SPACECRAFT FUNCTIONAL TESTS

Prior to integration of the flight instruments into the Spacecraft satisfactory performance of the integrated subsystems shall be verified.

4.3.3.2 SYSTEMS EMC/RFI

Tests shall be conducted on the integrated EOS Observatory to verify that the system level performance of the space vehicle is within specification and to verify integrated subsystem EMC. EMC tests shall consist of verification of Observatory performance in all mission operating modes while the Command and onboard Processing System is operated throughout its command and control matrix.

RFI tests shall consist of monitoring of the vehicle performance in launch configuration and operating mode when subjecting the Observatory to launch RF environments.

Specific measurement of induced energy for personnel and Observatory systems safety functions such as pyrotechnic lines and Spacecraft command telemetry lines shall be made during RFI and EMD tests.

4.3.3.3 WEIGHT AND CG

The weight and cg of the Observatory shall be measured. The tests shall be conducted with propellant or inert substitute fluid loaded into the OA/RCS tanks. This test shall verify the weight and cg requirements of Section 3 of this specification.

4.3.3.4 AMBIENT ENVIRONMENT MISSION PROFILE SYSTEMS TESTS

The Observatory System Performance shall be evaluated by running simulated Orbital Profiles on the Observatory Subsystems including bus voltage limits, simulated solar array sun tracking and power inputs, battery charge/discharge and OBP operational routines. The Observatory performance shall be evaluated for TBD simulated orbits.

4.3.3.5 WORKMANSHIP ACOUSTIC TESTS

The Observatory shall be exposed to a broad band random sound field. The octave band sound pressure shall be as specified in Section 3, Table 3-3. An abbreviated system functional test shall be performed after completion of the acoustic test to verify observatory systems survival.

4.3.3.6 SEPARATION AND DEPLOYMENT TESTS

The completion of the Workmanship Acoustic test, the separation mechanical system, and solar array Deployment mechanism and drive, shall be functionally checked. Separation Pyrotechnic latches shall be fired and the Observatory lifted off the adapter. Solar Array deployment pyrotechnics shall be verified. With the arrays supported in the extended position, the functional operation of the solar array drive mechanism shall be verified.

4.4 TEST VERIFICATION MATRIX

Table 4-6 defines the Observatory Verification Requirements. The definitions of verification methods shown in the matrix are presented in Paragraph 4.1. Each performance requirement specified in Section 3 of this document shall be verified as noted in the Verification Matrix.

Table 4-6 Requirements Verification Matrix (Sheet 1 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.0	Requirements	X																				
3.1.1	Program Driver Reqmts	X																				
3.1.2	General Description	X																				
3.1.3	Program Costs	X																				
3.1.4	Missions	X																				
3.1.5	Systems Diagrams	X																				
3.1.6	Observatory Interface Def.	X																				
3.1.7	GFE	X																				
3.1.8	Operational and Org. Concepts	X																				
3.2	Characteristics	X																				
3.2.1	Performance	X																				
3.2.1.1	Observatory Performance Characteristics					X																
3.2.1.2	Support Equipment Performance Characteristics					X																
3.2.2	Physcial characteristics	X																				
3.2.2.1	Observatory Physcial Characteristics			X																		
3.2.2.1.1	Mass Properties													X		X		X				
3.2.2.2	Support Equipment Physical Characteristics		X	X		X																
3.2.3	Reliability	X																				
3.2.3.1	Quantitative Requirements					X																
3.2.3.2	Reliability/Maintainability Program			X																		
3.2.5	Maintainability	X																				
3.2.4.1	Ground Refurbishment					X																
3.2.7	Environmental Conditions	X																				
3.2.7.1	Observatory Environmental Conditions	X																				
3.2.7.1.1	Transportation, Handling and Storage Environments					X																
3.2.7.1.1.1	Packaged Natural Environments					X																
3.2.7.1.1.1.1	Altitude-Air Transport					X																

VERIFICATION CATEGORIES

1 Inspection

2 Demonstration

3 Similarity

4 Analysis

5 Test

- A Development Test
- B Qualification Test
- C Integration Test
- D Acceptance Test

(1)5-58(1)

Table 4-6 Requirements Verification Matrix (Sheet 2 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.2.7.1.1.2	Temperature					X																
3.2.7.1.1.1.3	Solar Radiation					X																
3.2.7.1.1.1.4	Rain					X																
3.2.7.1.1.1.5	Humidity					X																
3.2.7.1.1.1.6	Fungi					X																
3.2.7.1.1.1.7	Atmospheric Corrosion					X																
3.2.7.1.1.1.8	Abrasion					X																
3.2.7.1.1.2	Packaged Induced Environments							X						X		X						
3.2.7.1.1.2.1	Sustained Acceleration-Hoisting							X								X						
3.2.7.1.1.2.2	Vibration							X						X		X						
3.2.7.1.1.2.3	Shock							X								X						
3.2.7.1.2	Pre launch Natural Environment					X																
3.2.7.1.2.1	Temperature					X																
3.2.7.1.2.2	Solar Radiation					X																
3.2.7.1.2.3	Rain					X																
3.2.7.1.2.4	Humidity					X																
3.2.7.1.2.5	Fungi					X																
3.2.7.1.2.6	Atmospheric Corrosion					X																
3.2.7.1.2.7	Explosive Atmospheric					X																
3.2.7.1.2.9	Particulates		X																			
3.2.7.1.3	Launch Ascent Environment					X		X			X			X		X		X		X		X
3.2.7.1.3.1	Acoustic Field										X			X				X		X		X
3.2.7.1.3.2	Sinusoidal Vibration							X			X					X						
3.2.7.1.3.3	Rand om Vibration							X			X					X						
3.2.7.1.3.4	Shock							X								X						
3.2.7.1.3.5	Sustained Acceleration							X								X						
3.2.7.1.3.6	Pressure					X		X		X										X		
3.2.7.1.3.7	Temperature					X		X			X											
3.2.7.1.4	Orbital Environment					X		X								X				X		
3.2.7.1.4.1	Acceleration and Sustained Loads							X														
3.2.7.1.4.2	Vibration							X			X					X						
3.2.7.1.4.3	Shock																			X		X

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Table 4-6 Requirements Verification Matrix (Sheet 3 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																				
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)				
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM				
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D	
3.2.7.1.4.4	Vacuum							X						X						X			
3.2.7.1.4.5	Meteroid					X																	
3.2.7.1.4.6	Temperature					X		X						X						X			
3.2.7.1.4.7	Solar Radiation					X		X															
3.2.7.1.4.8	Geometry Trapped Radiation Environment					X																	
3.2.7.1.4.9	Solar Flare Protons					X		X															
3.2.7.1.4.10	Solar Flare Alpha Particles					X		X															
3.2.7.2	Support Equipment					X																	
3.2.7.2.1	Transportation Handling and Storage Environments					X																	
3.2.7.2.2	Pre launch Environment					X																	

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Table 4-6 Requirements Verification Matrix (Sheet 4 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.3	Design and Construction	X																				
3.3.1	Materials Processes and Parts		X			X	X															
3.3.1.1	Observatory Materials Processes and Parts	X																				
3.3.1.1.1	Polymer Materials					X	X															
3.3.1.1.2	Lubricant					X	X															
3.3.1.1.3	Dissimilar Metals					X																
3.3.1.1.4	Corrosion of Metal Parts		X			X																
3.3.1.1.5	Moisture and Fungus Resistance					X																
3.3.1.1.6	Reaction of Material					X																
3.3.1.1.7	Drains		X			X																
3.3.1.1.8	Fasteners					X																
3.3.1.1.9	Wiring		X			X																
3.3.1.1.10	Stress Corrosion					X																
3.3.1.1.11	Soldering		X			X																
3.3.1.1.12	Glass Fiber Reinforced Plastics					X																
3.3.1.1.13	Electronic Electrical Electromechanical Parts	X																				
3.3.1.1.13.1	Parts Programs					X																
3.3.1.1.13.2	Parts Materials and Processes Control Board					X																
3.3.1.1.13.3	Selection					X																
3.3.1.1.13.4	Parts Manufacturer's Control		X			X																
3.3.1.1.13.5	Screening		X			X																
3.3.1.1.13.6	Derating					X																
3.3.1.1.14	Cleanliness		X			X																
3.3.1.1.15	General Processes	X																				
3.3.1.1.15.1	Heat Treatment					X																
3.3.1.1.15.2	Welding		X			X																
3.3.1.1.15.3	Brasing		X			X																
3.3.1.1.15.4	Sandwich Construction		X			X																
3.3.1.1.15.5	Potting and Encapsulation		X			X																
3.3.1.2	Support Equipment Materials Processes and Parts	X																				
3.3.1.2.1	Corrosion Resistance		X			X																
3.3.1.2.2	Tungus Resistance					X																

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Table 4-6 Requirements Verification Matrix (Sheet 5 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.3.1.2.3	Calibration		X			X																
3.3.1.2.4	Drains					X																
3.3.1.2.5	Fasteners					X																
3.3.1.2.6	Electrical, Electronic, and Electromechanical Parts	X																				
3.3.1.2.6.1	Parts Program					X																

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Table 4-6 Requirements Verification Matrix (Sheet 6 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.3.2	Electromagnetic Radiation																					
3.3.2.1	Observatory					X		X												X		X
3.3.2.2	Support Equipment Electromagnetic Radiation					X														X		
3.3.2.3	Grounding	X																				
3.3.2.3.1	Structure					X							X				X				X	
3.3.2.3.2	Electrical	X																				
3.3.2.3.2.1	Central Point Ground					X			X				X								X	
3.3.2.3.2.2	Shield Grounding		X			X																
3.3.2.3.2.3	Signal Grounds					X			X				X			X						
3.3.2.3.2.4	Party Line/Clock Grounds					X			X				X									
3.3.2.3.2.5	DC Power Circuit Grounds					X			X				X									
3.3.2.3.2.6	AC Circuit Grounds					X			X				X			X					X	
3.3.2.3.2.7	Chassis Grounding					X			X				X			X					X	
3.3.2.3.2.8	Telemetry Circuit Grounds					X			X				X			X					X	
3.3.3	Nameplates and Product Marking	X																				
3.3.3.1	Observatory		X																			
3.3.3.2	Support Equipment Nameplate and Markings		X																			
3.3.4	Workmanship		X																			
3.3.5	Interchangeability	X																				
3.3.5.1	Observatory					X																
3.3.5.2	Support E uipment					X																

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Table 4-6 Requirements Verification Matrix (Sheet 7 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.3.6	Safety	X																				
3.3.6.1	General					X																
3.3.6.1.1	Safety Analysis Report					X																
3.3.6.2	Space Vehicle Safety			X		X																
3.3.6.3	Support Equipment					X																
3.3.6.3.1	Fluid and Mechanical Support Equipment Safety Requirements					X																
3.3.6.3.2	Electrical Equipment					X																
3.3.6.4	Ground Crew Safety Equipment					X																
3.3.7	Human Engineering	X																				
3.3.7.1	Observatory					X																
3.3.7.2	Support Equipment					X																
3.3.8	Software Design and Construction					X			X				X				X				X	
3.4	Documentation	X																				
3.5	Logistics	X																				
3.5.1	Maintenance					X																
3.5.2	Supply					X																
3.5.3	Facilities	X																				
3.6	Personnel and Training	X																				
3.7	Functional Area Characteristics	X																				
3.7.1	Basic Subsystem Functional Characteristics	X																				
3.7.1.1	Communications and Data Handling			X		X	X	X	X				X	X			X			X	X	X
3.7.1.1.1	General Requirements		X			X																
3.7.1.1.2	Functions		X			X																
3.7.1.1.3	Configuration					X																
3.7.1.1.4	Modes of Operation		X			X																
3.7.1.1.5	Performance Requirements	X																				
3.7.1.1.5.1	Communications Group			X		X	X	X	X				X	X			X			X	X	X
3.7.1.1.5.1.1	Command			X		X	X	X	X				X	X			X			X	X	X
3.7.1.1.5.1.2	Telemetry,			X		X	X	X	X				X	X			X			X	X	X
3.7.1.1.5.1.3	Ranging			X		X		X	X				X	X			X			X	X	X
3.7.1.1.5.2	Data Handling Group			X		X	X	X	X				X	X			X			X	X	X

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Table 4-6 Requirements Verification Matrix (Sheet 8 of 19)

SECTION 3 PARAGRAPH			VERIFICATION CATEGORIES																			
		N/A	1	2	3	4	5 (a) COMPONENT				5 (b) MODULE				5 (c) SPACECRAFT				5 (d) OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.7.1.1.5.2.1	On Board Computer					X		X														
3.7.1.1.5.2.2	Command Decoder			X		X		X	X				X	X			X			X	X	X
3.7.1.1.5.2.3	Data Bus System			X		X		X	X				X	X			X			X	X	X
3.7.1.1.5.2.4	Space Craft Clock					X		X					X	X			X			X	X	X
3.7.1.1.5.2.5	Signal Conditioner Assembly					X		X					X	X			X			X	X	X
3.7.1.1.6	Physical Requirements		X			X								X								
3.7.1.1.7	Interface Requirements	X																				
3.7.1.1.7.1	Mechanical Requirements		X			X					X		X		X					X	X	
3.7.1.1.7.2	Thermal Interfaces					X					X			X						X		
3.7.1.1.7.3	Electrical Interfaces					X							X	X			X			X	X	X
3.7.1.1.8	Instrumentation Requirements					X							X	X			X			X	X	X
3.7.1.1.9	Ground Support Equipments								X													
3.7.1.2	Electrical Power Subsystem	X																				
3.7.1.2.1	General Requirements			X		X																
3.7.1.2.2	Functions			X		X																
3.7.1.2.2.1	Solar Energy Conversion					X		X														
3.7.1.2.2.2	Enery Storage			X		X		X	X				X	X			X			X	X	X
3.7.1.2.2.3	Power Control			X		X		X	X				X	X			X			X	X	X
3.7.1.2.2.4	Power Distribution			X		X		X	X				X	X			X			X	X	X
3.7.1.2.2.5	Monitoring			X		X							X	X			X			X	X	X
3.7.1.2.3	Configuration					X																
3.7.1.2.3.1	Basic EPS Configuration			X		X							X	X			X			X	X	X
3.7.1.2.3.2	EPS Configuration Options					X																
3.7.1.2.3.2.1	Main/Auxillary Solar Array Ratio					X																X
3.7.1.2.3.2.2	Battery Engery Storage Capacity					X		X						X								X
3.7.1.2.3.2.3	Redundancy					X																
3.7.1.2.3.2.4	Battery Reconditioning					X								X								X
3.7.1.2.3.3	Electromagnetic Compatibility					X		X						X						X		X
3.7.1.2.3.4	Connectors	X																				
3.7.1.2.3.4.1	Spacecraft Interface Connector												X	X		X				X		
3.7.1.2.3.4.2	Test Connectors					X							X	X								
3.7.1.2.3.5	Harness		X			X							X	X								

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Table 4-6 Requirements Verification Matrix (Sheet 9 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.7.1.2.4	Modes	X																				
3.7.1.2.4.1	Battery Modes					X		X						X						X		X
3.7.1.2.4.2	Main Solar Array Control Modes					X		X						X						X		X
3.7.1.2.4.3	Mission Peculiar Modes					X								X						X	X	X
3.7.1.2.5	Performnace Requirements		X			X								X						X		X
3.7.1.2.5.1	Bus Characteristics					X								X						X		X
3.7.1.2.5.2	Power Output					X								X						X		X
3.7.1.2.5.3	Batteries					X														X		X
3.7.1.2.5.4	Battery Charging													X						X		X
3.7.1.2.6	Physical Requirements		X			X								X								
3.7.1.2.7	Interfaces	X																				
3.7.1.2.7.1	Mechanical Interfaces		X			X					X		X			X				X	X	
3.7.1.2.7.2	Thermal Interfaces					X					X			X						X		
3.7.1.2.7.3	Electrical Interfaces					X							X	X			X			X	X	X
3.7.1.2.7.4	Command and Data Handling Interfaces					X							X	X			X			X	X	X
3.7.1.2.8	Instrumentation Requirements					X							X	X			X			X	X	X
3.7.1.2.8.1	Telemetry					X							X	X			X			X	X	X
3.7.1.2.8.2	Test					X							X	X			X			X	X	X
3.7.1.2.8.3	Resupply					X							X	X			X			X	X	X
3.7.1.2.9	Ground Support Equipment					X							X									
3.7.1.3	Attitude Control Subsystem Module	X																				
3.7.1.3.1	General Requirements			X		X		X						X					X	X		X
3.7.1.3.1.1	Spacecraft Mass Proporties					X														X		X
3.7.1.3.1.2	Disturbance due to Instruements					X																
3.7.1.3.1.3	Flexibility Parameters					X													X			
3.7.1.3.2	Functions					X																
3.7.1.3.3	Configuration			X		X																
3.7.1.3.4	Modes					X																
3.7.1.3.4.1	Launch Mode					X																
3.7.1.3.4.2	Control Mode					X																
3.7.1.3.5	Performance Requirements	X																				
3.7.1.3.5.1	Rate Damping					X		X						X						X		X
3.7.1.3.5.2	Coarse Sun Acquisition					X		X						X						X		X
3.7.1.3.5.3	Fine Sun Acquisition					X		X						X						X		X

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Table 4-6 Requirements Verification Matrix (Sheet 10 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																		
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)		
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM		
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.7.1.3.5.4	Rate Hold					X		X						X						X		X
3.7.1.3.5.5	Earth Acquisition					X		X						X						X		X
3.7.1.3.5.6	Earth Pointing Attitude Hold					X		X						X						X		X
3.7.1.3.5.7	Interial-Pointing Attitude Hold					X		X						X						X		X
3.7.1.3.5.8	Survival Mode					X		X						X						X		X
3.7.1.3.6	Physical Requirements			X		X								X								
3.7.1.3.7	Interface Requirements	X																				
3.7.1.3.7.1	Mechanical Interfaces		X			X					X		X			X				X	X	
3.7.1.3.7.2	Thermal Interfaces					X					X			X						X		
3.7.1.3.7.3	Electrical Interfaces					X							X	X			X			X	X	X
3.7.1.3.7.4	Command and Data Handling Interface					X							X	X			X			X	X	X
3.7.1.3.8	Instrumentation Requirements					X							X	X			X			X	X	X
3.7.1.3.8.1	Telemetry					X							X	X			X			X	X	X
3.7.1.3.8.2	Test Points					X							X	X			X			X	X	X
3.7.1.3.9	Ground Support Equipment					X							X	X			X			X	X	X
3.7.1.4	Structure Subsystem					X									X	X				X		
3.7.1.4.1	General Requirements	X																				
3.7.1.4.1.1	Design Approach	X																				
3.7.1.4.1.2	Design Environments					X									X	X				X		
3.7.1.4.1.2.1	External and Internal Load Distribution					X									X	X				X		
3.7.1.4.1.2.2	Combined Loads and Internal Pressure					X										X						
3.7.1.4.1.2.3	Misalignment and Dimensional Tolernace					X										X						
3.7.1.4.1.2.4	Dynamic Loads					X										X						
3.7.1.4.1.2.5	Repeated Loads and Thermal Fatigue					X										X				X		
3.7.1.4.1.2.6	Vibrational and Acoustical Loadings					X										X						
3.7.1.4.1.2.7	Creep Deformation					X										X						
3.7.1.4.1.2.8	Thermal Stresses					X														X		
3.7.1.4.1.2.9	Malfunctions	X																				
3.7.1.4.1.3	Materials Properties and Allowables	X																				
3.7.1.4.1.3.1	Sources					X	X									X				X		
3.7.1.4.1.3.2	Single Load Path Structures.					X										X						
3.7.1.4.1.3.3	Multiple Load Path Structures					X										X						
3.7.1.4.1.4	Strength Requirements	X																				
3.7.1.4.1.4.1	At Limit Load					X										X						

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Table 4-6 Requirements Verification Matrix (Sheet 11 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.7.1.4.1.4.2	At Ultimate Load					X										X						
3.7.1.4.1.4.3	Margain of Safety					X										X						
3.7.1.4.1.5	Stiffness Requirements	X																				
3.7.1.4.1.5.1	Under Limit Loads					X									X	X						
3.7.1.4.1.5.2	Under Ultimate Loads					X										X						
3.7.1.4.1.5.3	Dynamic Properties					X									X	X						
3.7.1.4.1.5.4	Minimum Frequencies					X									X	X						
3.7.1.4.1.5.5	Component and Attachment Stiffness					X									X	X						
3.7.1.4.1.6	Thermal Requirements					X					X									X		
3.7.1.4.1.7.1	Flight Loads														X	X				X		
3.7.1.4.1.7.2	Non Flight Loads														X	X				X		
3.7.1.4.1.7.3	Pressure Vessels					X		X		X				X						X		X
3.7.1.4.1.8	Dynamic Environment Safety Factors					X					X					X				X		
3.7.1.4.1.8.1	Acoustic Levels					X					X					X				X		
3.7.1.4.1.8.2	Sinusoidal Levels					X					X					X						
3.7.1.4.1.8.3	Random Levels					X					X					X						
3.7.1.4.1.9	Flight Vehicle Mission Phases					X										X				X		
3.7.1.4.1.9.1	Ground Phase					X										X						
3.7.1.4.1.9.2	Pre Launch and Erection Phases					X										X						
3.7.1.4.1.9.3	Launch Release					X										X						
3.7.1.4.1.9.4	Powered Flight					X										X						
3.7.1.4.1.9.5	Orbit Phase					X								X		X						
3.7.1.4.2	Functions					X								X		X				X		X
3.7.1.4.3	Configuration			X																		
3.7.1.4.4	Performance Requirements	X																				
3.7.1.4.4.1	Spacecraft Core Structure			X																		
3.7.1.4.4.2	Subsystem Modules			X																		
3.7.1.4.4.3	Orbit Adjust Reaction Control System			X																		
3.7.1.5	Thermal Subsystem					X					X				X					X		
3.7.1.5.1	General Requirements	X																				
3.7.1.5.1.1	Equipment and Structure Temperatures Operating Mode					X					X			X						X		
3.7.1.5.1.2	Equipment and Structure Temperatures - Survival Mode					X					X			X						X		
3.7.1.5.1.3	Control					X					X			X						X		

Table 4-6 Requirements Verification Matrix (Sheet 12 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.7.1.5.2	Subsystem Functions					X								X						X		
3.7.1.5.3	Configuration					X					X									X		
3.7.1.5.3.1	Passive Control	X																				
3.7.1.5.3.2	Active Control	X																				
3.7.1.5.4	Modes	X																				
3.7.1.5.5	Performance Requirements					X		X			X			X						X		
3.7.1.5.6	Interface Requirements					X							X	X			X			X	X	
3.7.1.5.7	Instrumentation Requirements					X							X	X			X			X	X	
3.7.1.5.8	Ground Support Equipment													X			X				X	
3.7.1.6	Orbit Adjust Reaction Control Subsystem Module					X		X		X		X		X						X		X
3.7.1.6.1	General Requirements					X		X		X				X						X		X
3.7.1.6.2	Operational Functions					X																
3.7.1.6.3	Configuration			X																		
3.7.1.6.4	OA/RCS Modes					X		X		X		X		X						X		X
3.7.1.6.4.1	Nominal Mode					X		X		X		X		X						X		X
3.7.1.6.4.2	Off Nominal Mode							X		X		X		X						X		X
3.7.1.6.4.3	Survival Mode							X		X		X		X						X		X
3.7.1.6.5	Performance Requirements																					
3.7.1.6.5.1	Impulse							X		X		X		X								
3.7.1.6.5.2	Propellants and Pressurant			X																		
3.7.1.6.5.3	Operating Pressure			X		X																
3.7.1.6.5.4	Leakage					X						X		X								
3.7.1.6.5.5	Equipment Performance Requirements					X		X				X		X		X				X		X
3.7.1.6.5.5.1	Thrusters											X		X						X		X
3.7.1.6.5.5.2	Propellant Tanks					X						X										
3.7.1.6.5.3	Isolation Valves					X						X		X		X						
3.7.1.6.5.4	Reflief Valve and Burst Disk Assembly					X		X				X										
3.7.1.6.5.5	Vent Valves					X		X				X										
3.7.1.6.5.6	Heaters					X							X	X						X	X	
3.7.1.6.6	Physical Requirements	X																				
3.7.1.6.6.1	Mass Properties											X		X								
3.7.1.6.6.2	Dimensional and Volume Limitations		X																			
3.7.1.6.6.3	Plume Impingement					X																
3.7.1.6.6.4	Proof and Burst Pressure Factors							X		X		X		X								

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Table 4-6 Requirements Verification Matrix (Sheet 13 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.7.1.6.6.5	Cleanliness		X																			
3.7.1.6.7	Interface Requirements	X																				
3.7.1.6.7.1	Electrical Interfaces			X		X							X	X			X			X		X
3.7.1.6.7.1.1	Connectors			X		X							X	X			X			X		X
3.7.1.6.7.1.2	Harness		X	X		X								X								
3.7.1.6.7.1.3	Power					X											X			X		X
3.7.1.6.7.2	Command and Data Handling Interface					X							X	X			X			X		X
3.7.1.6.7.2.1	Telemetry					X							X	X			X			X		X
3.7.1.6.7.2.2	Commands					X											X			X		X
3.7.1.6.7.3	Mechanical Interface					X										X	X					
3.7.1.6.7.4	Thermal Interfaces					X							X	X			X			X		
3.7.1.6.7.5	Attitude Control Subsystem Interface					X											X			X		X
3.7.1.6.7.6	Subsystem/Ground					X							X	X						X		X
3.7.1.6.7.6	Subsystem Equipment Servicing Equipment Interfaces					X							X	X						X		X
3.7.1.6.8	Instrumentation Requirements					X							X	X			X			X		X
3.7.1.6.9	Ground Support Equipment												X								X	
3.7.1.7	Electrical Integration	X																				
3.7.1.7.1	General Requirements	X																				
3.7.1.7.2	Functions	X																				
3.7.1.7.2.1	Power Distribution					X							X				X					
3.7.1.7.2.2	Signal Distribution					X																
3.7.1.7.3.1	Configurations					X																
3.7.1.7.3.2	Pyrotechnic/Actuator Harnesses					X								X						X		X
3.7.1.7.3.3	Instrument Harness					X																
3.7.1.7.4	Requirements	X																				
3.7.1.7.4.1	Electromagnetic Compatibility					X																
3.7.1.7.4.2	Redundancy					X							X				X				X	
3.7.1.7.4.3	Spacecraft Interface Connector					X											X					X
3.7.1.7.4.4	Harness Components		X			X							X				X					X
3.7.1.8	Observatory Software	X																				
3.7.1.8.1	Basic Software					X	X						X	X			X			X	X	X
3.7.1.8.1.1	Executive Software Module					X	X						X	X			X			X	X	X

Table 4-6 Requirements Verification Matrix (Sheet 14 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.7.1.8.1.2	Self Test Software Module						X						X	X			X			X	X	X
3.7.1.8.1.3	Program Change Software Module						X						X	X								
3.7.1.8.1.4	Command Handling Software Module						X						X	X			X			X	X	X
3.7.1.8.1.5	Mode Control Software Module						X										X			X	X	X
3.7.3.1.8	Instrument Data Handling and Wide Band Communications	X																				
3.7.3.1.8.1	Wide Band Data Handling and Compaction			X		X		X						X			X			X	X	X
3.7.3.1.8.1.1	Functions					X								X			X			X	X	X
3.7.3.1.8.1.2	Configuration			X		X																
3.7.3.1.8.1.3	Modes of Operation					X								X			X			X	X	X
3.7.3.1.8.1.4	Interface Requirements					X								X			X			X	X	X
3.7.3.1.8.1.5	Performance Requirements					X		X						X			X			X	X	X
3.7.3.1.8.2	Primary Relay (TDRS) Wide Band Communications			X		X		X						X	X		X			X	X	X
3.7.3.1.8.2.1	Functions					X								X	X		X			X	X	X
3.7.3.1.8.2.2	Configuration			X		X									X							
3.7.3.1.8.2.3	Modes of Operation					X								X			X			X	X	X
3.7.3.1.8.2.4	Performance Requirements					X		X						X	X		X			X	X	X
3.7.3.1.8.3	Primary Direct Wideband Communications			X		X		X						X	X		X			X	X	X
3.7.3.1.8.3.1	Functions					X								X			X			X	X	X
3.7.3.1.8.3.2	Configuration			X		X									X							
3.7.3.1.8.3.3	Modes of Operation					X								X			X			X	X	X
3.7.3.1.8.3.4	Performance Requirements					X		X						X	X		X			X	X	X
3.7.3.1.8.4	Local User Wideband Communications			X		X		X						X	X		X			X	X	X
3.7.3.1.8.4.1	Functions					X								X			X			X	X	X
3.7.3.1.8.4.2	Configuration			X		X										X						
3.7.3.1.8.4.3	Modes of Operation					X									X							
3.7.3.1.8.4.4	Performance Requirements					X		X						X	X		X			X	X	X
3.7.1.8.1.6	Operations Scheduling Software Module						X													X	X	X
3.7.1.8.1.7	Data Compression Software Module						X										X			X	X	X
3.7.1.8.1.8	History Software Module						X													X	X	X
3.7.1.8.1.9	Situation Assessment Software Module						X													X	X	X
3.7.1.8.1.10	Computer Dump Software Module						X						X	X						X	X	X
3.7.1.8.1.11	Stabilization Software Module						X										X			X	X	X
3.7.1.8.1.12	Position Computation Software Module						X													X	X	X

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Table 4-6 Requirements Verification Matrix (Sheet 15 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a) COMPONENT				5 (b) MODULE				5 (c) SPACECRAFT				5 (d) OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.7.3.1.4	Structure (Instrument Support)	X																				
3.7.3.1.4.1	General Requirements	X																				
3.7.3.1.4.2	Functions					X					X			X	X	X	X			X		X
3.7.3.1.4.3	Configuration			X		X																
3.7.3.1.4.4	Solar Array Installation	X																				
3.7.3.1.4.4.1	Solar Array					X		X							X					X		X
3.7.3.1.4.4.2	Storage Tie Down and Release Mechanism					X									X	X				X		
3.7.3.1.4.4.3	Deployment andLock Mechanism							X							X					X		
3.7.3.1.4.4.4	Solar Array Drive Motor Assembly					X		X							X					X		
3.7.3.1.4.5	Spacecraft to Delta Launch Vehicle Adapter					X									X					X		
3.7.3.1.5	Thermal Subsystem					X								X	X					X		
3.7.3.1.5.1	General Requirements	X																				
3.7.3.1.5.1.1	Equipment Operating Temperatures					X					X			X						X		
3.7.3.1.5.1.2	Instrument Temperature					X														X		
3.7.3.1.5.1.3	Instrument Structure Temperature					X														X		
3.7.3.1.5.1.4	Instrument Mission Peculiar Temperature					X					X			X						X		
3.7.31.15.1.5	Solar Array Temperatures					X		X		X												
3.7.3.1.5.1.6	Design Requirements					X					X			X						X		
3.7.3.1.5.1.7	Control					X					X			X						X		
3.7.3.1.5.2	Subsystem Function					X					X			X						X		
3.7.3.1.5.3	Configuration					X					X			X						X		
3.7.3.1.5.3.1	Passive Control	X																				
3.7.3.1.5.3.2	Active Control	X																				
3.7.3.1.5.4	Modes					X					X			X						X		
3.7.3.1.5.5	Performance Requirements					X					X			X						X		
3.7.3.1.5.6	Interfaces					X							X	X						X	X	X
3.7.3.1.6.7	Instrumentation Requirements					X							X	X						X	X	X
3.7.3.1.5.8	Ground Support Equipment																			X	X	X
3.7.3.1.6	Orbit Adjust/Reaction Control Subsystem					X							X	X	X							
3.7.3.1.6.1	Orbit Adjust/Reaction Control Subsystem					X		X			X		X		X							
3.7.3.1.6.2	Shuttle Resupply					X		X			X		X		X							
3.7.3.1.7	Electrical Integration	X																				
3.7.3.1.7.1	Harness	X																				
3.7.3.1.7.1	Shuttle Umbilical Provisions					X							X							X	X	X

Table 4-6 Requirements Verification Matrix (Sheet 16 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.7.3.1.7.2	Pyrotechnic Actuator Control					X								X						X		X
3.7.3.1.7.3	Solar Array Drive	X																				
3.7.3.1.7.3.1	Function					X														X	X	X
3.7.3.1.7.3.2	Description					X		X								X				X	X	X
3.7.3.1.7.3.3	Modes					X																
3.7.3.1.7.3.5	Pefformance Requirements					X		X								X				X	X	X
3.7.3.1.7.3.5	Physical Requirements					X		X								X				X	X	X
3.7.3.1.7.3.6	Interface Requirements					X														X	X	X
3.7.3.1.9	Mission Peculiar Software	X																				
3.7.3.1.9.1	Experiment Software Module					X	X							X						X	X	X
3.7.3.1.9.2	Experiment Control and Maintenance Software Modue																					
3.7.3.1.9.3	Antenna Steering Software Module					X	X							X						X	X	X
3.7.3.1.9.4	Experiment Data Software Module					X	X							X						X	X	X
3.7.3.2	Follow On Mission Driver Requirements					X																
3.7.3.2.1	Communications and Data Handling					X																
3.7.3.2.2	Electrical Power					X																
3.7.3.2.3	Attitude Control					X																
3.7.3.2.4	Structure					X																
3.7.3.2.5	Thermal					X				X												
3.7.3.2.6	RCS/Orbit Adjust/Orbit Transfer					X																
3.7.3.2.7	Instrument Data Handling and Wide Band Communications					X																

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Table 4-6 Requirements Verification Matrix (Sheet 17 of 19)

SECTION 3 PARAGRAPH No.	TITLE	N/A	1	2	3	4	5 (a) COMPONENT A	5 (a) B	5 (a) C	5 (a) D	5 (b) MODULE A	5 (b) B	5 (b) C	5 (b) D	5 (c) SPACECRAFT A	5 (c) B	5 (c) C	5 (c) D	5 (d) OBS SYSTEM A	5 (d) B	5 (d) C	5 (d) D
3.7.1.8.1.13	Subsystem Service Software Module						X										X			X	X	X
3.7.1.8.2	Adaptable Basic Software					X	X			X												
3.7.1.8.2.1	Down Link Software Module						X						X	X			X			X	X	X
3.7.1.8.2.2	Guidance Software Module						X										X			X	X	X
3.7.1.8.2.3	Sensing Software Module						X										X			X	X	X
3.7.1.8.2.4	Pre-Launch Test Software Module						X															
3.7.1.8.2.5	Pre-Maneuver Test Software Module						X						X							X	X	X
3.7.1.8.2.6	System Monitor Software Module						X										X			X	X	X
3.7.1.8.2.7	System Troubleshoot Software Module						X										X			X	X	X
3.7.2	Instrument Functional Characteristics						X															
3.7.2.1	Multi-Spectral Scanner						X															
3.7.2.2	Thematic Mapper						X															
3.7.3	Mission Peculiar Equipment						X															
3.7.3.1	Land Resources Mission A						X															
3.7.3.1.1	Communications and Data Handling						X															
3.7.3.1.1.1	Communications Group					X								X						X		X
3.7.3.1.1.1.1	Configuration Impact	X																				
3.7.3.1.1.1.2	Modes of Operation					X								X						X		X
3.7.3.1.1.1.3	Performance Requirements					X							X	X						X		X
3.7.3.1.1.2	Data Handling Group					X																
3.7.3.1.2	Electrical Power					X								X						X		X
3.7.3.1.2.1	Solar Cell Array					X		X							X					X		X
3.7.3.1.2.2	Energy Storage Capacity					X		X						X						X		X
3.7.3.1.3	Attitude Control					X																

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Table 4-6 Requirements Verification Matrix (Sheet 18 of 19)

SECTION 3 PARAGRAPH		N/A	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a) COMPONENT				5 (b) MODULE				5 (c) SPACECRAFT				5 (d) OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.7.4	Support Equipment Functional Characteristics	X																				
3.7.4.1	Electrical Equipment	X																				
3.7.4.1.1	Test and Integration Station		X	X																		
3.7.4.1.2	Break out Box Set		X																			
3.7.4.1.3	Battery Conditioner		X	X																		
3.7.4.1.4	Test Battery Set		X	X																		
3.7.4.1.5	Spacecraft Power Set and Cables		X	X																		
3.7.4.1.6	Ranging Test Assembly		X	X																		
3.7.4.1.7	Pyro Test Set		X	X																		
3.7.4.1.8	Interface Cable Set		X	X																		
3.7.4.1.9	Solar Simulator		X	X																		
3.7.4.1.10	Instrumenl Interface Simulator		X	X																		
3.7.4.1.11	Umbilical Simulator		X	X																		
3.7.4.1.12	DITMCO-Program and Cable Set		X	X																		
3.7.4.1.13	Power Module c/o Bench		X	X																		
3.7.4.1.14	C & DH Module c/o Bench		X	X																		
3.7.4.1.15	ACS Module c/o Bench		X	X																		
3.7.4.1.16	Propulsion c/o Bench (RCS)		X	X																		
3.7.4.1.17	S/C Monitor and Control		X	X																		

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Table 4-6 Requirements Verification Matrix (Sheet 19 of 19)

SECTION 3 PARAGRAPH		NA	VERIFICATION CATEGORIES																			
			1	2	3	4	5 (a)				5 (b)				5 (c)				5 (d)			
							COMPONENT				MODULE				SPACECRAFT				OBS SYSTEM			
No.	TITLE						A	B	C	D	A	B	C	D	A	B	C	D	A	B	C	D
3.7.4.1.18	IMP Module C/O Bench		X	X																		
3.7.4.2	Mechanical Equipment	X																				
3.7.4.2.1	Interface Adapter Set		X	X																		
3.7.4.2.2	Hoist Bar and Sling Set		X	X																		
3.7.4.2.3	Support Dolly Vertical		X	X																		
3.7.4.2.4	Support Dolly Horizontal		X	X																		
3.7.4.2.5	Access Walk Stand		X	X																		
3.7.4.2.7	Weigh and C. G. Fixture		X	X																		
3.7.4.2.8	Mass Simulator Set		X	X																		
3.7.4.2.9	Support Dolly Module		X	X																		
3.7.4.2.10	Shipping Containers (Modules)		X	X																		
3.7.4.2.11	Observatory Cover Set		X	X																		
3.7.4.2.12	Humidity Control Kit		X	X																		
3.7.4.2.13	Shipping Container Solar Array		X	X																		
3.7.4.2.14	Solar Array Installation and Deployment Fixture		X	X																		
3.7.4.2.15	Transporter		X	X																		
3.7.4.2.16	Indicating Accelerometer Kit		X	X																		
3.7.4.2.17	Pyro Installation Tool Kit		X	X																		
3.7.4.2.18	Storage Motor Installation Kit		X	X																		
3.7.4.2.20	Tie Down Kit		X	X																		
3.7.4.1.21	Battery Shipping Container		X	X																		
3.7.4.1.22	Battery Installation Tool		X	X																		
3.7.4.1.3	Fluid Equipment	X																				
3.7.4.1.3.1	GN_2 Conditioning Unit		X	X																		
3.7.4.1.3.2	GN_2 Regulator Unit		X	X																		
3.7.4.1.3.3	Volumetric Leak Detector		X	X																		
3.7.4.1.3.4	R.C.S. Vacuum Test Cart		X	X																		
3.7.4.1.3.5	Fluid Distribution System		X	X																		
3.7.4.1.3.6	Pressure Maintenance Unit		X	X																		
3.7.4.1.3.7	GN_2 Storage System (Transporter)		X	X																		
3.7.4.1.3.8	GN_2 Manifold and Supply Platform		X	X																		
3.7.4.1.3.9	Fluid Distribution System - Launch Site		X	X																		
3.7.4.1.3.10	Propellant Transfer Assembly		X	X																		
3.7.4.1.3.11	Mass Spectrometer Leak Detector		X	X																		
3.8	Precedence	X																				

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PREPARATION
FOR-DELIVERY

5 - PREPARATION FOR DELIVERY

5.1 OBSERVATORY

The Observatory shall be packed in a container suitable for transport and temporary storage, in accordance with the provisions of NASA Handbook NHB 600.1 (IA) Edition. The container shall secure the EOS and prevent mechanical damage during handling and transportation. To prevent contamination and maintain an appropriate cleanliness level, a constant blanket pressure of gaseous nitrogen shall be provided within the container.

5.2 SUPPORT EQUIPMENT

Other EOS elements, such as GSE, shall be packaged for delivery to conform to Level C, MIL-STD-794B, March 1969.

5.3 MARKING

Marking for shipping and storage shall be in accordance with MIL-STD-129.

APPENDIXES

Appendix A

OPTIONAL USE OF TAPE RECORDERS

A.1 INTRODUCTION

An option in lieu of the TDRS relay of data acquired while the Observatory is not in view of primary or local user ground stations is to tape record this data and read-out later, when the Observatory is in view. This option requires three recorders: one Wideband Video Tape Recorder (WBVTR) for the TM instrument output, and two ERTS-type recorders, one each for the MSS and CTM data.

A.2 WIDEBAND VIDEO TAPE RECORDER (WBVTR)

The Observatory spaceborne wideband video tape recorder shall meet the minimum requirements specified below.

A.2.1 FUNCTION

The WBVTR shall act as a data storing and transfer mechanism between the thematic mapper (TM) and the transmitter when commanded to do so.

(a) The WBVTR shall be capable of recording and reproducing data rates of 100 Mbps for periods of up to 15 minutes - a total capacity of 9×10^{10} bits.

(b) The bit error rate (BER) at the output of the WBVTR shall be $\leq 10^{-6}$.

(c) The WBVTR shall operate, as herein specified, for two years in orbit or for no less than 25,000 full length tape passes.

A.2.2 CONFIGURATION AND OPERATION

(a) The WBVTR shall be configured as a multitrack reel-to-reel type. Inter-reel tape tension shall be maintained by negator springs. Designs using clutches or solenoids shall not be used. All rotating assemblies shall utilize redundant bearings.

(b) All heat producing devices shall be packaged separately from the record and playband electronics.

(c) The tape transport shall be housed in a pressurized container. The fill gas shall be clean air with 10 percent helium for leak detection. The initial pressure and the leak rate shall be such that the pressurization at the end of the specified life shall not be less than standard atmospheric pressure.

(d) The geometry and surface hardness of the tape heads and the materials used therein shall comply with the GSFC Specification S-715-P-14.

(e) The tape shall be high quality instrumentation grade. It shall meet the guidelines and acceptance levels cited in GSFC Specification S-715-P-14 for:

(1) Thermal stability

(2) Lubrication content

(3) Resistivity

(4) Chlorine content

(5) Oxide dispersion

(6) Flexibility.

(f) The capstan motor shall be brushless and contain an internal tachometer.

(g) The servo system shall use either the internal tachometer, or a prerecorded signal on a tape track as a reference, and shall switch reliably from one to the other.

(h) The WBVTR shall include diagnostic instrumentation for measuring a variety of functions, and converting the measurements to voltage and impedance levels suitable for interfacing with the Spacecraft telemetry system. The function shall include but not be limited to the following:

(1) Motor speed

(2) Motor direction

(3) Motor current

(4) Motor voltage

(5) Tape reel speeds

(6) Tape tension

(7) Pre-amplifier signal level (all tracks)

(8) End of tape sensor outputs

(9) Temperature

(10) Container pressure

(11) Command status flags.

(i) The WBVTR shall accept a number of commands from spacecraft telemetry. The commands shall include but not be limited to the following:

(1) Power on

(2) Power off

(3) Tape speed control

(4) Tape direction control

(5) Record, playback, erase (each track)

(6) Stop

(7) Use internal tachometer for servo

(8) Use tape tachometer signal for servo

(9) Diagnostic telemetry on

(10) Diagnostic telemetry off.

(j) The tape speed shall not exceed 100 ips during record or reproduce.

(k) The WBVTR shall reach operating speed within two seconds.

A.2.3 KEY PARAMETERS, VALUES, AND TOLERANCES

A.2.3.1 ELECTRICAL

(a) The WBVTR shall accept and output serial NRZ-L data at a 100 Mbps rate, and a coherent clock with a frequency of one full cycle per bit period.

(b) All variation in the output data due to time base error (TBE), including wow and flutter, shall not exceed 0.01 percent deviation from the nominal value.

(c) The power requirements of the WBVTR shall not exceed the following ratings:

(1) 205 W during record (peak)

(2) 270 W during reproduce (peak)

(3) 60 W orbital average.

(d) The WBVTR must operate within specification with noise or ripple on the primary power bus of as much as 0.25 volts peak-to-peak.

(e) The peak-to-peak value of ripple or noise voltage feedback by the WBVTR to the primary power bus shall not exceed 25 millivolts, nor shall the noise or ripple current feedback exceed 10 percent of the steady-state current.

A.2.3.2 PHYSICAL

(a) The total weight of the WBVTR shall not exceed 200 pounds.

(b) The total size of the WBVTR shall not exceed 5.3 $feet^3$.

A.3 MEDIUM RATE RECORDERS

Two tape recorders of the ERTS-type will also be required. One will be used for the MSS data, the other for CTM. These recorders will be in accordance with NASA GSFC Specification S-731-P-79, except for the following changes:

(a) Bit error rate $\leq 10^{-6}$

(b) Data rate(s) (selectable): 16 Mbps
20 Mbps

(c) Capacity: 15 minutes of data.

Appendix B

LOCAL USER OPTIONAL WIDEBAND COMMUNICATIONS SYSTEM

The local user optional wideband communications subsystem shall satisfy the requirements for transmitting wideband experiment data to selected ground stations. It shall be compatible with the operational requirements defined in TBD document.

B.1 FUNCTIONS

The local user optional wideband communications subsystem shall:

(a) Provide transmission of one wideband data channel to the ground.

(b) Provide telemetry points for monitoring of critical functions.

(c) Provide command capability for controlling subsystem modes of operation.

B.2 CONFIGURATION

The major components of the local user optional wideband communications subsystem shall be:

(a) Modulator/Exciter

(b) Ku-Band RF Power Amplifier

(c) Directional Antenna.

The local user optional wideband communications subsystem shall be configured as shown in Fig. B.2-1.

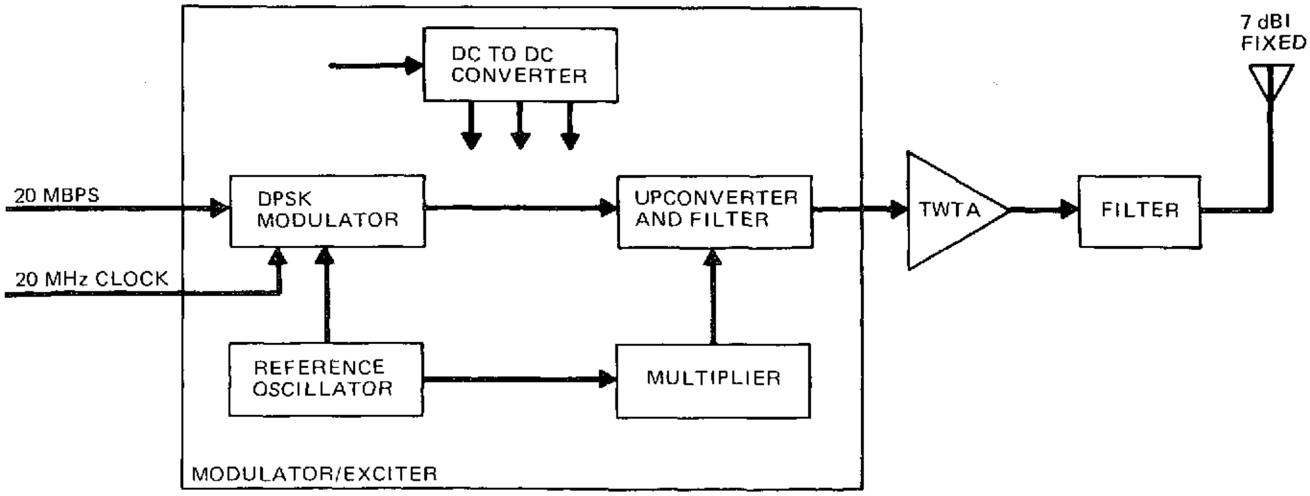

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Fig. B.2-1 Block Diagram of the Local User Optional Wideband Communications Subsystem

B.3 MODES OF OPERATION

The local user optional wideband communications subsystem modes of operation shall be selected by execution of ground or stored commands:

(a) Data Transmission.

B.4 PERFORMANCE REQUIREMENTS

B.4.1 LINK CONSIDERATIONS

The local user optional wideband communications link shall transmit DPSK encoded data to the ground with a 3 dB system margin above the signal level required for a 10^{-5} bit error rate under the following conditions:

(a) Frequency: TBD MHz in the 14.5 to 15.35 GHz

(b) Ground G/T: 23 dB/^{o}K

(c) Minimum Elevation Angle of Ground Antenna: 50^{o}

(d) Atmosphere loss: 5.0 dB

(e) Polarization: RHCP

(f) Data Rate: 20 Mbps

(g) Modulation: DPSK

(h) E/No Required: 12 dB for Pe=10^{-5}

(i) EOS E/RR Required: 20 dBW.

B.4.2 LOW GAIN FIXED ANTENNA

The low gain fixed antenna shall be installed in a location and possess a pattern as to illuminate local users within 500 Km of nadir. The antenna pattern shall be shaped so as to yield a constant signal flux density within the specified ground coverage area. Such shaping should not be carried to such an extent that users close to nadir actually receive a weaker signal than those further out, nor should it achieve uniformity at the expense of minimum gain level.

In addition, the following design requirements shall be provided:

(a) Frequency: 14.5 to 15.4 GHz

(b) Coverage: Conical sector of 70 degrees with a nominal antenna gain of 7 dBi.

(c) Gain Reference: Right hand circularly polarized isotropic radiator

(d) Polarization: RHCP.

B.4.3 MODULATOR/DRIVER

(a) Frequency: The output frequency shall be fixed in the 14.5 to 15.35 GHz frequency range. The frequency shall remain within ± 0.000 TBD percent of the assigned frequency and shall include tolerance, stability and environmental effects.

(b) RF Power Output: Minimum power output under worst-case specified environment and at 24 VDC input voltage shall be 30 milliwatts minimum. Rated power shall be provided with a load of 50 ohms at a maximum VSWR of 1.8: 1 at any phase angle. No damage to the modulator/driver shall occur if the load is open or shorted.

(c) Modulator Type: Differential PSK modulation shall be employed.

B.4.4 TWT AMPLIFIER

(a) Frequency Range: The operational frequency range of the TWTA shall be 14.5 to 15.35 GHz.

(b) Power Output: The TWTA output power shall not degrade below a minimum of TBD dBW under all orbital operations.

(c) Power Output Variation with Frequency: With a constant level swept input signal equal to that level required to produce saturation of the TWTA at mid-band, the maximum RF power output variation over the 14.5 to 15.35 GHz frequency range shall not exceed + 0.2 dB.

(d) Gain: The saturated gain of the TWTA with a constant signal level input (which produces saturated power output) at the center of the frequency range (14.925 GHz) shall not be less than TBD dB nor more than TBD dB.

B.4.5 BANDPASS FILTER

A bandpass filter shall be used in the output transmission line of the TWTA.

(a) VSWR: 1.5: 1 maximum

(b) Insertion loss: 1.0 dB maximum at center frequency

(c) Bandwideth: The minimum 3 dB bandwidth shall be 80 MHz

(d) Passband: TBD.

B.5 TELEMETRY MONITORING POINTS

The local user optional wideband communications subsystems shall include diagnostic instrumentations for increasing a variety of functions and converting the measurements to voltage and impedance levels suitable for interfacing with the Spacecraft telemetry system. The functions shall include, but not be limited to, the following:

(a) Modulator/Exciter

(1) Output level

(2) Module temperature

(b) TWTA

(1) Helix current

(2) Cathode current

(3) Converter reference volts

(4) Converter temperature

(5) Collector temperature.

B.6 PHYSCIAL

The wideband communication 20 Mbps subsystem major components shall be housed in a standard module as specified in Paragraph 3.7.4.6. Component physical requirements shall be specified in Specification EOS-SS-200. The weight allocation for the wideband communication 20 Mbps subsystem, which is part of the C&DH subsystem weight as specified in Paragraph 3.2.2.1.1. The maximum weights and volumes for the wideband communication 20 Mbps subsystem equipments shall be as follows:

Equipment	Weight - lb	Volume - in^3
• Modulator/Exciter	6.5	280
• TWTA	18.0	700
• Filter	0.3	25

Appendix C

PRIMARY DIRECT OPTIONAL WIDEBAND COMMUNICATIONS SUBSYSTEM

The optional primary direct wideband communications subsystem shall satisfy the requirements for transmitting wideband experiment data to the NASA ground (ETF, GDS, ULA) and the DOI ground station (Sioux Falls). It shall be compatible with the operational requirements defined in the NASA/GSFC Users Guide No. 101.1.

C.1 FUNCTIONS

The primary direct optional wideband communications subsystem shall:

(a) Provide simultaneous transmission of two wideband data channels to the ground directly.

(b) Provide telemetry points for monitoring of critical functions.

(c) Provide command capability for controlling subsystem modes of operation.

C.2 CONFIGURATION

The major components of the primary direct optional wideband communications subsystem shall be:

(a) QPSK Modulator/Exciter

(b) Ku-Band RF Power Amplifier

(c) Directional Antenna.

The direct optional wideband communications subsystem shall be configured as shown in Fig. C.2-1.

C.3 MODES OF OPERATION

The direct optional wideband communications subsystem modes of operation shall be selected by execution of ground or stored commands.

(a) Antenna Acquisition

(b) Data Transmission

(c) Antenna Selection.

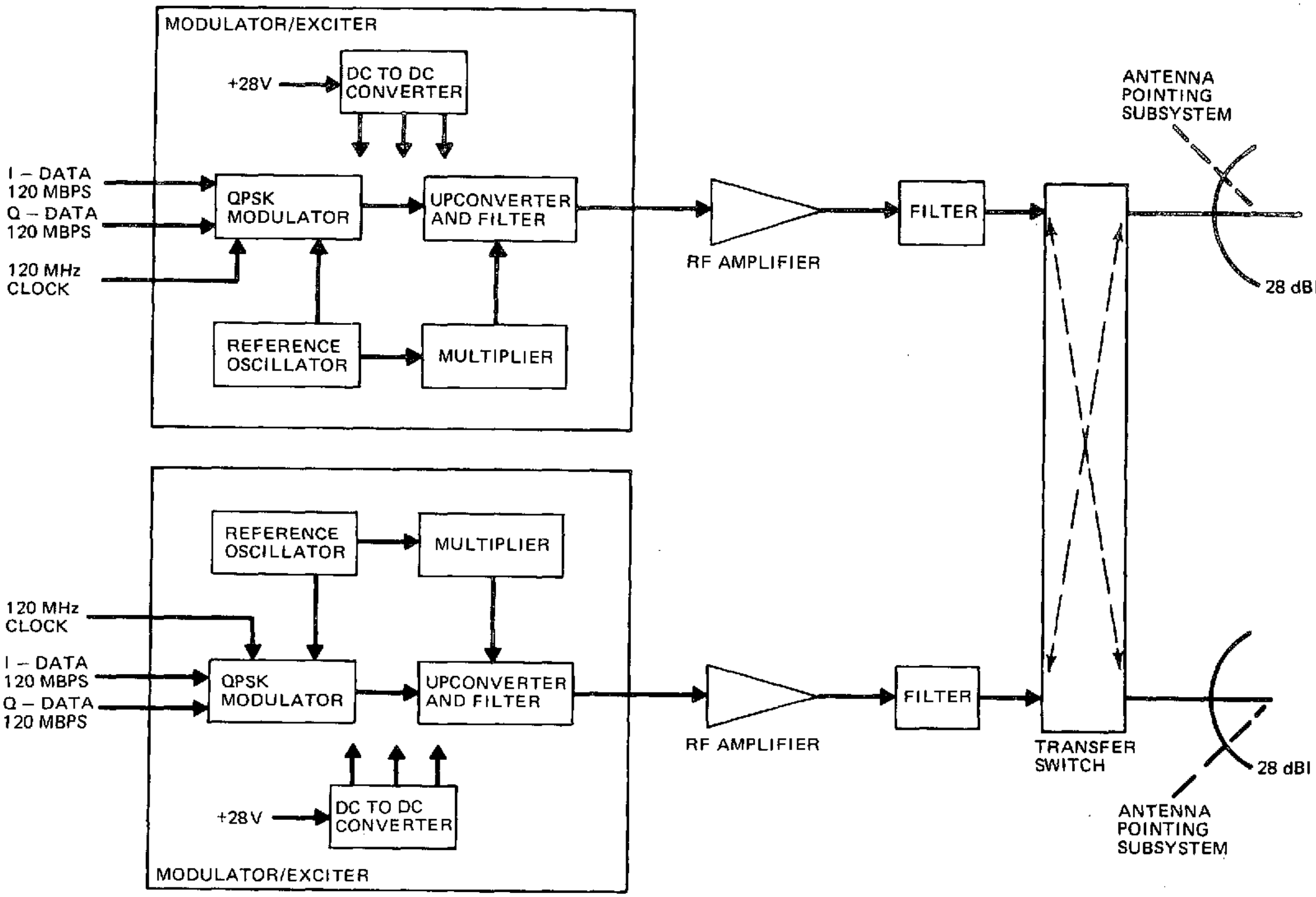

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Fig. C.2 - 1 - Block Diagram of the Primary Direct Optional Wideband Communications Subsystem

C.4.1 LINK CONSIDERATIONS

The primary direct optional wideband communications link shall transmit QPSK encoded data to the ground with a 6dB system margin above the signal level required for a 10^{-6} bit error rate under the following conditions:

(a) Frequency: TBD MHz in the 14.5 to 15.35 GHz

(b) Ground G/T: 30 dB/^{o}K

(c) Minimum Elevation Angle of Ground Antenna 2^{o}

(d) Atmosphere Loss (Rain, Cloud, Oz): 7.1 dB

(e) Polarization: RHCP

(f) Pointing Loss: 0.5 dB

(g) Data Rate: 120 Mbps/channel
240 Mbps/2 channels (Quadriphase)

(h) Modulation: QPSK

(i) E/No required: 13 dB at Pe=10^{-6}

(j) EOS EIRP Required: 36 dBW

C.4.2 HIGH GAIN DIRECTIONAL ANTENNA

The High Gain Directional Antenna shall provide the capability for pointing toward any point on the earth disc visible from the Spacecraft, upon ground command, and to continue to re-direct its position (i.e., track) as the look angles change during the pass.

The High Gain Directional Antenna, shown in Fig. C.4.2-1, shall have the following design requirements:

(a) Frequency:	14.5 to 15.4 GHz
(b) Antenna Type:	Parabolic
(c) Feed Type:	Waveguide Horn
(d) Polarization:	RHCP
(e) Axial Ratio:	1.5 dB max
(f) Side and Back lobes:	$\leq$ 17 dB
(g) Antenna Dish Size:	1.0 ft nominal
(h) Net Antenna Gain:	30 dB measured at rotary joint

(i)	Pointing Accuracy:	TBD
(j)	Gimbal Stop Size:	TBD
(k)	Slew Rate: Velocity: Acceleration:	 3 deg/sec max 5 deg/sec max
(l)	Scan Angle off Boresight, 2 Axis (XY Gimbal):	X (inner) Gimbal ± 70 degrees Y(outer) Gimbal ± 70 degrees

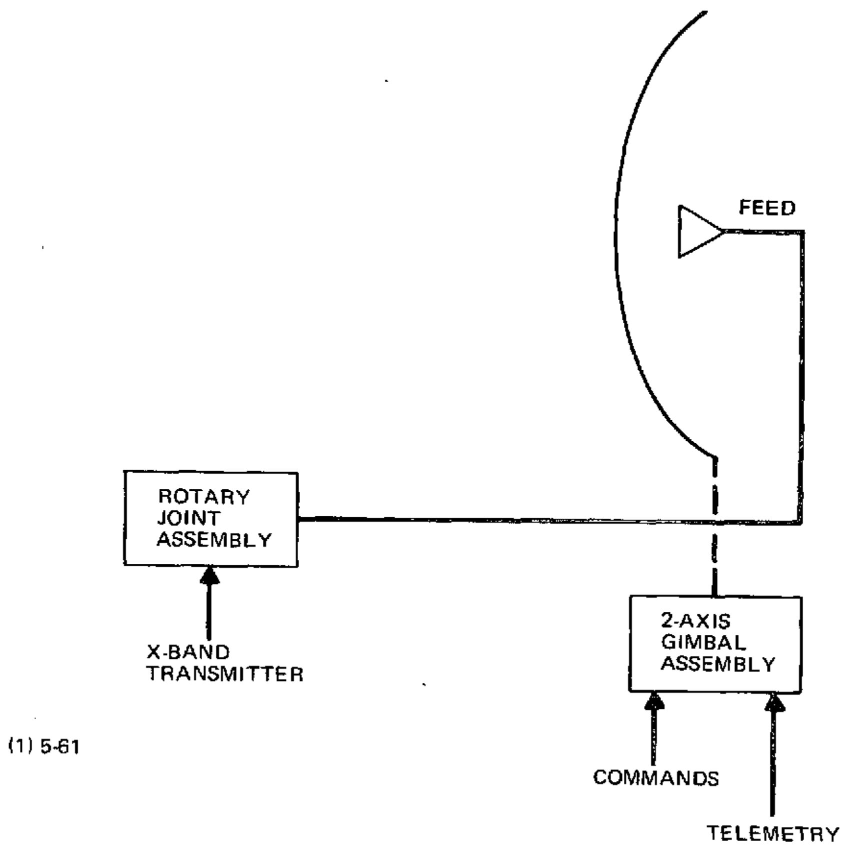

Fig. C.4.2-1 Ku-Band Steerable Antenna

C.4.3 MODULATOR/EXCITER

(a) Frequency - The output frequency shall be fixed in the 14.5 to 15.35 GHz frequency range. The frequency shall remain within ± 0.000 TBD percent of the assigned frequency and shall include tolerance stability and environmental effects.

(b) RF Power Output - Minimum power output under worst-case specified environment and at 24 VDC input voltage shall be 1 milliwatt minimum. Rated power shall be provided with a load of 50 ohms at a maximum VSWR of 1.8:1 at any phase angle. No damage to the modulator/driver shall occur if the load is open or shorted.

(c) Modulation Type - In phase/Quadrature PSK modulation shall be employed. The modulator shall be capable of accepting two 120 Mbps data streams with the capability of re-clocking the data pulses. Channel encoding shall be differential to resolve carrier phase ambiguities. Output filtering shall provide minimum overall transmission loss and detection loss at a BER of 10^{-6}.

C.4.4 RF POWER AMPLIFIER (PA)

(a) Frequency Range - The operational frequency range of the RF PA shall be 14.5 to 15.35 GHz.

(b) Power Output - The output power shall not degrade below a minimum of TBD dBW under all orbital operations.

(c) Power Output Variation with Frequency - With a constant level swept input signal equal to that level required to produce rated power output at mid-band, the maximum power output variation over the 14.5 to 15.35 GHz frequency range shall not exceed + 0.2 dB.

(d) Gain - The saturated gain of the RF PA with a constant signal level input at the center of the frequency range (14.925 GHz) shall not be less than 43 dB Nor more than 45 dB.

C-4.5 RF TRANSFER SWITCH

An RF transfer switch shall be a latching type switch with a position indicator circuit, The switch is used to connect either RF PA output to either directional antenna.

(a) VSWR - 1.15:1 maximum referenced to 50 ohms.

(b) Insertion loss - 0.2 dB maximum at $f_t \pm 70$ MHz.

(c) Isolation - 40 dB minimum between two output ports.

(d) Switch time - 1.0 sec maximum.

C.4.6 BANDPASS FILTER

A bandpass filter shall be used in the transmission line of each channel.

(a) VSWR - 1.5:1 maximum.

(b) Insertion loss - 1.0 dB maximum at center frequency.

(c) Bandwidth - The minimum 3 dB bandwidth shall be 300 MHz.

C.5 TELEMETRY MONITORING

The direct optional wideband communications subsystem shall include diagnositic instrumentation for measuring a variety of functions and converting the measurements to voltage and impedance levels suitable for interfacing with the Spacecraft telemetry system. The functions shall include but not be limited to the following:

(a) Modulator/Exciter:

(1) Output level

(2) Module temperature

(b) RF Power Amplifier:

(1) Helix, collector, and/or other DC power inputs

(2) Converter reference volts

(3) Converter temperature

(4) Tube or amplifier device temperature

(c) Switches - position.

C.5.1 PHYSICAL

The wide band communication 240 Mbps subsystem major components shall be housed in a standard module as specified in Paragraph 3.7.4.6. Component physical requirements shall be specified in Specification EOS-SS-200. The weight allocation for the wideband communication 240 Mbps subsystem, which is part of the C&DH Subsystem, is included in the overall C&DH weight as specified in Paragraph 3.2.2.1.1.

The maximum weights and volumes for the wideband communication 240 Mbps subsystem equipments shall be as follows:

Equipment	Weight - lb	Volume - in^3
• Modulator/Exciter	7.0	300
• RF Power Amplifier	8.5	325
• Filter	0.3	25
• Transfer Switch	0.8	50

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