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ACE

Advanced Composition Explorer

Spacecraft Design Specification

The Johns Hopkins University
Applied Physics Laboratory
Johns Hopkins Road, Laurel, Maryland 20723

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THE JOHNS HOPKINS UNIVERSITY
APPLIED PHYSICS LABORATORY
JOHNS HOPKINS ROAD LAUREL, MD. 20723

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1.0. GENERAL

The Advanced Composition Explorer (ACE) Observatory consists of 10 scientific instruments, one data processor which collects science from three instruments, and five spacecraft subsystems which support these instruments. ACE has three missions: a long-term primary scientific mission, a long-term secondary scientific mission and a short-term secondary mission.

The primary mission of the ACE observatory is to perform coordinated measurements of the elemental and the isotopic composition of nuclei accelerated in the solar wind, in solar flares, in interplanetary space and in galactic cosmic ray sources. This data is transmitted to the ground station during the approximately 3 hour contact time, nominally once per day. The science measurements are accomplished with the following types of instruments:

- five scientific instruments which simultaneously measure samples of the solar corona, the interplanetary medium, the local interstellar medium and galactic matter;
- one data processing unit which services three of the instruments;
- five scientific instruments which provide measurements used for interpreting the data from the other instruments.

The long-term secondary mission is to provide Near Real Time Solar Wind (RTSW) information to distributed NOAA ground receiving stations. This is accomplished by selecting data from three instruments and continuously transmitting this data to the ground, in a standard observatory format, for approximately 21 of 24 hours. No separate hardware is required for the RTSW.

The short-term secondary mission is Spacecraft Loads and Acoustic Measurement (SLAM). For SLAM, measurements are made of launch loads and the acoustic environment during the first 7 minutes of launch. SLAM is a self contained instrument package with its own battery power supply (designed for a $\leq$20 minute life), transmitter and antenna system.

ACE observatory is comprised of the APL designed and fabricated spacecraft bus, the above ten scientific instruments, the data processing unit, and the SLAM hardware. The spacecraft structure is the main load carrying structure that supports the instrument and spacecraft hardware. It is comprised of upper and a lower octagonal shaped decks and eight rectangular side decks which are all supported by a frame. All decks are constructed of aluminum honeycomb material. The observatory top deck supports most of the instrument sensors and electronics. The side panels support most of the spacecraft hardware and the CRIS and SWIMS instruments. The bottom deck supports the RF and much of the propulsion subsystem including the hydrazine fuel tanks. The observatory is connected to the Delta II 7920 launch vehicle at the lower deck with a cylindrical observatory attach fitting.

The spacecraft on-board processing is limited. Attitude is determined on the ground from the on-board star tracker data. Scientific data will be stored on two solid state data recorders. Electrical power during launch and until first ground contact is provided by a 12 A-hr battery. Electrical power during the cruise phase and on-orbit is supplied by four solar arrays attached to four of the sides of the spacecraft structure. Two opposing solar arrays support booms with magnetometer sensors mounted at the ends.

The observatory will be integrated with an 8 ft fairing and will be launched by a Delta II 7920 launch vehicle. The second stage of the launch vehicle will be used to point the observatory and to spin the observatory up to close to 5 rpm. Under ground control, the spacecraft propulsion subsystem will be used to correct the post launch attitude and maneuver the observatory into final position in orbit around the earth-sun libration point at L_1. The spacecraft star tracker and the propulsion subsystem will be used, under ground control, to determine and correct the attitude and the orbit in support of mission requirements.

The Goddard Space Flight Center will command the spacecraft via the JPL Deep Space Network (DSN) and will receive real-time housekeeping and science data as well as recorded housekeeping and science data, nominally once per 24 hours. The GSFC Sensor Data Processing Facility will provide level zero processing of the mission data.

1.1. PURPOSE OF THIS DOCUMENT

This document addresses the top level requirements for the design, fabrication, and performance of the ACE spacecraft subsystems and spacecraft structure which flow down to the spacecraft subsystems. Instrument and DSN interfaces, where these affect the spacecraft design, are also addressed. More detailed requirements for each subsystem are given in the subsystem and component specifications listed in section 2. Specifications for other aspects of the mission are also given in documents listed in Section 2.

1.2. RESPONSIBILITIES

APL is responsible for:

- design, fabrication and test of the spacecraft subsystems and structure,
- integration of the spacecraft bus, the instrument payload and SLAM,
- performance testing of the observatory from integration to launch,
- post launch support for 3 months.

The instruments, RF transponders and SLAM are GFE to the spacecraft. The instruments and the SLAM are not addressed in this document. Mission design and operation are the responsibility of GSFC and are not addressed here further than the design and test requirements passed to the spacecraft.

1.3. OBSERVATORY HARDWARE DEFINITIONS

Throughout this document, the terminology for the ACE Observatory hardware shown in Table 1.3-1 will be used. This table is based on GSFC-410-ACE-005 release, date June 1, 1992 (Figure 3-1), and has been tailored to fit the ACE flight hardware. The spacecraft and instrument components and subsystems are defined in Sections 4.0.

1.4. PHASES OF THE MISSION

The phases of the mission, as defined in this document, are:

1. Ascent Phase: from lift-off to separation from launch vehicle and autonomous solar array deployment.
2. Transfer Orbit Phase: from lift-off to first ground contact.
3. Cruise Phase: from first ground contact to L_1 point.
4. On-orbit Phase: orbit about the L_1 point.

Table 1.3-1 ACE Observatory Hardware Breakdown

Hardware	Description (from GSFC-410-ACE 005)	ACE
Observatory or System	Devices, manned or unmanned, which are designed to be placed into a suborbital trajectory, an orbit about the earth, or into a trajectory to another celestial body.	In ACE Project Office terminology, observatory is defined as the integrated spacecraft and science instrument payload.
Subsystem	The next functional subdivision of an observatory and is generally composed of two or more components.	Each instrument and each spacecraft subsystem (e. g., CRIS, ULEIS are instrument subsystems; Power, C&DH are spacecraft subsystems).
Component	A functional sub-division of a subsystem and is generally a self-contained combination of items performing a function necessary to the subsystem's operation.	Also referred to as a 'unit' in this document. Instrument examples: DPU, sensor. Spacecraft examples: battery, transponder including diplexer, parabolic dish, solar array, C&DH component, thruster, nutation damper.
Assembly	The next functional subdivision of a component and consists of parts which perform functions necessary to the operation of the component as a whole.	Modular power supply used within a component or subsystem, etc.
Subassembly	A subdivision of an assembly.	Wire harness, populated printed circuit board, etc.
Part	An element of a component, assembly or subassembly which is not normally subject to further subdivision or disassembly without destruction of the designed use.	Resistor, etc.

1.5. ORGANIZATION OF THIS DOCUMENT

This document is organized into the following 8 sections:

Section 1 General information applying to the observatory, its subsystems and components.
Section 2 Applicable documentation to this specification.
Section 3 ACE mission requirements flowdown to spacecraft subsystems.
Section 4 Observatory hardware definition for spacecraft and instrument hardware.
Section 5 Top level spacecraft description and requirements.
Section 6 Spacecraft subsystem performance requirements for each subsystem.
- 6.1 RF communications subsystem
- 6.2 Command and data handling subsystem
- 6.3 Power subsystem
- 6.4 Attitude determination and control components
- 6.5 Propulsion subsystem
- 6.6 Structure
- 6.7 Thermal

Section 7 Spacecraft Loads and Acoustic Measurement (SLAM) hardware.
Section 8 Ground Support System.

1.6. APPLICABLE DOCUMENTS AND CHANGE CONTROL

The following sections list the documents which form part of this specification for the particular hardware. This document and those listed in Figure 2.2-1 constitute a baseline as of date of issue of this specification.

The 'ACE Spacecraft Design Specification' is the baseline for the spacecraft hardware design, as of its date of issue. In case of conflict with spacecraft component and subsystem documents, the requirements of this specification shall govern unless program level exemption is granted. Conflicts with program and mission level documents that result in changes to the agreed observatory requirements shall not be implemented until negotiated with APL.

This document, after review and release, may be revised only through formal control procedures as described in the APL ACE Configuration Management Plan, 7345-9101.

2.0 APPLICABLE DOCUMENTS

The applicable portions of the following documents, to the extent indicated in this specification, will form part of this specification.

2.1. GENERAL GOVERNMENT DOCUMENTS

MIL-STD-461B	Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference
MIL-STD-462D	Measurement of Electromagnetic Interference Characteristics
NASA Pub 1124	Outgassing for Spacecraft Materials
MIL-C-45662	Calibration System Requirements
MSFC-SPEC-522A	Design Criteria for Controlling Stress Corrosion
MIL-HDBK-5D	Metallic Materials and Elements for Aerospace Vehicle Structures
MIL-STD-889	Dissimilar Materials
MIL-STD-1522	Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems
NSS/HP 1740.1	NASA Aerospace Pressure Vessel Safety Standards
MIL-B-5087B	Bonding Electrical and Lightning Protection for Aerospace Systems
MIL-STD-1541	Electromagnetic Compatibility Requirements for Space Systems
DOD-E-83578A	Explosive Ordnance for Space Vehicles General Specifications for
GEVS-SE	General Environmental Verification Specification for STS and ELV Payloads Subsystems and Components
MIL-P-55110	Printed Wiring Boards
MIL-STD-275	Printed Wiring for Electronic Equipment

2.2. JHU/APL DOCUMENTS

Figure 2.2-1 gives a top level ACE document tree for the JHU/APL documents. Subsystem and component design specification tree is given in Section 6.0. Verification documents are listed in the ACE Environmental Definition, Spacecraft and Observatory Test Requirements and Instruments Test Recommendation Document (7345-9007), referred to as APL environmental specification throughout the remainder of this document.

7345-9100	Product Assurance Implementation Plan (PAIP)
7345-9101	ACE Configuration Management Plan
7345-9102	ACE Contamination Control Plan
7345-9002	ACE Interface Control Documentation Plan
7345-9003	Telemetry and Command Handbook
7345-9004	APL Input to the ACE Observatory to Launch Vehicle Interface Document
7345-9005	ACE General Instrument Interface Specification
7345-9006	ACE Observatory Integration and Test Plan
7345-9007	ACE Environmental Definition, Spacecraft and Observatory Test Requirements and Instruments Test Recommendation Document (referred to as APL environmental specification throughout the remainder of this document)

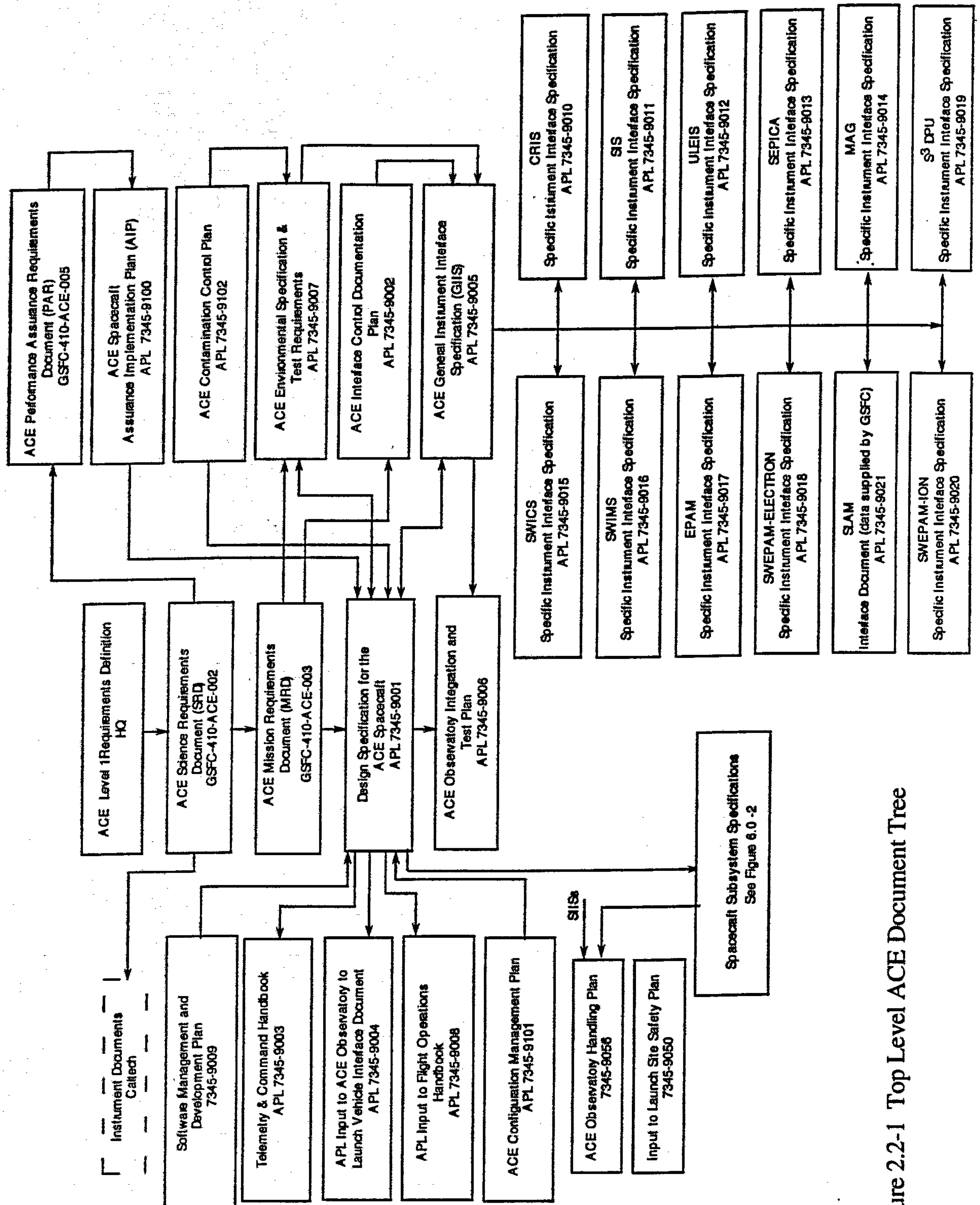

Figure 2.2-1 Top Level ACE Document Tree

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7345-9008	APL Input to Flight Operations Handbook
7345-9049	Ground Support System Requirements
7345-9050	Inputs to the Safety Plan (Launch Site)
7345-9056	ACE Observatory Handling Plan
SDO-8090	Connector Savers for Flight Hardware
SDO-7887	Protecting Flight Hardware from Electrical Damage
SDO-2387-1	Integrated Test Specification for Space Payload Equipment
7345-9009	Software Development and Management Plan
7345-9010	CRIS Specific Instrument Interface Specification
7345-9011	SIS Specific Instrument Interface Specification
7345-9012	ULEIS Specific Instrument Interface Specification
7345-9013	SEPICA Specific Instrument Interface Specification
7345-9014	MAG Specific Instrument Interface Specification
7345-9015	SWICS Specific Instrument Interface Specification
7345-9016	SWIMS Specific Instrument Interface Specification
7345-9017	EPAM Specific Instrument Interface Specification
7345-9018	SWEPAM - E Specific Instrument Interface Specification
7345-9019	S^3DPU Specific Instrument Interface Specification
7345-9020	SWEPAM - I Specific Instrument Interface Specification
7345-9021	SLAM Interface Document (data to be supplied by GSFC)
7345-9022	Sun Sensor Procurement Specification
7345-9023	Star Tracker Procurement Specification
7345-9024	Nutation Damper Specification
7345-9025	Propulsion Subsystem Specification
7345-9030	Command and Data Handling (C&DH) Component Specification
7345-9031	Power Switching Component Specification
7345-9068	Ordnance Component Specification
7345-9032	Mass Memory (SSR) Specification
7345-9033	SSR Interface Control Document
7345-9034	Telemetry Frame Format Description
7345-9035	RF Communications Subsystem Specification
7345-9036	Transponder Specification (APL Input)
7345-9037	Pre-Mod Conditioner Specification
7345-9041	High Gain Antenna Specification
7345-9042	Broad Beam Antenna Specification
7345-9027	Power Subsystem Software Requirements Specification
7345-9028	Power Processor Requirements Specification
7345-9029	Power Subsystem Software Design Specification
7345-9026	Power Electronics Specification
7345-9045	Power Subsystem Specification
7345-9046	Solar Array Procurement Specification
7345-9047	Battery Specification & Test Plan
7345-9091	Battery Cell Procurement Specification

2.3. GSFC DOCUMENTS

NASA Headquarters	ACE Level 1 Requirements Definition
GSFC-410-ACE-002	ACE Science Requirements Document (SRD)
GSFC-410-ACE-003	ACE Missions Requirements Document (MRD)
GSFC-410-ACE-011	System Safety Implementation Plan for the ACE Program
GSFC-410-ACE-013	ACE Execution Phase Project Plan

2.4. OTHER DOCUMENTS

MDC H3224B	Delta II Payload Planners Guide
ACE-CT-100-22	Environmental Design and Test Requirements for the ACE Payload Rev A Sept 12, 1994

3.0 ACE MISSION REQUIREMENTS FLOWDOWN TO SPACECRAFT

In support of the ACE mission, the following requirements are levied on the spacecraft by the various sections of the ACE Mission Requirements Document (GSFC-410-ACE-003 Rev A dated April 18, 1994, Change Notice #1 dated October 17, 1994). These requirements may also flow down to the instruments and mission design but only the spacecraft is addressed here. Some requirements may be satisfied after ground processing.

Table 3.0 -1 ACE Mission Requirements Flowdown to the Spacecraft

Requirement	MRD Section	Spacecraft Subsystem or Component Affected
Mission lifetime shall not be less than 2 years with a 5 year goal.	Mission Section 3.1.1	•All spacecraft parts shall be chosen accordingly and redundancy incorporated as appropriate. •Spacecraft consumables shall not preclude a 5 year lifetime.
The ACE mission shall be implemented as Class C with modifications commensurate with requirements, acceptable technical risks and cost.	Mission Section 3.1.2	•All spacecraft parts shall be chosen accordingly.
The ACE Observatory shall be launched from Cape Canaveral Air Force Station (CCAFS).	Mission Section 3.1.3	•EMC design and test shall consider range frequencies. •Spacecraft will follow range safety requirements.
ACE telemetry shall be compatible with CCSDS recommendations and DSN.	Mission Section 3.1.4	• Portions of RF subsystem will be tested for compatibility with DSN. •C&DH Component shall be designed for compatibility with CCSDS.
Recorded and real time data shall be dumped in accordance with a pre-scheduled spacecraft to DSN contact, nominally once per day.	Mission Section 3.1.5	•Data handling portion of C&DH component must incorporate interleaving of real time and recorded data.
The spacecraft to DSN contact periods shall be for a minimum of 3 hours, each day.	Mission Section 3.1.6	• Each recorder shall be sized to store at least 26 hours of data. • Power subsystem and thermal design shall support data recorder in playback mode.

Table 3.0 -1 ACE Mission Requirements Flowdown to the Spacecraft (cont.)

Requirement	MRD Section	Spacecraft Subsystem or Component Affected
ACE science data and related documentation shall be transferred to a public domain site such as the National Space Science Data Center (NSSDC) in accordance with the schedule specified in the ACE Project Data Management Plan.	Mission Section 3.1.7	None
The ACE Observatory shall be placed into a modified halo orbit about the L_1 libration point in a plane roughly perpendicular to the Sun-Earth line.	Mission Section 3.1.8	•On-board propulsion subsystem shall be required.
The line of sight position of the observatory shall be known with an accuracy of 10,000 km.	Mission Section 3.1.9	None
The observatory orientation shall be known (after the fact) to $\pm 0.7°$ and shall be stable to $\pm 0.5°$ during normal operations. The stability requirement does not apply during observatory maneuvers.	Mission Section 3.1.10	•Star Tracker alignment and attitude error budget. •Nutation damper needed for stability.
The ACE Observatory shall have sufficient capability and flexibility in its data recording to permit the recovery of data in case one scheduled DSN contact is missed.	Mission Section 3.1.11	• Two recorders shall be available, each with ≥26 hr data storage capability. Each recorder shall be capable of indicating EOM to C&DH component. • C&DH components shall control automatic switching to second recorder when EOM is reached (except during fault conditions).
Health/housekeeping and science data shall be recorded continuously. During the contact period, this data shall be available for near real-time monitoring.	Mission 3.1.12	•C&DH record observatory telemetry continuously. •C&DH interleave real-time telemetry and science data with recorded telemetry and science data for downlink. •Real-time telemetry and science data shall be recorded as well as downlinked.

Table 3.0 -1 ACE Mission Requirements Flowdown to the Spacecraft (cont.)

Requirement	MRD Section	Spacecraft Subsystem or Component Affected
Capability shall exist for near real-time observatory commands and responses in a single contact period.	Mission Section 3.1.13	• C&DH component design requirement. (Near real-time is interpreted to include processing time)
The spin axis of the observatory shall be pointed within 20 deg of the sun.	Mission Section 3.1.14	•Power subsystem shall provide required power. •Thermal control shall support thermal variation. • C&DH subsystem shall determine sun pulse, spin clock and phase angle from sun sensor output, except below 4° (see mission requirement 3.2.14).
Science data shall be accurate to 0.1 seconds absolute with respect to a datum, and 0.025 seconds relative to the other ACE Instruments.	Mission Section 3.1.15	• C&DH component shall use a counter based on its TCXO for data collection and recording (there is no on-board spacecraft clock). Ground operations shall correlate science data to time on the ground.
The launch vehicle shall be a Delta II 7920, two stage, ELV.	Mission Section 3.1.16	• Observatory weight shall be consistent with Delta II 7920, 8 ft fairing, 5624 PAF.
Target launch date shall be August 1997. Actual date to be no later than December 1997.	Mission Section 3.1.17	None
A minimum of 90% of the science payload data provided by the instruments must be sent to the ASC for analysis. This requirement applies to both solar active and solar quiet periods.	Mission Section 3.1.18	• C&DH shall collect 100% of the instrument output and shall store this on the recorder(s).
The spacecraft shall be capable of supporting continuous operation of all instruments.	Observatory Section 3.2.1	• Power Subsystem shall provide required power. • C&DH shall provide timing signals, sun crossing pulses, spin clock and collect housekeeping and science data from instruments.

Table 3.0 -1 ACE Mission Requirements Flowdown to the Spacecraft cont.

All deployable appendages shall be deployed in such a manner as not to remain in the primary FOV of any instrument or sensor.	Observatory Section 3.2.3	• Observatory layout and structure design, including the effect of covers, shall. accommodate requirement.
Instrument location, alignment and FOV on the spacecraft shall be required to meet all science objectives.	Observatory Section 3.2.4	• Observatory layout shall be in accordance with the instrument FOV and pointing requirements.
The ACE Observatory shall be compatible with a Delta II 7920-8 for pre-launch integration, launch, and separation of the launch vehicle from the spacecraft.	Observatory Section 3.2.5	• I&T, stress and structure designs, plans and activities shall be compatible with a Delta II 7920-8 and shall be coordinated with OLS.
The ACE Observatory shall be maintained through launch to a cleanliness level of 100,000.	Observatory Section 3.2.6	• Following TV and final electrical testing, components shall be placed in an anti-static N_2 purged bag and sealed. • Solar arrays shall be protected in a carrying container. • During observatory integration and until the observatory is in the fairing, the cleanliness level shall be maintained at a class 100,000 or better. • Low pressure nitrogen purge shall be available to instruments from observatory integration to launch.
The magnetic interference at the MAG sensor shall not jeopardize the scientific performance of the ACE mission.	Observatory Section 3.2.7	• Booms for magnetometer sensors shall be made as long as possible within design constraints. • Magnetic guidelines shall be given to all designers. • Magnetic sniffing shall be performed for all appropriate components (by the magnetometer team).

Table 3.0 -1 ACE Mission Requirements Flowdown to the Spacecraft cont.

During an uplink the spacecraft shall be capable of accepting executable commands and immediately transferring them to the appropriate instruments for execution.	Observatory Section 3.2.8	• C&DH component design requirement. ("Immediately" is interpreted to include processing time.)
The observatory shall accept a science payload data rate of 6552 bps 24 hours a day.	Observatory Section 3.2.9	• C&DH design requirement and RF link design requirement.
The Observatory weight shall not exceed 785 kg.	Observatory Section 3.2.10	• Design baseline.
The total ACE Observatory electrical power requirements shall not exceed 500 watts*. (*included RTSW)	Observatory Section 3.2.11	• The solar arrays design shall (for 20° sun angle) support 443 watts at +28 V for first 2 years (required mission lifetime) and 430 watts at +28 V for the remaining 3 years (5 years total EOL). The observatory budget is 425 watts.
The maximum bit error rate shall not exceed 10^{-6} from the instrument output on the observatory to the ground data acquisition facility. This requirement shall apply to both solar active and solar quiet times.	Observatory Section 3.2.13	• C&DH parts shall have no more than 1 error/240 years (FIFOs, 5 chips) except Actel FPGA which have $\leq 5 \times 10^{-10}$ bit error rate. • SSR shall be designed for $\leq 10^{-7}$ bit errors/26 hours. • RF Link shall be designed for $\leq 8 \times 10^{-7}$ bit error rate. • Solar active time defined as 1300 SFU max, including background.
The spacecraft shall provide a sun pulse when the spin axis of the observatory is pointed between 4 and 20 deg. of the sun.	Observatory Section 3.2.14	• C&DH component shall provide a sun pulse when the spin axis of the observatory is pointed between 4 and 20 deg. of the sun.
Instruments shall be designed to measure science data as specified in the SRD.	Science Payload 3.3.1	None
All instruments shall be capable of continuous operation.	Science Payload 3.3.2	None

Table 3.0 -1 ACE Mission Requirements Flowdown to the Spacecraft cont.

All instruments shall be capable of receiving and acknowledging commands.	Science Payload 3.3.3	None
Total power required by all science instruments shall not exceed 152 watts. (budget has been removed)	Science Payload 3.3.4	None
Total weight of all science instruments shall not exceed 149 kg. (budget has been removed)	Science Payload 3.3.5	None
GSFC facilities shall be used for flight operation, command management, data capture and attitude and orbit determination.	Ground Operations 3.4.1	None
Science data processing, storage, and management shall be performed at the ACE Science Center (ASC) and the ACE Science Analysis Remote Sites (ASARs).	Ground Operations 3.4.2	None
The MOC and the ASC shall have the capability to monitor the health and safety of the instruments and the spacecraft in near real time.	Ground Operations 3.4.3	None
Under normal operation, the MOC shall have the capability to uplink commands through the DSN during each DSN acquisition.	Observatory Section 3.4.4	None
All ground operations facilities shall be compatible with CCSDS recommendations.	Observatory Section 3.4.5	None
The angle between the observatory-Earth line and the observatory spin axis shall be maintained to provide an RF link margin that meets the 10^{-6} bit error rate requirement.	Observatory Section 3.4.6	None. (see Observatory Section 3.2.13)

4.0 OBSERVATORY HARDWARE DEFINITION

4.1. SPACECRAFT HARDWARE

The spacecraft subsystems and their associated components are defined in Table 4.1-1, shown in Figure 4.1-1 and are located on the upper (+Z), lower (-Z) and side decks as indicated.

Table 4.1-1 Spacecraft Subsystem and Component Definition and Deck Locations

Subsystem	Component Name (number)	Deck Location
Attitude Determination and Control (AD&C)	• Optical heads (2) • Optical heads (2) • Sun sensor electronics (2) • Star tracker (1) • Nutation dampers (2)	• Upper (+Z deck) • Side • Side • Side • Side
Propulsion (valves and plumbing are considered part of subsystem, not individual components)	• Thrusters (10) • Tanks (4) • Pressure transducers (2)	• Upper (4); Lower, +X (3); Lower, -X (3) • Lower (-Z deck) • Lower
Command and Data Handling Subsystem (C&DH)	• C&DH (2) • Data recorder (2) • Power switching (1) • Ordnance (1)	• Side • Side • Side • Side
RF Communications (RF)	• Transponders (2) (including diplexers) • RF switch assemblies (2) • Pre-modulation conditioners (2) • High gain parabolic antenna (1) • Broad beam antennas (4) (backfire bifilar helices)	• Lower • Lower • Lower • Lower • Upper (2), Lower (2)
Power	• Solar array (4 panels), including solar panel hardware • Shunt boxes (4) • Dissipative electronics (1) • Control Electronics (1) • Battery (1) • Analog Shunt Resistors	• Side • Side • Side • Side • Side • Side (panel on top of terminal boards)
Terminal Boards	(2)	• Side
Magnetometer Booms	(2)	• Side
SLAM	(1)	• Lower

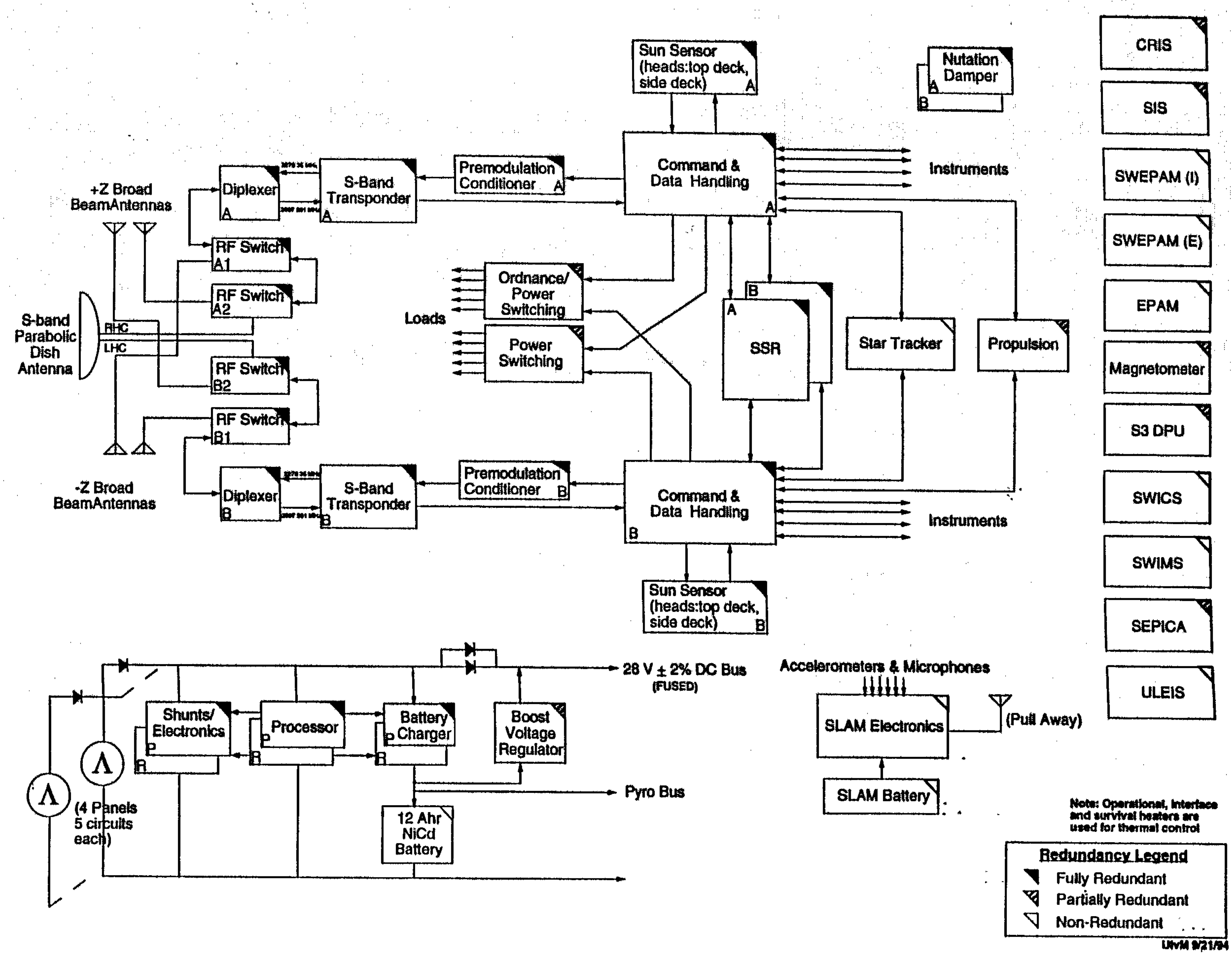

Figure 4.1-1 Observatory Block Diagram

FSCM NO.	SIZE	DWG. NO.
88898	A	JHU/APL 7345-9001 October 26, 1994 'Rev A' February 10, 1995
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4.2. INSTRUMENT HARDWARE

The instrument subsystems functions are described in Table 4.2-1. The instrument subsystems, responsible organization, associated components and deck locations are defined in Table 4.2-2. Instruments are located mainly on the upper (+Z) deck except for the CRIS and SWIMS instruments which are mounted on the side decks of the observatory.

Table 4.2-1 Instrument Subsystem Description and Function

Cosmic Ray Isotope Spectrometer (CRIS)	measures stable and long-lived isotopes of galactic cosmic ray nuclei from He to Zn over the energy range ~100 to 600 MeV/nuc.
Solar Isotope Spectrometer (SIS)	measures the elemental and isotopic composition of solar energetic particles, anomalous and galactic cosmic rays and interplanetary particles for elements from He to Zn in the energy range from ~10 to 150 MeV/nuc. It will also measure low energy He isotopes and extend measurements of galactic cosmic ray nuclei to lower energies than is possible with CRIS.
Ultra Low Energy Isotope Spectrometer (ULEIS)	measures solar and interplanetary element an isotope fluxes from He to Ni in the energy range ~0.1 to 10 MeV/nuc.
Solar Energetic Particle Ionic Charge Analyzer (SEPICA)	measures the ionic charge state, the kinetic energy and the nuclear charge of energetic solar and interplanetary ions from He to Ni in the energy range from ~0.2 to ~20 MeV/nuc.
Solar Wind Ionic Composition Spectrometer (SWICS)	determines the elemental and ionic-charge composition, the temperature and mean speeds of all major solar wind ions from H through Fe over a broad range of solar wind speeds.
Solar Wind Ionic Mass Spectrometer (SWIMS)	measures the isotopic composition of solar wind elements in the mass range from ~4 to 60 amu (He to Ni) under all solar wind conditions.
Magnetometer (MAG)	measures interplanetary magnetic field and its dynamics to provide supporting data for the primary ACE instruments.
Solar Wind Electron, Proton, and Alpha-particle Monitor (SWEPAM) (Electron (I) & Ion (I))	measures the velocity, temperature and density of solar wind electrons (from 1 to 900 eV) and ions (from 0.26 to 35 keV).
Energetic Electron, Proton, and Alpha-particle Monitor (EPAM)	measures low energy solar and interplanetary electron and ion fluxes from ~0.05 to 5 MeV over nearly all directions of the full unit sphere.
S^3DPU	serves as the single command and data interface for the SWICS, SWIMS, and SEPICA instruments and the spacecraft. Data from these instruments will be processed by the S^3DPU and supplied to the spacecraft.

Table 4.2-2 Instrument Subsystem/Component Definition and Location

Instrument Subsystem	Responsible Organization	Component	Mounting Location
CRIS	Caltech		+Y/-X Side Deck
SIS	Caltech		Upper Deck +Z
SEPICA	U of NH		Upper Deck +Z
ULEIS	APL/U of MD	• Telescope • Electronics • DPU	Upper Deck +Z Upper Deck +Z Upper Deck +Z
EPAM	APL	• Sensor 1 • Sensor 2 • Electronics • Adapter Electronics	Upper Deck +Z Upper Deck +Z Upper Deck +Z Upper Deck +Z
SWEPAM ION	LANL		Upper Deck +Z
SWEPAM ELECTRON	LANL		Upper Deck +Z
SWICS	U of MD		Upper Deck +Z
SWIMS	U of MD		-Y/+X Side Deck
MAG	U of DE	• Sensor 1 • Sensor 2 • MFI (internally redundant)	+Y Boom -Y Boom Upper Deck +Z
S^3DPU	U of NH		Upper Deck +Z

5.0 SPACECRAFT DESCRIPTION

5.1. SPACECRAFT TOP LEVEL REQUIREMENTS

5.1.1. General Requirements

To accomplish the mission described in section 1.0, the spacecraft and its subsystems shall meet the following top-level requirements:

- The spacecraft shall be compatible with the launch vehicle, both mechanically and electrically. The latter shall spin up the observatory to approximately 5 rpm and orient the observatory towards the sun.
- The spacecraft shall be capable of deploying the solar arrays and magnetometer booms, autonomously, without ground intervention.
- The AD&C subsystem shall provide sun sensor and star data to the ground for attitude processing.
- The Propulsion subsystem design shall provide capability for the ground controllers to perform attitude, spin and orbit corrections for the mission lifetime.
- The C&DH component shall receive commands from the ground and perform bit error checks. Commands shall be executed or stored for later execution.
- The C&DH component shall execute predetermined commands based on the results of instrument and spacecraft subsystem health checking (autonomy function).
- The spacecraft C&DH component shall collect the instrument bit stream and the spacecraft housekeeping data and store these on a on-board data recorder.
- The spacecraft C&DH component shall provide the real-time and recorded spacecraft housekeeping and the real-time and recorded instrument data to the RF subsystem for downlink transmission.
- The complement of data recorders shall be designed to accommodate one missed contact.
- The spacecraft RF Communications subsystem shall provide for transmitting combined real-time information and recorder playback information to the ground and receiving commands from the ground. The downlink to DSN will be initiated by ground command and shall occur nominally once per day. The transponder shall provide for a ranging function.
- The power subsystem shall provide a regulated 28v $\pm$ 2% bus for the instruments and spacecraft subsystems. A 12 A-hr battery shall provide power during launch until ground contact and shall provide power for the solar array pyro circuits.
- The observatory layout shall accommodate the instrument, sun sensor and star sensor FOV requirements.
- The thermal subsystem shall provide the required thermal environment.
- Spacecraft and Instrument mass and power shall not exceed the levels given in Table 5.1.1-1 and 5.1.1-2.

All spacecraft hardware shall meet the requirements given in 7345-9007 ACE Environmental Definition, Spacecraft and Observatory Test Requirements and Instrument Test Recommendations Document (referred to as APL environmental specification).

Table 5.1.1-1 Observatory Mass Budget

	Mass (kg)
Spacecraft & Instruments	587
SLAM	9
Fuel	189
Observatory (Wet) (max)	785
Launch Vehicle Capability	785

Table 5.1.1-2 Observatory Power Budget

	Power (watts)
Spacecraft & Instruments	425
SLAM	0
Observatory (max)	425
Solar Array Capability: (20° sun angle) - for first 2 years (required mission life) - remaining 3 years (total 5 year EOL)	 443 430

5.1.2. Mission Lifetime

The mission on orbit lifetime shall be ≥2 years (goal of 5 years). The spacecraft hardware parts, the design of the battery, and the solar cells shall not preclude a five year lifetime.

The APL environmental specification 7345-9007 gives the detailed spacecraft component design requirements for the ACE radiation environment.

5.1.3. Limited Life Items

Limited life items shall be identified. The appropriate test plans and procedures shall specify the method for tracking the item operational use.

5.1.4. Redundancy

Mission catastrophic single point failures shall be reduced through selected redundancy of the spacecraft components or redundancy within a component as permitted within mission cost constraints. Redundancy descriptions shall be given in the spacecraft subsystem/component documents.

5.1.5. Dry Nitrogen Purge

Dry nitrogen purge shall be available for all instruments (except EPAM, S^3DPU and MAG) from the time the instrument is first integrated with the spacecraft on the observatory to launch. The instrument housing shall contain a purge connector through which the dry nitrogen, at the flow rate controlled by the instrument designer, is supplied from the observatory. On the launch pad, nitrogen purge from a bottle will be supplied to the instruments through the spacecraft manifold until the fairing integration, when connection will be made to the tower purge source. The instrument designer is responsible for sealing the instrument housing/connector to provide a controlled leak as required during liftoff. (See GIIS).

5.1.6. Observatory Coordinate System

Observatory coordinate system is right hand system as shown in Figure 5.1.6-1. The observatory spins about the +Z axis and the high gain antenna points along the -Z axis towards the earth.

5.1.7. Control of Observatory

Control of the observatory attitude and orbit are the responsibility of GSFC. In addition to the requirements given in section 3, the requirements of table 5.1.7-1 apply as they pertain to spacecraft subsystems.

Table 5.1.7-1 Observatory Attitude/Orbit Required to be Maintained by Missions Operations

Parameter	Requirement on Missions Operations	Reason
Angle between the observatory -Z axis and the earth/observatory line	Shall be maintained within ±3° .	To stay within the required beam width of the high gain antenna
Angle between the sun/earth line and the observatory/earth line	Shall be maintained to be no less than 5°	The minimum angle is to limit the solar noise contribution to the receiving system noise temperature.
Spin Rate	Shall be maintained at 5±0.1 (goal ±0.05) rpm.	To meet science requirements.

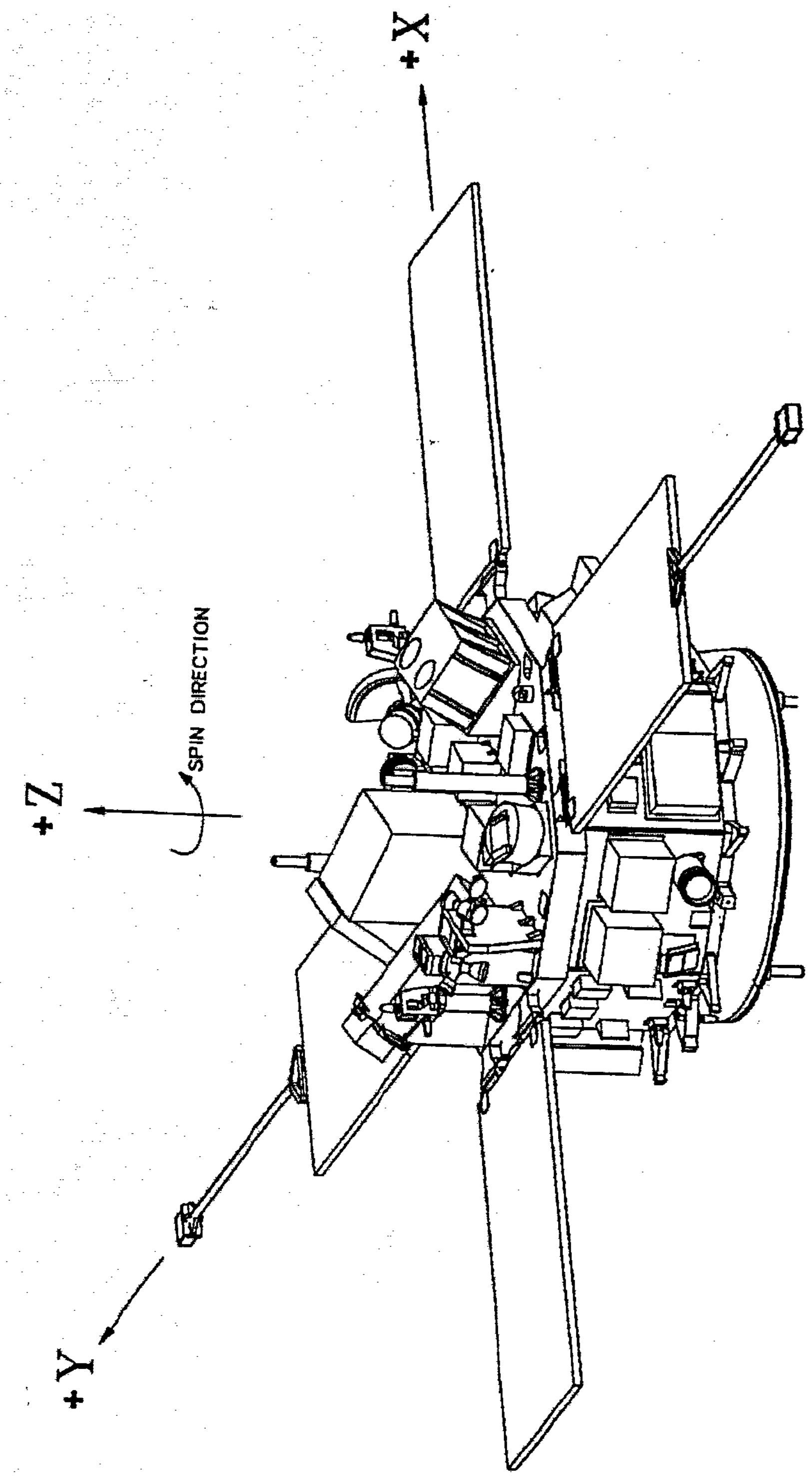


Figure 5.1.6-1 ACE Spacecraft Coordinate System

5.1.8. Observatory Instantaneous Attitude Error Budget

The spacecraft hardware shall meet the instantaneous attitude error budget given in Table 5.1.8-1.

Table 5.1.8-1 Observatory Instantaneous Attitude Error Budget

Parameter	**Angular Error (degrees)**
Star tracker accuracy	0.046 (3 sigma)
Star tracker mechanical mounting (total)	0.023
Side panel thermal distortion	negligible
Principal axis misalignment (max)	0.20
Residual nutation (max)	0.25
TOTAL (RSS)	**0.324**

5.2. OBSERVATORY ENVIRONMENTS AND VERIFICATION PROGRAM

The APL environmental specification (7345-9007) details the environments for the spacecraft components, subsystems, and the observatory for all phases of the ACE Mission and the required testing. This includes the phases from component design and test, through component/subsystem integration on the observatory, through observatory testing, through launch site preparation, to launch.

The document specifies the radiation, vibration, thermal, and electromagnetic environments for the observatory. The observatory shall be designed to meet all performance requirements under these environments and shall be tested as given in 734-9007. To minimize the observatory magnetic contamination at the magnetometer sensors, the document lists design recommendations for minimizing observatory contributions to the magnetic field at the sensors.

5.3. OBSERVATORY INTERFACE DEFINITION

The spacecraft hardware shall be functionally and physically compatible with the payload instruments, the Delta II launch vehicle, the ETR, the DSN 26 m Standard Subnet and the MOC for the phases of the mission given below:

- Pre-launch testing: The observatory shall be tested upon arrival at the launch facility, mated with the launch vehicle and then tested for pre-launch checkout as detailed in the I&T Test Plan. Replacement of instrument components at the launch site shall be minimized and such requirements shall be identified in the instrument SIIS.
- From lift off until observatory separation from launch vehicle: The launch facility will track the launch vehicle. The launch vehicle will spin up the observatory to near 5 rpm prior to separation and point the observatory.
- From observatory separation until first ground contact. The observatory shall automatically deploy the solar arrays without ground monitoring of telemetry (GSFC Program directive).
- First ground contact. The observatory will be controlled through the DSN (from the Mission Operations Center, MOC). Correction to observatory spin and attitude after separation will be performed by the GSFC ground controllers. As soon as possible after first ground contact, GSFC

will command the magnetometer booms to be deployed and will calibrate and/or verify observatory performance. Outgassing commences.

- Cruise phase. Instrument turn on sequence and instrument specific events will be determined by the mission operations.
- Orbit insertion. Insertion and corrections to the final halo orbit will be accomplished under ground control of the spacecraft propulsion subsystem.
- On-orbit. The observatory shall be controlled by the DSN (from GSFC MOC). Observatory shall be fully operational, performing within specifications.

5.3.1. Interface with Launch Vehicle

The observatory will be launched in August to December 1997 from the Eastern Test Range (ETR) launch complex 17 (LC-17) using a two-stage Delta 7920 launch vehicle. The two-stage launch vehicle has 9 light weight motors using unsegmented solid propellant for thrust augmentation, an extended first stage, and an upgraded second stage. The observatory to launch vehicle interface will be defined in APL 7345-9004 'APL Input to ACE Observatory to Launch Vehicle Interface Document'. The observatory will be mated to the launch vehicle approximately 38 days (TBR) prior to launch. A 5624 PAF will be used to attach the observatory to the launch vehicle. Separation switches located on the OAF will be used to provide separation indication to the spacecraft. Figure 5.3.1-1 shows the observatory in the launch configuration within the fairing.

5.3.1.1. Fairing for Delta II Launch Vehicle

An 8 ft fairing shall be used. The mechanical design shall provide for clearance for the static, dynamic, and thermal deflections of the observatory within the launch vehicle fairing. The launch environment inside the fairing is given in APL document 7345-9007. The fairing shall:

- Contain acoustic blanketing inside the fairing halves to reduce the acoustic level experienced by the observatory during launch and ascent.
- Provide the five standard access doors. These doors are RF transparent and are located near the broad beam antennas and so serve as RF windows. These doors are also for the observatory battery charging, umbilical and ordnance arming plug, for nitrogen purge lines to the instruments and for access to the drain/fill valves. (2 doors for fueling/defueling, 2 doors for instrument and spacecraft access, 1 door for umbilical, arming connectors and nitrogen purge lines access).
- Provide for an air-conditioning umbilical to control the temperature and humidity on the observatory. (The environmental requirements for the observatory in the fairing are given in APL environmental specification 7345-9007). The battery shall be kept between 0 and 25°C.

5.3.2. Launch

The general sequence of events from launch to spacecraft separation is as given in the Table 5.3.2-1. The launch vehicle will spin the spacecraft up to close to 5 rpm. The solar arrays will be deployed shortly after the spacecraft separates from the launch vehicle. This event shall be initiated by the C&DH separation event interface some seconds (to be defined by Flight Operations) after the separation indication is received from the separation switches. The 12 A-hr battery will support the critical loads both during launch and until some time after first ground contact is established (in case of a total solar array deployment failure). The actual time is dependent on the loads powered at first contact.

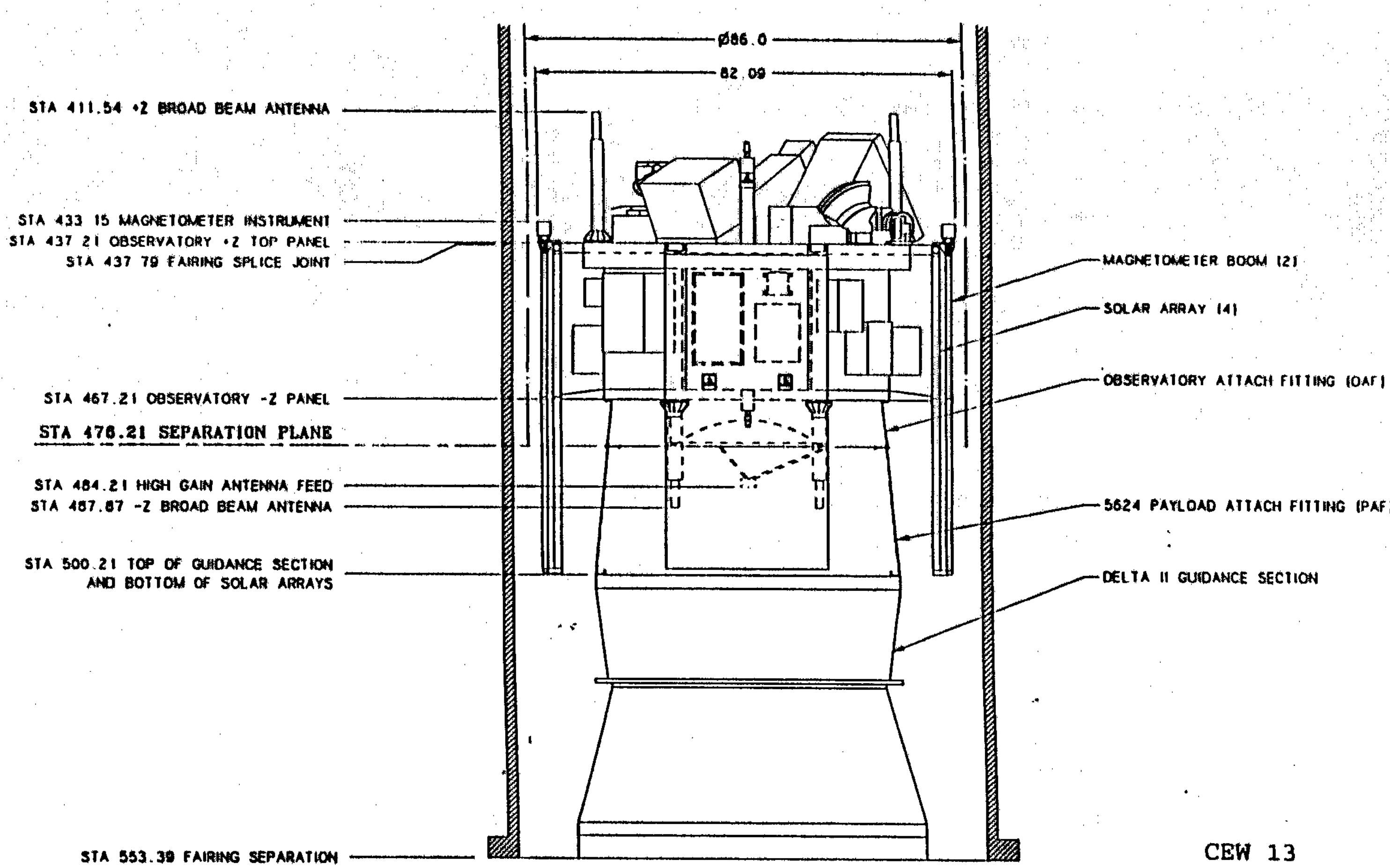


Figure 5.3.1-1 Observatory/Launch Vehicle Interface

5.3.3. Interface with DSN and MOC

After launch and first ground contact, the observatory will be controlled, commanded and tracked by GSFC/MOC through the DSN using the 26 m ground station dish. The observatory communications with the GSFC is shown in Figure 5.3.3-1. GSFC will compute orbit track, observatory attitude, spin rate and plan attitude and spin corrections and orbital maneuvers.

Compatibility between the DSN Standard Subnet and selected RF hardware shall be verified prior to observatory integration and test.

The format of the telemetry downlink shall be compatible with the CCSDS format. There is no format requirement for the command uplink although the design baseline is compatibility with COP-1.

A worst case solar radio emission of 1,300 solar flux units (including 300 SFU background) has been assumed in the design of the observatory to DSN downlink.

Figure 5.3.3-1 ACE Observatory Communication with the GSFC/MOC via the DSN

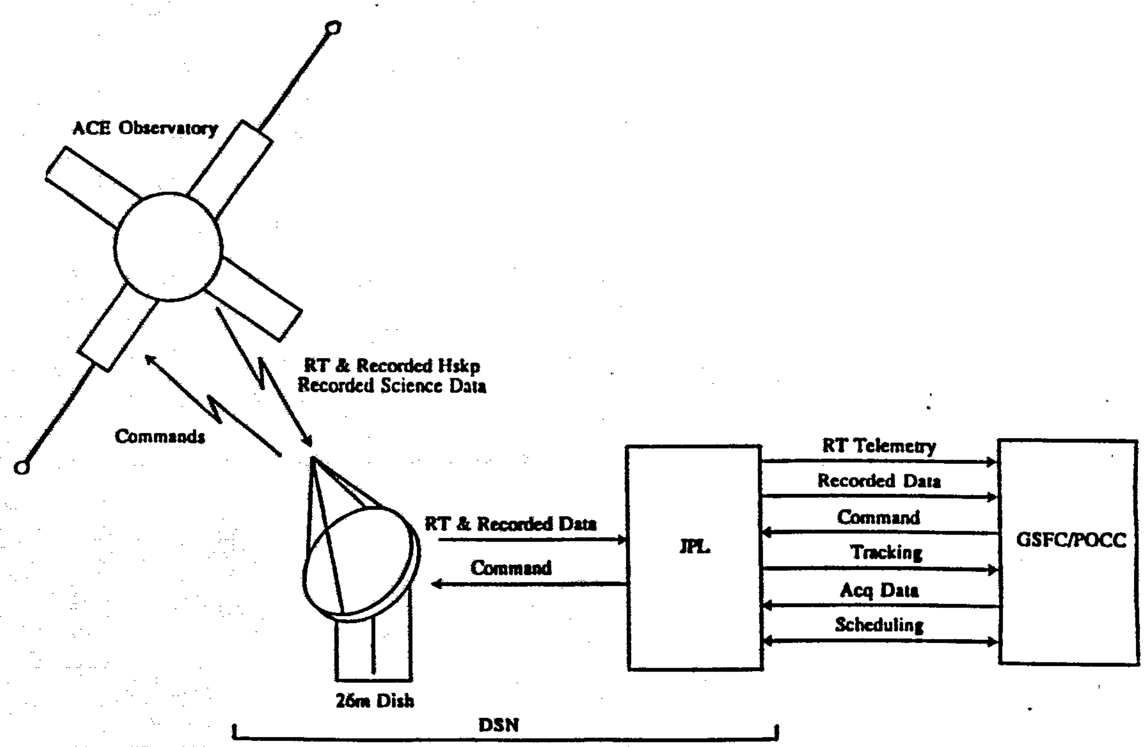


Table 5.3.2-1 General Sequence of Events from Launch to Solar Array Deployment

Event	Time (approx) sec
First Stage	
Lift off from pad	0.0
Mach 1	31.6
Maximum Dynamic Pressure	48.3
GLS Motor (6) Burnout	63.1
ALS Motor (3) Ignition	65.5
GLS Motor (3,3) Jettison	66/67
ALS Motor (3) Burnout	128.8
ALS Motor (3) Jettison	131.5
MECO	260.7
Second Stage	
Booster Separation	268.7
Second-Stage Ignition	274.2
Fairing Separation	280.0
SECO-1	453.3
Begin Coast Phase	550.0
Transfer Injection Burn	
Restart Stage 2	3403.3
SECO-2	3653.8
Maneuver to S/C Separation Attitude	3708.3
Begin Spacecraft Spin-Up	4008.3
Spacecraft Separation	4023.3
Solar Array Deployment	4023.3+TBD sec

5.3.3.1. DSN Contact

Nominally once per day, real time and recorded instrument housekeeping and science data and real time and recorded spacecraft housekeeping data will be downlinked from the observatory to the DSN. Spacecraft data storage capability shall allow for one missed contact without a loss of data. Downlinked data will be time stamped at the DSN and level zero processed (LZP) at GSFC. Ground operators will correlate the spacecraft counter to time and will calculate the actual time at which the scientific measurement were made on the spacecraft.

5.4. CONTINGENCIES

Observatory will take automatic action for the on-orbit problems given in table 5.4-1.

Table 5.4-1 On-Orbit Problems for which Observatory takes Automatic Action

Problem	Subsystem from which action is required	Implementation
Sun angle exceeds TBR deg due to ground command error or thruster malfunction.	C&DH	Autonomy feature to shut off thrusters. Can be masked by ground command.
Low voltage sense	Power, C&DH	Load shedding for the three LVS thresholds is undertaken (see also section on Power Subsystem).
No downlink signal to DSN due to loss of signal caused by solar flare or DSN problems	C&DH, Data recorder	Total storage permits one lost contact
No uplink signal	RF, C&DH	RF watchdog timer times out; C&DH changes selected antenna from high gain to broadband antennas or from one pair of broad beam antennas to the other pair (see also section on RF subsystem).

5.5. CONSTRAINTS

The ACE Flight Operations Handbook shall specify the operational constraints for the spacecraft subsystems.

5.6. Observatory Power Status for all Phases of the Mission

Table 5.6-1 gives the component power status or all phases of the mission. The hardware shall meet all requirements for the various phases of the mission, as given in this document and in APL environmental specification (7345-9007) including bus behavior as given in 7345-9007.

5.7. REQUIREMENTS ON INSTRUMENTS & SPECIFIC INSTRUMENT RELATED REQUIREMENTS ON SPACECRAFT

Instrument interface requirements are given in the ACE General Instrument Interface Specification (APL 7345-9005) and the Specific Instrument Interface Specifications for each instrument. General requirements for the instruments and for spacecraft are as follows:

- Observatory design shall accommodate, to the extent possible, all instruments such that they can be readily installed and removed without perturbations to other subsystem's performance. Instrument designs shall not preclude this and shall specify removal requirements in their SIIS, if removal after integration is necessary.
- Instrument interfaces with the spacecraft shall be designed to protect the spacecraft (and the other instruments, as applicable) in case of an instrument's failure.
- Calibration of the instruments shall be performed without affecting the operation of the other instruments or spacecraft subsystems.
- Instrument designers shall design and incorporate required dc to dc converters within their own subsystems, obtaining only regulated power from the spacecraft.
- Instrument designers shall provide for dual spacecraft interfaces to the C&DH as defined in the instrument SIIS.
- Fields of view (FOV) for the CRIS, SIS, SEPICA, ULEIS, SWICS, SWIMS, EPAM, and SWEPAM E & I instruments shall be kept clear. The required FOVs are given in Table 5.7-1.
- C&DH subsystem shall provide an unfiltered sun pulse to the instruments when the sun crosses the sun sensor's field of view (for sun angles between 4 and 20° (outside of this range, the sun pulse may be degraded) . No pseudo sun pulse, should the sun pulse be absent, is provided.
- C&DH subsystem shall provide spin clock pulses (16,384 $\pm$10 pulses per spin) to the instruments based on the above sun pulse except during maneuvers (when nutation is greater than TBD degrees).

Table 5.6-1 Component Power Status for Phases of Mission (1)

	Ascent (2)	Transfer Orbit	Cruise Phase (3)	On-Orbit (3)
AD&C Subsystem:				
Sun Sensor A & B with electronics	off	off	on	on
Star Tracker	off	off	on	on
Nutation Dampers A & B (passive)				
Propulsion Subsystem:				
Thrusters	off	off	on (4)	on (4)
Tanks				
C&DH Subsystem:				
C&DH: Command A & B Data Handling A & B	on disabled	on disabled	on enabled	on enabled
Data recorder A & B	off	off	on (5)	on (5)
Power Switching			(6)	(6)
Ordnance		(6)	(6)	(6)
RF Subsystem:				
Transponder: Transmitter A & B Receiver A & B	off on	off on	on (7) on	on (7) on
RF Switch Assemblies A & B			(6)	(6)
Pre-modulation Conditioner A & B	off	off	on (7)	on (7)
High Gain Antenna				
Broad Beam Antennas 1 to 4				
Power Subsystem:				
Solar Panels 1 to 4		active (8)	active (8)	active (8)
Shunt Boxes 1 to 4 and Analog Shunt Resistors		(6)	(6)	(6)
Dissipative Electronics	on	on	on	on
Control Electronics	on	on	on	on
Battery	discharge	(9)	charge	charge
Instruments:				
All (except magnetometer)	off	off	on (10)	on
Magnetometer	off	on	on	on
Heaters:	on(11)	on(11)	on(11)	on(11)

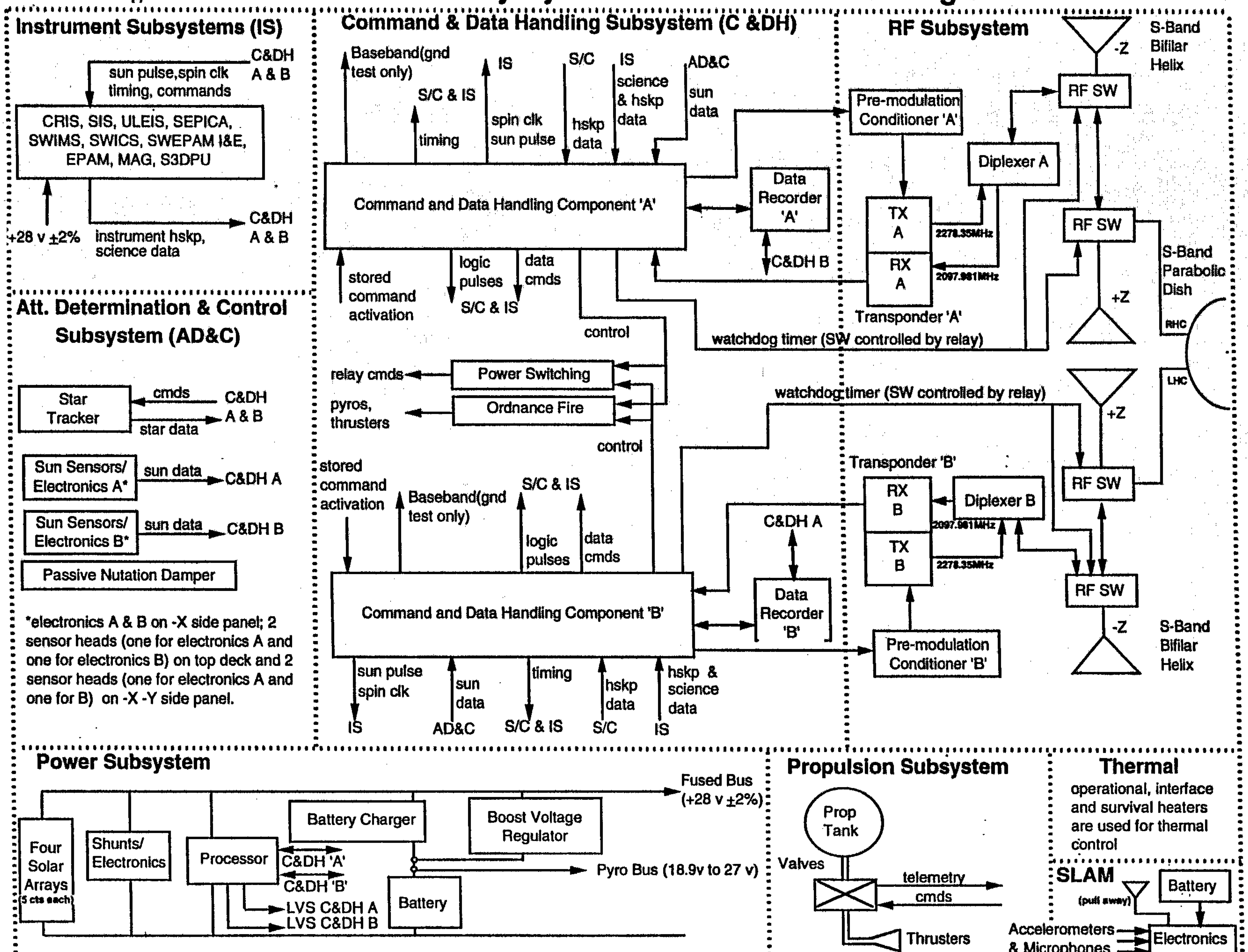
Notes for table 5.6-1: (1) See section 1.2 for definition of phases of mission.
(2) Critical Components are powered during launch.
(3) For most redundant components, only one of the components is powered. For the redundant transponders, both receivers are 'on' at all times, but only one transmitter is 'on' when required. For the redundant C&DHs, both command parts are 'on' at all times but the data handling parts of the C&DHs, although powered at all times, have only one 'enabled' when required while the other is 'disabled'.
(4) On for ground controlled maneuvers.
(5) One recorder is recording, the other recorder is in standby or playback operation.
(6) 'On' as required.
(7) On continuously.
(8) Once solar array is deployed.
(9) Discharge prior to solar array deployment and exit from eclipse, charge afterward.
(10) Turn on conditions and sequence will be determined by Mission Operations.
(11) Operational, interface and survival heaters are controlled by thermostats and will turn on when required depending on the observatory operational status (see section 6.7.2).

Table 5.7-1. Instrument Pointing Requirement

Observatory/ Instrument	Requirement (deg)	Source of Requirement (1)
Observatory	± 0.7	MRD para 3.1.10
CRIS	≤ 2	SRD para 4.2
SIS	≤ 2	SRD para 4.2
SEPICA	≤ 1	SRD para 4.2
EPAM	≤ 1	SRD para 4.2
ULEIS	≤ 1	SRD para 4.2
SWICS	≤ 1 (goal 0.5)	SRD para 4.2
SWIMS	≤ 1 (goal 0.5)	SRD para 4.2
SWEPAM Ion	≤ 1 (goal 0.5)	SRD para 4.2
SWEPAM Electron	≤ 1	SRD para 4.2
MAG	no requirement - goal 0.5	SRD para 4.2

(1) MRD - Mission Requirements Document GSFC-410-ACE-003 Rev A dated April 18, 1994, Change Notice #1 dated October 17, 1994.
SRD - Science Requirements Document GFSC-410-ACE-002 Rev A - Change Notice #1 dated April 18, 1994.

Figure 6.0-1 ACE Observatory System Functional Block Diagram



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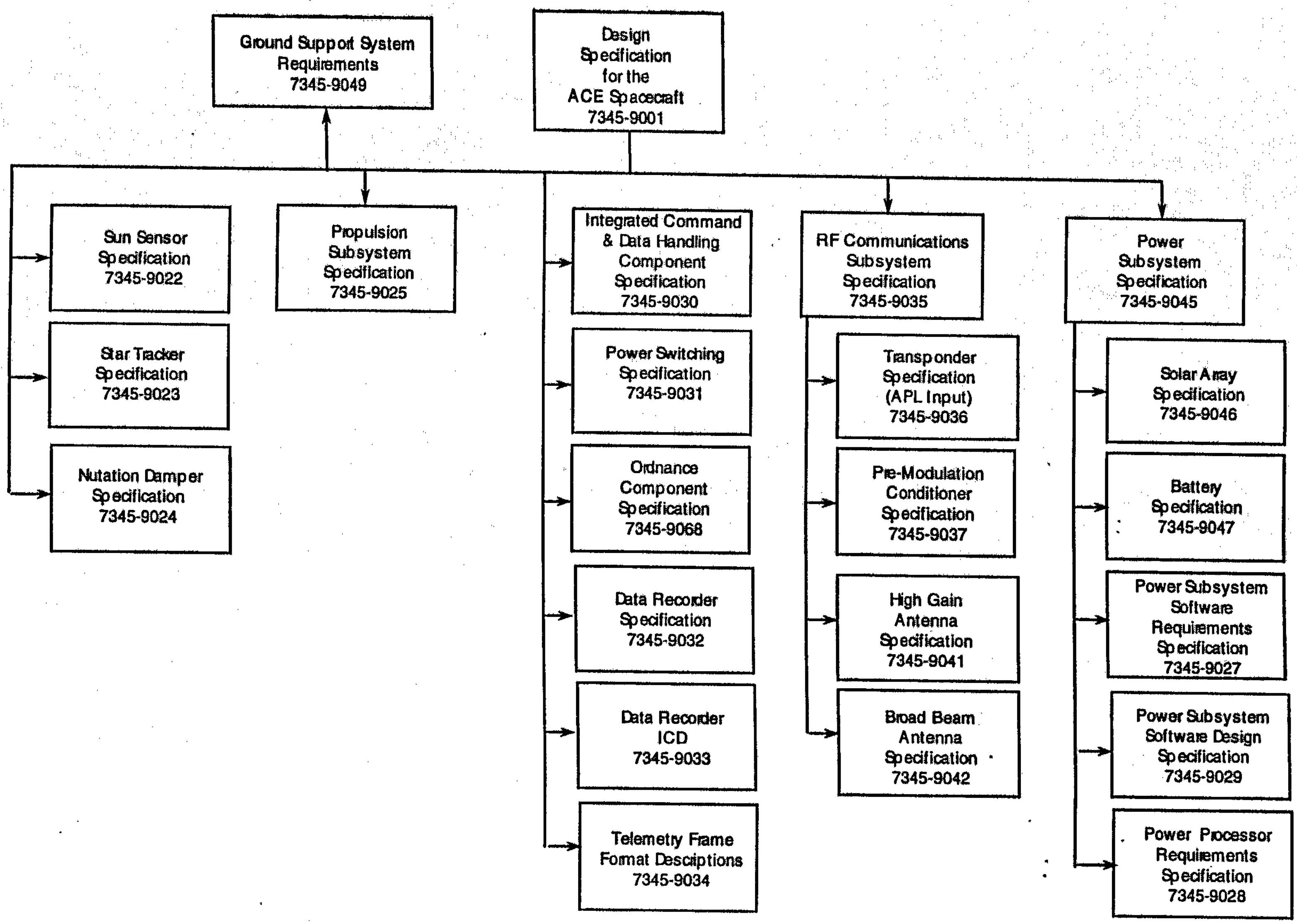

Figure 6.0-2 Specification Tree for the Spacecraft Subsystems

6.1. RF COMMUNICATIONS SUBSYSTEM

RF communication subsystem support shall begin with first ground command after launch vehicle separation and shall continue to the end of the mission. Both command receivers shall be powered throughout launch until the end of the mission. The transmitters can be turned off but one is planned to be powered continually (except during a fault or power management conditions). The link margins shall provide for a maximum spacecraft to ground station range of 1.7×10^6 km.

The RF communication subsystem block diagram is given in Figure 6.1-1. Each of the transmit/receiver chains shall be connected either to the dual polarized high gain antenna or to one pair of broad beam antennas (see also Table 6.2.3-1). The RF subsystems components shall be compatible with and interface to the C&DH subsystem for all command/control and telemetry functions. The components shall support ranging, command and telemetry functions.

The RF Communications subsystem shall be:

- Fully redundant to minimize single point failures (except for high gain antenna).
- Compatible with the 26 m DSN Standard Subnet (shall also not preclude operation with the 34 m antenna as backup).
- Comprised of the following components for each of the two independent RF communications systems (some telemetry is cross strapped at C&DH):
    - Pre-modulation conditioner
    - Transponder (including diplexer)
    - RF coaxial switch assemblies (each shall consist of two switches)
    - Parabolic antenna with dual-polarized feed.
    - 4 bi-filar helix broad beam antennas.

6.1.1. Requirements

6.1.1.1. Uplink

Throughout the mission, the subsystem shall provide for the following functions:

- Receive and demodulate uplink commands from the DSN and route these to the C&DH subsystem command portion. Data ambiguity shall be resolved by the C&DH component.
- Design for both transponder command receivers being continually powered.
- Link budget shall provide for uplink margin ≥ 6 dB.
- Design for an S-band uplink carrier at 2097.981 MHz.
- Detect modulation:
    - Subcarrier shall be bi-phase-shift-keyed modulated by command data.
    - S-band carrier shall be phase modulated by subcarrier.
- Command data rate shall be 1 kbps.
    - Command data PCM code shall be NRZ-L.
- Uplink encoding:
    - Uplink command data shall be encoded on ground to ensure that one bit transition occurs in every consecutive string of 64 bits and 125 transitions in every string of 1000 consecutive bits.

The subsystem shall also provide for an auxiliary command interface for ground testing to accept a baseband data modulated subcarrier signal and a receiver out-of-lock override signal to the transponder.

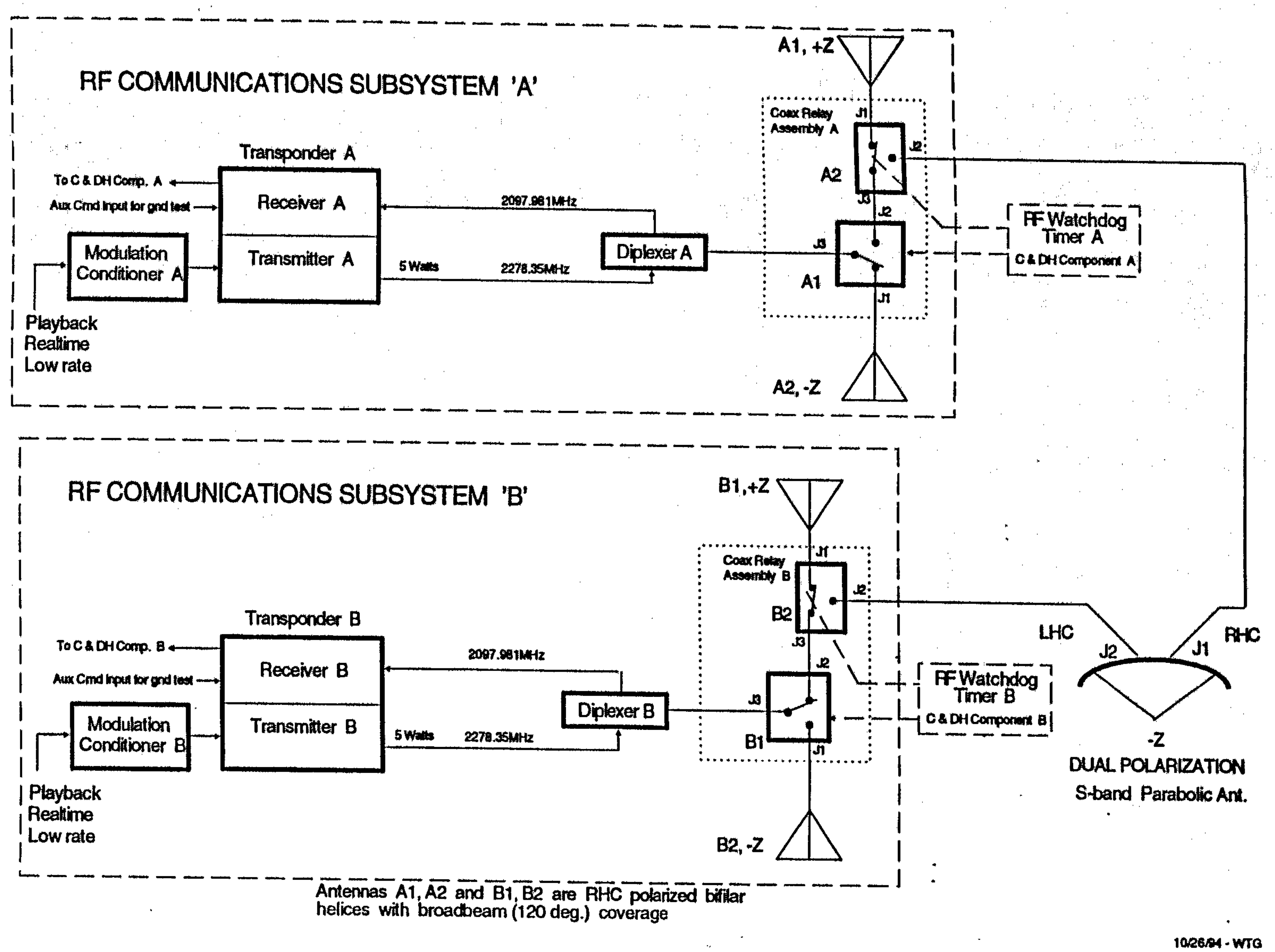

Figure 6.1-1 RF Communication Subsystem Block Diagram

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6.1.1.2. Downlink

Throughout the mission, the subsystem shall provide for the following functions:

- The active pre-modulation conditioner shall accept housekeeping and science telemetry data collected and encoded by the C&DH subsystem, and route to the active transponder where this telemetry data is modulated onto an S-Band RF signal and transmitted to the DSN ground station.
- Support data rates of 434 bps, 6,944 bps and 76,384 bps.
- Design for one pre-modulation conditioner and transponder being continually powered and transmitting to support RTSW operation.
- Support low data rate downlink via the broad beam backfire bi-filar helical antennas.
- Support high data rate operation via the parabolic antenna.
- Provide that max BER does not exceed $8x10^{-7}$ (observatory antenna to DSN antenna including path loss, for solar quiet and solar active times (SFU $\leq$1300, including background)).
- Provide a downlink margin $\geq$3 dB.
- Provide for an S-band downlink carrier at 2278.35 MHz.
- Provide modulation:
  - Encoded telemetry data from C&DH shall be phase-shift-keyed modulated on the carrier and produce a residual carrier component.
  - Modulation index shall be selectable (one of two values) by ground command and selected based on the downlink data rate.
- Downlink encoding:
  - C&DH shall encode telemetry data with a convolutional code inner code concatenated with a Reed Solomon outer code and shall convert the encoded telemetry data symbols from NRZ-L to bi-phase-L symbols.

6.1.1.3. Ranging

Throughout the mission, the subsystem shall provide for the following functions:

- Compatible with the NASA Ground Network Tone Ranging System.
- Provide for ground command to select transponder ranging mode.
- Support coherent (two-way) operation. The receiver shall detect ranging modulation and re-modulate on the downlink carrier. In the absence of an uplink, downlink carrier frequency shall be derived from the transponder internal auxiliary oscillator (one-way operation).
- Accept modulation:
  - Sine wave (primary) or square wave (alternate) signal shall directly phase modulate S-band uplink carrier.

6.1.2. RF Switches

Antennas shall be selected by controlling two RF switches for each transmit/receive chain as shown in figure 6.1-1. Each switch is a single-pole, double-throw coaxial relay. The control shall be initiated by either ground command or autonomously on-board if the RF watchdog timer times-out (see C&DH subsystem section).

6.1.3. RF Transponder/Diplexer

The transponder shall include a 5 watt transmitter, receiver, ranging function and a diplexer. The transponder with diplexer shall be GFE.

6.1.4. Pre-modulation Conditioner

The pre-modulation conditioner shall produce a stable amplitude signal to drive the transmitter linear modulator with the encoded downlink telemetry stream received from the C&DH component.

6.1.5. Antennas

Two types of antennas shall be used for communications between the ground and spacecraft; parabolic and broad beam. The parabolic antenna shall be used for the RTSW downlink.

6.1.5.1. Parabolic Dish

A 76 cm (outer diameter) high gain parabolic antenna shall be used for uplink transmissions and the 6,944 and 76,384 bps downlink transmissions. It provides LHC and RHC polarized feeds for independent connection to the two RF communications subsystems. The antenna design shall provide for sufficient gain such that the performance requirements for the RF communications subsystem are met over a cone with half-angle 4.25 degrees (min) relative to the electrical axis of the parabolic antenna. Alignment errors, between the electrical axis of the antenna and the spin axis of the spacecraft shall be no greater than 1.25 degrees (including observatory instantaneous attitude error table 5.1.8-1). Operationally the spacecraft spin axis shall be kept within 3 degrees of the spacecraft earth line.

6.1.5.2. Broad beam Antenna

Four broad beam bifilar helical antennas shall be used for uplink transmissions and the 434 bps downlink transmissions. During the cruise phase, these antennas may also be used for the high data rate downlinks, depending on the distance between the spacecraft and the ground station. Two antennas are located on the upper deck and two on the lower deck. Selection of these antennas shall be made from the ground. Each antenna shall provide wide beam width coverage over a cone with half-angle of 60 degrees (min) off the spacecraft spin axis.

6.1.5.3. Antenna Selection

Antennas shall be selected based on the configuration given in table 6.1.5.3-1 for operating during different phases of the mission. High data rate downlink transmission with the parabolic dish antenna shall be available only when the antenna pointing meets the requirements of section 6.1.5.1. The default configuration shall be selected autonomously (and is unchangeable) if the RF watchdog timer times out (see section 6.2.3 RF Watchdog Timer). All antennas support the command uplink data rate.

6.1.6. Types of RF Communications Subsystem Interfaces

Table 6.1.6-1 lists the types of interfaces between the RF subsystem and the C&DH component.

Table 6.1.5.3-1 Transponder/Antenna Configuration

	Downlink Rate	Earth Pointing Broad Beam Antennas	Parabolic Dish
Cruise Phase Operation	434 bps	x	
	6944 bps	x (depending on distance)	x (depending on distance)
	76,384 bps	x (depending on distance)	x (depending on distance)
On orbit operation	434 bps	x	
	6944 bps		x
	76,384 bps		x
Default	434 bps	x	
	6944 bps		Not Available
	76,384 bps		Not Available

Table 6.1.6-1 Types of RF Communications Subsystem Interfaces

Source	Type of Signal	Destination
C&DH Component	Logic Pulse Command	Transponder
C&DH Component	Remote Relay Command	Coax Switches
Power Switching Component	Latching Relay Contact Closure 28 V dc bus power	Transponder/Pre-modulation Conditioner (Power On/Off)
Transponder (Receiver) Lock Signals	Command data; data clock; lock signal	C&DH Component
Transponder	Status Telemetry - Analog & Digital	C&DH Component
Pre Modulation Conditioner	Status Telemetry - Analog Signal	C&DH Component
RF Coax Switches	Status Telemetry - Digital Signal	C&DH Component
C&DH Component	Encoded Downlink Telemetry Data	Pre Modulation Conditioner
Umbilical connection	Aux command enable	Transponder

6.2. COMMAND AND DATA HANDLING (C&DH) SUBSYSTEM

A block diagram of the C&DH subsystem is shown in Figure 6.2-1. The C&DH Subsystem consists of two integrated command and data handling components, one power switching component, one ordnance component (also performs some power switching functions), and two data recorders. The C&DH component and data recorders shall be redundant, the power switching component and ordnance component shall be internally redundant. C&DH software shall meet Software Management and Development Plan (7345-9009).

The C&DH subsystem shall perform the following functions:

- Command functions
  - Receive commands, verify and execute. Data ambiguity on the command data from the RF receiver shall be resolved by the C&DH component. (Data commands will be bit checked but will not be interpreted, the C&DH component performing a 'bent-pipe' function to all observatory components see table 6.2.1.2-1).
- Power switching control
  - Perform all power switching functions.
- Antenna selection
  - Select the operating antennas through ground control or autonomously when on-board RF watchdog timer times out.
- Launch Vehicle separation events
  - Perform solar array deployment (and transmitter chain turn on ..this is TBR).
- Attitude and orbit control
  - Activate thruster select and arm relays and fire FET switch in response to ground commands.
- Obtain sun angle from sun sensors and generate sun pulse and spin clock and distribute both to instruments.
- Housekeeping and science data collection
  - Collect analog and digital data from instrument and spacecraft components, time tag the data with the C&DH counter.
- Data storage and retrieval
  - Store analog and digital data on the on-board storage devices.
  - Retrieve the recorded telemetry and interleave with real time telemetry to form a composite stream for transmission to the ground via the RF subsystem
  - Record real time telemetry while previously recorded telemetry is transmitted during ground contact.
- Downlink coding
  - Code downlink with a convolutional encoded inner code concatenated with a Reed-Solomon block-oriented outer code.
- Generate CCSDS compatible downlink formats.
- Extract required science data from instrument bit stream and compose Real Time Solar Wind experiment bit stream to be continuously transmitted to ground except during ACE primary mission ground contact time.
- C&DH component shall be capable of recognizing and resetting C&DH processor in case of on-board identifiable failures within the C&DH component.

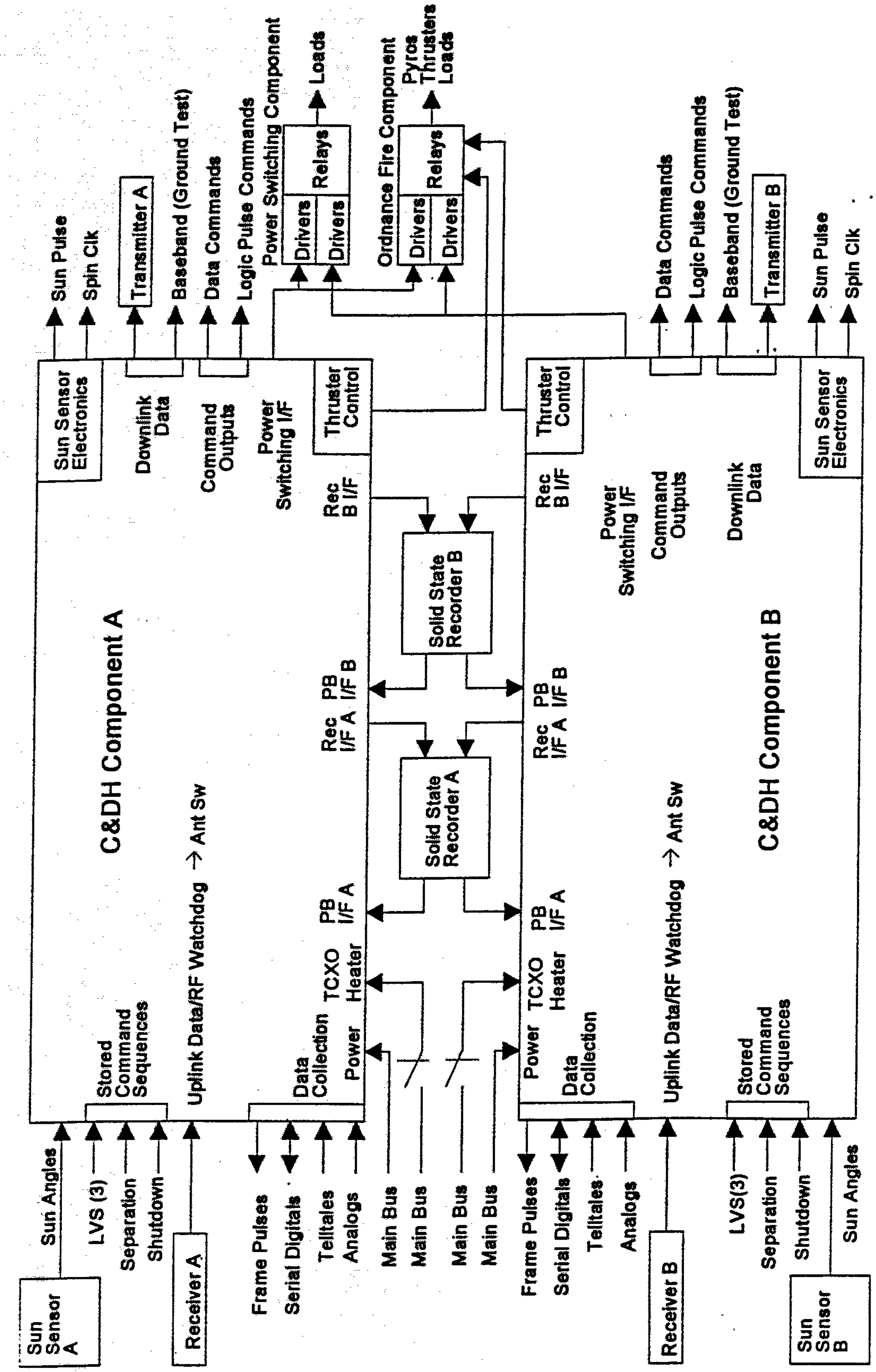

Figure 6.2-1 Command and Data Handling Subsystem Block Diagram

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6.2.1. Command Function

The uplink command data structure, shown in Figures 6.2.1-1 and -2, can contain up to 100 individual commands at a 1 kbps bit rate. A telecommand frame can contain a combination of commands that are to be executed immediately and commands that are to be stored for future execution. The command component shall support Real Time and Stored Commands as follows:

- Real Time Commands shall be executed, in the order received, as the transfer frame (see figure 6.2.1-2) is processed. These commands are not stored. The C&DH component shall be capable of intermixing previously uplinked time tagged commands, block commands and autonomy commands with real time commands.

- Stored Commands are for later execution. Four types shall be supported:

  - Time tagged commands to be stored and executed when the C&DH component's time counter matches or exceeds the stored time tag.
  - Block bin* commands for execution of multiple, stored commands based on a single event. Block mode commands shall be initiated with real-time, time tagged or autonomy commands that specify the start and end address of the block. (A pause command can be used to insert a delay between components.) Time Tagged commands are stored with a checksum which shall be verified before the command is executed.
  - Autonomy commands to perform simple limit checking on selected instrument and spacecraft component housekeeping parameters (voltages, currents, temperatures etc). An autonomy command shall be stored with a rule which contains a pointer to a specific byte of telemetry. That byte of telemetry shall be compared to a fixed value or range and if true, the pre-defined sequence of commands for out-of-range conditions associated with the rule shall be executed. Capability shall exist for autonomy rules to be true multiple times before the command is executed. Instrument housekeeping parameters that require limit checking by the C&DH component shall be in a fixed location in each minor and major frame (otherwise instrument bit stream is not examined). Autonomy commands, including the telemetry byte pointer and comparison value(s), shall be changeable and can be reloaded by ground command.
  - Fixed block bin commands to perform predefined (and unchangeable) sequence of commands for the Launch Vehicle/Spacecraft separation sequence and for the RF watchdog antenna selection sequence. LVS (section 6.2.2) and shutdown (section 6.2.7) bins are predefined and changeable.

  *Bin - a pre-determined fixed number of bytes in the C&DH component RAM memory space.

Figure 6.2.1-1 CCSDS Telecommand Data Structure

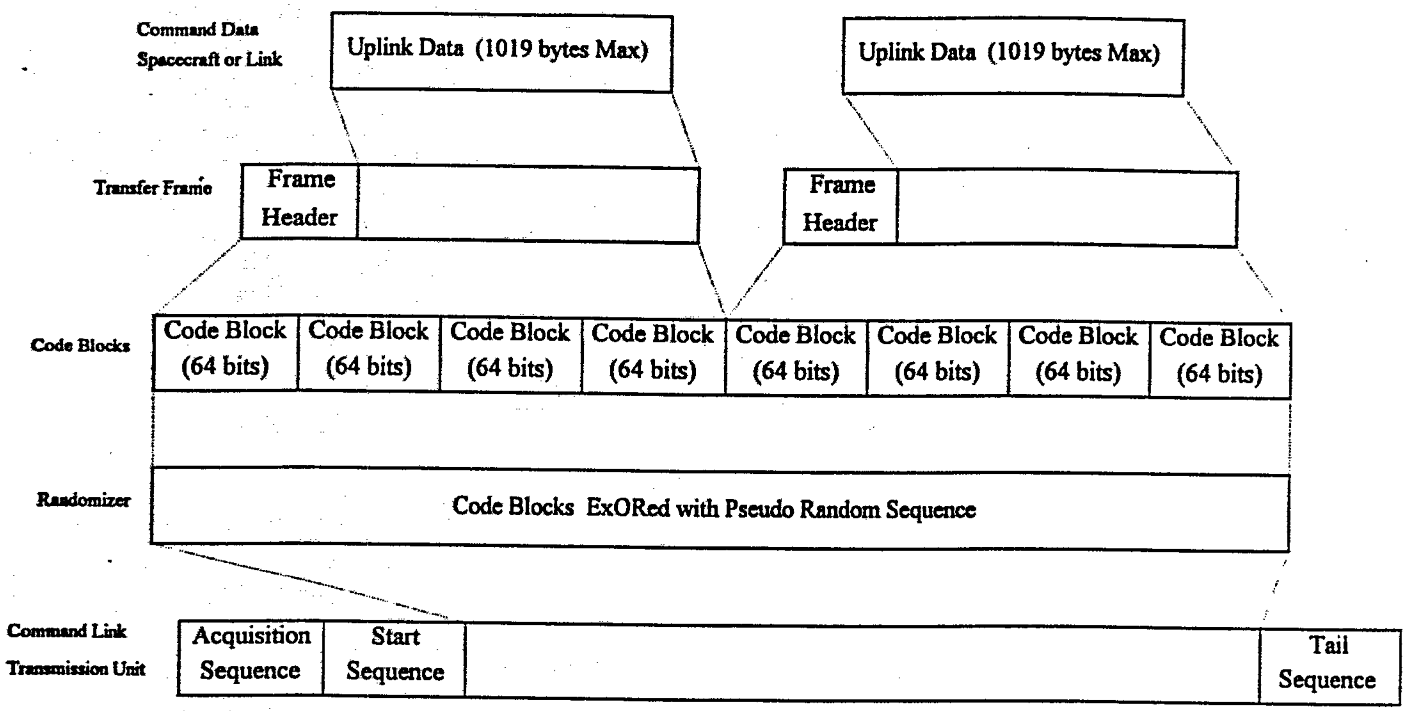

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Figure 6.2.1-2 Telecommand Transfer Frame Format

40 - bits								≤ 1019 bytes
Version #	Bypass Flag	Control Command Flag	Spare	Spacecraft I.D.	Virtual Channel I.D.	Frame Length	Frame Sequence #	Frame Data
2 bits	1 bit	1 bit	2 bit	10 bits	6 bits	10 bits	8 bits	

6.2.1.1. Command Verification

Both C&DH components shall always be capable of executing complete telecommand transfer frames, but the telecommand virtual channel ID shall be for only one of the two C&DH components. Thereby, only one C&DH component will execute uplinked commands. The entire telecommand frame shall be checked for errors (including command errors) before any command in the frame is executed. The entire frame(s) shall be rejected if an error is found. An uplink protocol shall be used to prevent any successive command sequence (within the telecommand frame or over multiple frames) from being processed if the previous telecommand frame was rejected. Additionally, individual commands also contain an error detection code. A command will not be executed if an error is detected. Specifically, the C&DH component shall provide for verification of the command as follows:

- Telecommand frame shall be verified as follows (entire frame is rejected if any errors are found):
  - Spacecraft ID shall be verified.
  - Virtual Channel ID shall be verified.
  - Frame length shall be verified.
  - Each 64 bit code block shall be verified to be free of errors.

- Each command within telecommand frame shall be verified prior to processing. Entire frame shall be rejected if any errors are found. Command reject flag in the Command Link Control Word (CLCW) shall be set (and sequence number not incremented) if an error is detected. Additional protection against commands capable of causing serious damage to the instrument shall be the responsibility of the instrument experimenter.
  - Each command field shall be checked for illegal values.
  - Each command shall include a 16 bit CRC code for verification.

6.2.1.2. Command Interfaces

Four types of commands given in Table 6.2.1.2-1 shall be supported. Detailed command interfaces shall be given in APL document 7345-9030.

Table 6.2.1.2-1 C&DH Commands to Instrument and Spacecraft Components

Command Types	Description
Logic Pulse Commands	Provide a 40-millisecond transistor switch closure to ground, except for transponder logic pulse commands which are +5 V.
Data Commands	Provide variable length data to observatory components. The C&DH component shall only act as a bent-pipe so that no header or checksum will be added to the data by the C&DH component. The data interface consists of (no hand-shaking): • gated clock - present when commands are sent to components. • enable - indicates when command data will be output in response to clock signal. • command data Observatory components shall be capable of responding to the data command interfaces from both of the C&DH components although only one is active at a time. Memory load shall be accomplished over the data interface.
Relay Commands	These shall be pulses for latching and non-latching relays in the Power Switching and Ordnance Components to apply power (or turn off) observatory components; select and arm thrusters; and control the RF coaxial switches.
Remote Relay Commands (for relays not in Power Switching or Ordnance Components)	These shall provide a pulsed +28 V signal or a pulsed (20, 40, 60 or 80 millisecond duration) ground signal for switching relays in instruments. The remote relay command shall be implemented with non-latching relays in the Power Switching Component. Remote relays shall not be used to switch primary power in the instrument. The command drives the instrument relay coils by providing momentary closure to ground or spacecraft power bus for the above duration.

6.2.2. Bus Protection

The C&DH component shall respond to three possible LVS (low voltage sense) events by executing a sequence of commands to remove loads from the bus. Command sequences for removing loads can be uploaded from the ground. A pre-defined default sequence shall be executed in the case of an LVS condition after a C&DH component reset. (Note, the sequence for the LVS events can be uploaded from the ground, but will be replaced by the default sequence in case of a C&DH reset.)

Three LVS signals shall be provided by the Power Subsystem to the C&DH component:

- Main Bus Low Voltage Sense indicating a bus voltage <26 V.
- Battery Low Voltage Sense threshold #1 indicating a battery bus voltage of <19.8V.
- Battery Low Voltage Sense threshold #2 indicating a battery bus voltage of <18.9V.

Each LVS signal can be masked by ground command. The LVS sequence shall include aborting thruster firing during an LVS condition.

6.2.3. RF Watchdog Timer

The C&DH component shall include an RF watchdog timer which, after 80 hours without valid uplink commands, shall cause the C&DH subsystem to command the RF coaxial switches to select one of the two antenna configurations given in table 6.2.3-1 (see also figure 6.1.-1) and toggle between configuration #1 and #2 every 80 hours without processed commands. The default time is 80 hours, however ground can uplink any other time > 1/2 hr. In the event of a C&DH reset, the uplinked time is replaced by the default time. (Configurations given below unchangeable by ground command.) The C&DH component does not automatically change bit rates.

The ground shall be responsible for sending commands to the inactive C&DH component to reset the last resort timer which also resets the RF watchdog timer (LRT resets after 96 hours). (80 and 96 hour timer reset intervals were specified by GSFC MOM)

Table 6.2.3-1 RF Watchdog Timer Time Out - Antenna Configurations

Antenna Configuration	Prime Side (A)	Redundant Side (B)
Configuration #1	Antenna #2 (A2) -Z	Antenna #1 (B1) +Z
Configuration #2	Antenna #1 (A1) +Z	Antenna #2 (B2) -Z

6.2.4. Pseudo Randomizer

The ground shall have randomized the uplink data prior to transmission to the observatory so that the RF receiver does not loose synchronization due to a long string of ones or zeros. The randomization ensures that one bit transition occurs in every consecutive string of 64 bits and 125 transitions in every string of 1000 consecutive bits. The C&DH component shall reverse the command data randomization.

6.2.5. Thruster Control

The C&DH component shall activate relays in the Ordnance Component to control thrusters. The following three commands shall be required to activate the thrusters. Design shall include provision to protect against unplanned activation of thrusters due failure or C&DH resets. Autonomy shall be used to shut down thrusters in the event that the sun angle >25 degrees (TBR).

- Thruster setup command shall specify:
  - Whether maneuver start time is referenced to sun pulse or a ground command.
  - Thruster on and off time (specified separately for top deck and bottom deck thrusters).
  - Number of times thrusting cycle is to be repeated.
  - Which of the 10 thrusters are to be activated.
- Thruster arm command shall turn on the latching arming relay(s).
- Thruster fire command shall close a FET switch (or open/close sequence for sector firing).

6.2.6. Spacecraft Separation Events

The C&DH component shall provide for a sequence of commands for the following functions to be autonomously initiated after the spacecraft separates from the launch vehicle. For redundancy there are two switches, one for C&DH A and one for C&DH B. Separation shall be indicated by two signals from each separation switch located on the observatory attach fitting. Both signals on one switch must change state to indicate separation. Both C&DH components shall be capable of initiating the separation sequence. The sequence occurs without ground intervention prior to first ground contact (the sequence cannot be changed by ground command).

- deployment of the four solar panels TBD seconds after the spacecraft separates from the launch vehicle.
- Turn on Pre-modulation Conditioner and RF Transmitter TBD minutes prior to first ground contact. (Requirement is TBR and can be accomplished via commands not using the separation sequence bin)

6.2.7. Ground Test Shutdown Sequence

During ground testing, should the Ground Support System become inoperable, the C&DH component shall provide a shutdown sequence for orderly turn off of all observatory components. The sequence shall be able to be masked by command. Reasonable means shall be taken in implementing this feature such that the sequence cannot be invoked after launch.

6.2.8. Data Handling Function

The C&DH component shall collect digital science data from the instruments and housekeeping data from multiple sources and format these into a CCSDS compatible stream. Specifically, the component shall:

- Generate the timing signals (no hand shaking) for data transfer given in table 6.2.8.7-1.
- Collect science and housekeeping data.
- Extract selected housekeeping for autonomy function parameter checking.
- Record data on selected recorder.
- Format the real-time bit stream, interleave with recorder data and encode for downlink transmission during DSN contact time. Format shall be compatible with the NASA ground data system Virtual Channel Transfer Frame concept.
- Record and downlink formats shall be independently selectable.
- Provide for a baseband output for ground testing.

6.2.8.1. Telemetry Description

Instrument bit allocation per minor frame is shown in Table 6.2.8.1-1. Spacecraft housekeeping bit allocation is 176 bits (not including any headers or selectable telemetry byte) per minor frame (there are 16 minor frames per major frame) in the Science Format. Housekeeping which needs to be accessible in real time shall be in a fixed position in each minor and each major frame or synchronized to the 2 x or 8 x major frame pulse.

Table 6.2.8.1-1 Instrument Serial Digital Channel Allocation Per Minor Frame

Instrument	Science Format bits(1)	RTSW Format	Total bits Read out by C&DH
CRIS	464		464
SIS	1992		1992
ULEIS	1000		1000
EPAM	168	168	168
MAG	304	48	304
SWEPAM Ion	544	168	712
SWEPAM Electron	456		456
S^3DPU	1624		1624
TOTAL	6552	384	6552

(1) Instrument bit stream includes instrument housekeeping and science data.

The following housekeeping shall be collected. Non-redundant components, including instruments, shall duplicate the following telemetry interfaces for each of the two C&DH components:

- Digital tell-tales (switch and logic) - shall be used to sample the state of a two state device (such as a switch).
- Serial digital - shall be used to collect a fixed amount of serial digital data at a periodic interval. Data collection shall occur in exactly the same spot and will be the same length in any minor frame. Each C&DH component shall be capable of limit checking telemetry data, and executing a command if an out-of-limit condition is detected. For instruments, such data must be in a fixed location in the spacecraft minor (and major) frame. To dump memory contents, instruments shall replace their normal allocation of science data with their memory data (for a fixed number of frames). Science and memory content data should be formatted so that the ground decommutation process can detect which type of data is present.
- 0-5 V single-ended Analog voltages - shall be used to sample and digitize voltages which have been conditioned to be within a 0 to 5V range.
- 0-50 mV differential Analog voltages - shall typically be used to sample the voltage across a current sensing resistor.
- AD 590 Sensor - shall be used for temperatures over the range of -60 to +100° C.
- PT103 Sensor -shall be used for temperatures over the range of -100 to +150 ° C.

Provision shall be made for the active C&DH component to collect limited telemetry status from the inactive C&DH component.

6.2.8.2. Data Collection and Formatting

The C&DH component shall collect instrument bit streams and all housekeeping data and shall collect and digitize analog data. These shall be formatted for storage on the recorders, for real time downlink transmission and for recorder playback, as appropriate. CCSDS compatible Virtual Channel Transfer Frames (VCTF) shall be created to be compatible with NASA's ground receiving data system.

Two collection formats, as follows, shall be available:

- all housekeeping and instrument bit stream every second to form a single serial bit stream.
- all housekeeping and instrument bit stream every second except housekeeping serial digital channels (TBR, power switching/ordnance telemetry may be collected) to form a single serial bit stream.

6.2.8.2.1. Recorder VCTF

The above single serial bit stream shall be formatted into a Time Division Multiplexed Major/Minor frame structure and a recorder VCTF (see figure 6.2.8.2.1-1) as follows:

Minor Frame (1 second long):	bit stream and selectable telemetry shall be placed in the data field of a minor frame and a minor frame header appended.
CCSDS compatible Packet :	Minor frame shall be placed in data field of a packet and packet header appended.
Recorder VCTF:	Packet and a Command Link Control Word* shall be placed in the data field of a recorder VCTF with a recorder VCTF header and recorder sync appended. The data field in the packet is one minor frame and is locked to the recorder VCTF. Virtual channel ID shall be 4. (* CLCW provides status information on the uplink telecommand transfer frame.)

6.2.8.2.2. Downlink VCTF

The Downlink VCTF shall be identical to the Recorder VCTFs except real-time VCTF header and downlink sync are appended instead of recorder VCTF and sync. The data field in the packet is one minor frame and is locked to the real-time downlink VCTF. Virtual channel ID shall be 1. The downlink VCTF shall have Reed-Solomon and convolutional coding appended.

6.2.8.2.3. Playback VCTF

The recorder playback data VCTF (section 6.2.8.2.1) shall be placed in the data field of a playback VCTF and a playback VCTF header and downlink sync appended (see Figure 6.2.8.2.3-1). Recorder playback data is unsychronized to Playback VCTF. Virtual channel ID shall be 2.

6.2.8.3. Record Formats

Recorder data rate shall be 6944 bit/sec. Any one of the formats given in table 6.2.8.3-1 can be selected for recording. The Science format shall be the default format.

Figure 6.2.8.2.1-1 Record/Downlink VCTF Format

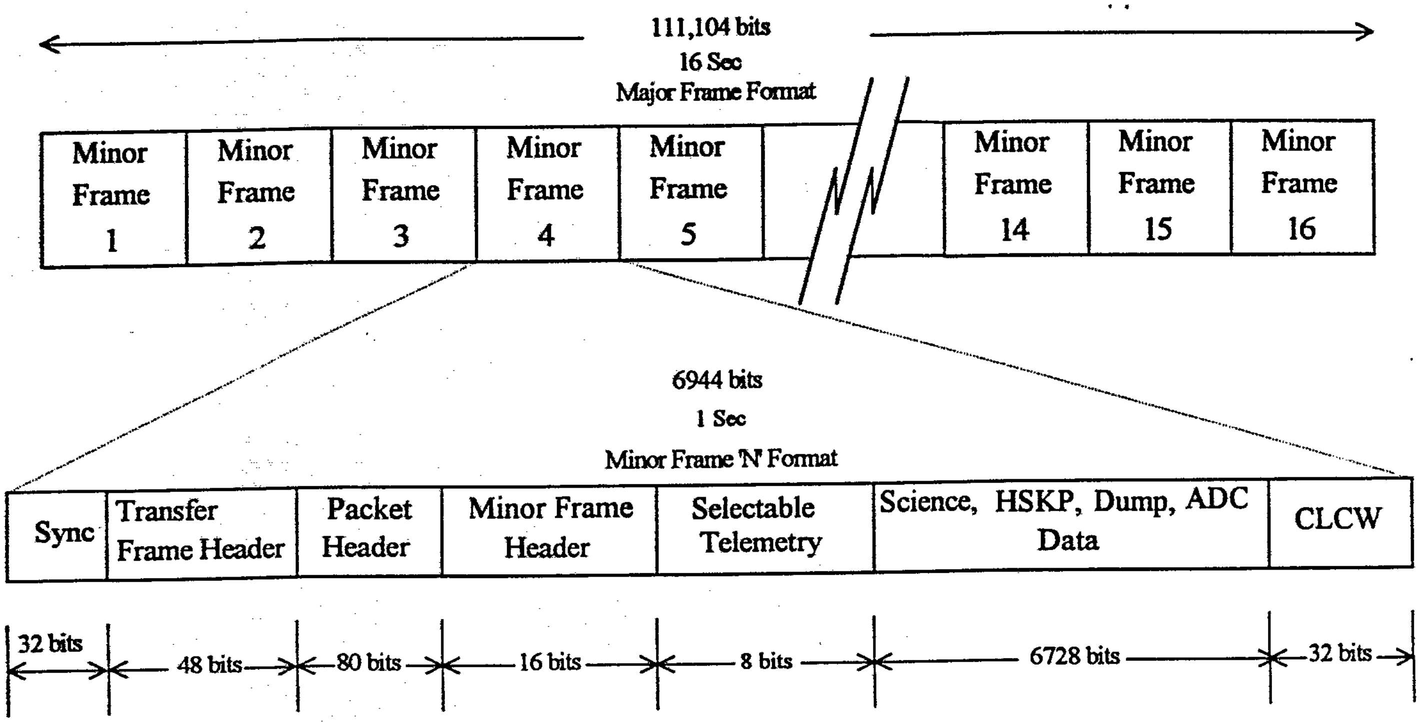


Figure 6.2.8.2.3-1 Playback VCTF Format

Sync	VC Header	Forward Ordered Recorder Playback Data Not Locked to Playback VCTF
32	48	6864

Table 6.2.8.3-1 - 6944 bit/sec Formats

Format	Description (data repeats every minor frame, i.e. 1 sec, unless otherwise noted)
Science	All instruments' bit streams and all housekeeping. Housekeeping repeats every major frame (16 sec)
C&DH Bin Dump	Replace Science format housekeeping data with C&DH Bin Dump Data
C&DH Memory Dump	Replace Science format housekeeping data with C&DH Memory Dump Data
AD&C	Complete AD&C data and housekeeping every second; reduced science bit stream

6.2.8.4. Downlink Formats

The C&DH component shall support three downlink data rates;

- 434 bits/sec low data rate - observatory housekeeping and RTSW only (no instrument bit stream)
- 6,944 bits/sec real-time data - instrument bit stream and observatory housekeeping
- 76,384 bits/sec real-time/recorder data - real-time data interleaved with recorder playback

6.2.8.4.1. 434 bit/sec Low Data Rate Downlink

The 434 bit/sec Low Data Rate Downlink shall consist of one real-time VCTF (section 6.2.8.2.2) per 16 seconds. Formats given in Table 6.2.8.4.1 shall be available. Figure 6.2.8.4.1-1 shows the frame structure. This downlink supports the RTSW experiment.

Table 6.2.8.4.1 - 434 bit/sec Low Data Rate Downlink

Format	Description (data repeats every minor frame, i.e. 1 sec, unless otherwise noted)
Low Rate Housekeeping	Complete observatory housekeeping and AD&C data. Housekeeping repeats every 16 sec.
Low Rate C&DH Bin Dump	Same as Low Rate Housekeeping format with C&DH Bin Dump Data added.
Low Rate C&DH Memory Dump	Same as Low Rate Housekeeping format with C&DH Memory Dump Data added.
Low Rate A&DC	Complete AD&C data and reduced housekeeping. Housekeeping repeats every 16 sec.
Real Time Solar Wind	Solar wind science data from selected instruments.

Figure 6.2.8.4.1-1 434 bit/sec Low Data Rate Downlink Frame Structure

16 Seconds
6944 Bits

Sync	Transfer Frame Header	Packet Header	Minor Frame Header	Selectable Tlm Byte	HSKP, Dump, ADC, RTSW Data	CLCW
32	48	80	16	8	6728	32

6.2.8.4.2. 6944 bit/sec Downlink

The 6944 bit/sec Downlink shall consist of one downlink VCTF (section 6.2.8.2.2), per second. The formats given in Table 6.2.8.3-1 shall be available. Figure 6.2.8.2.1-1 shows the frame structure.

6.2.8.4.3. 76,384 bit/sec Real Time/Recorder Playback Downlink

The 76,384 bit/sec Real Time/Recorder Playback Downlink shall consist of one real-time VCTFs (section 6.2.8.2.2) and 10 playback VCTFs (section 6.2.8.2.3) per second. The formats given in Table 6.2.8.3-1 shall be available. Figure 6.2.8.4.3-1 shows the frame structure. A major frame (16 sec long) shall consist of 16 minor frames.

6.2.8.5. Coding

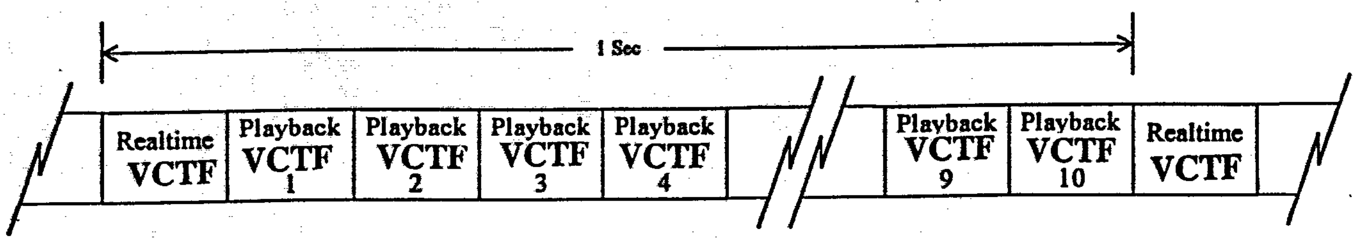
The downlink data shall be coded to protect against error using two levels of coding. Each VCTF shall be encoded using a (233,255) Reed-Solomon code with an interleave of I=4. The Reed-Solomon encoded data shall then be convolutional coded using the CCSDS standard (7, 1/2) convolutional code. Encoded symbols shall be converted to Bi-Phase-L for the RF subsystem.

6.2.8.6. Spacecraft Master Oscillator

The C&DH component shall contain a master oscillator to provide the timing and synchronization for the spacecraft and instrument components. All data collection clocks, downlink clocks, and data time tags, shall be derived from this master oscillator. The on-board 'time' shall be implemented with a master oscillator based counter. Oscillator performance shall allow any science datum to be known with an accuracy of 0.1 seconds absolute and an accuracy 0.025 sec relative to data from other ACE instruments after ground processing.

The start of minor frame and the sun pulse (see section 6.2.9) shall be 'time' tagged in the telemetry. Instruments shall be required to provide for the timing relationship of the science data to the minor frame pulse or sun pulse within their own instrument bit stream.

Figure 6.2.8.4.3-1 76,384 bit/sec Real Time/Recorder Playback Downlink Frame Structure



6.2.8.7. Data Collection and Synchronization Interface Signals

The C&DH component shall supply the following signals to observatory components.

Table 6.2.8.7-1 Data Collection and Synchronization Interface Signals

Signal	Purpose	Characteristic
Major Frame Pulse	synchronization	Active at start of each major frame.
2 x Major Frame Pulse	synchronization	Active at start of every second major frame.
8 x Major Frame Pulse	synchronization	Active at start of eighth major frame.
Minor Frame Pulse	synchronization	Active at start of each minor frame.
Sun Pulse	synchronization	Nominally once per 12 seconds.
Spin Clock	synchronization	16384±10 pulses between each rising edge of sun pulse. (except during maneuvers, where nutation precludes this accuracy, ±10 is max)
Clock	data collection	10,956 clock pulses between minor frame pulses.
Read Out Gate	data collection	Indicates when serial data should be output in response to Clock.

6.2.9. Attitude Information

6.2.9.1. Sun Pulse and Spin Clock

A dedicated interface between the C&DH component and a sun sensor shall provide the C&DH component with X and Y axes sun angle data. From these data, the C&DH component shall generate a sun pulse when the sun angle is ≥4 deg which shall be distributed to instruments and available in the downlink telemetry.

The C&DH component shall generate a spin clock from the sun pulses. The spin clock shall contain 16384+/-10 pulses between each rising edge of the sun pulse. This +/-10 requirement does not apply during maneuvers, where nutation precludes this accuracy.

The C&DH component is not required to provide a pseudo sun pulse in the event of a sun sensor failure. Instruments shall be designed to survive the absence of a sun pulse. The sun pulse shall be 'time' tagged in the telemetry format.

The C&DH component shall compute a phase angle (a running count of the number of spin clock pulses since the last sun pulse). Phase angle counter shall be zeroed by sun pulse and incremented by spin clock. Value of phase angle shall be latched at the beginning of every minor frame. Thus, observatory rotation angle can be referenced to on-board 'time'.

6.2.9.2. Star Tracker Data in ADC Format

For ADC format, complete star tracker information shall repeat every major frame (16 sec) - information on ten star images shall be contained in ten minor frames (one star/sec), with the remaining

6 minor frames used for star tracker housekeeping. In non-ADC format, data for one star image shall be included in one major frame, or not at all, depending on format selected.

6.2.10. Power Switching Component and Ordnance Component

All primary power switching and ordnance functions for the observatory shall be performed by the C&DH component which controls the power switching component and ordnance component.

The C&DH subsystem shall control the power switching component or ordnance component to:

- Based on uplink relay commands:
  - turn off any instrument or non-critical spacecraft component.
  - turn on any instrument or non-critical spacecraft component.
  - select thrusters.
  - arm thrusters.
  - remove instrument covers.
  - deploy magnetometer boom (at first ground contact).

- Based on on-board indicators, autonomously
  - select the default antenna configuration if the RF watchdog timer has timed out.
  - **turn off spacecraft or instrument components if autonomy rules are violated.
  - turn off spacecraft or instrument components in the event of an LVS.
  - turn off thrusters in the event of a C&DH reset,.
  - **turn off thrusters in event sun angle >25 deg (TBR) (can be masked by ground command)
  - deploy solar panels (after separation from the launch vehicle as indicated by a signal from the separation switch. Ground commands can be sent in case of faulty deployment).

  (** autonomy not available after C&DH reset, because bins are cleared.)

Each latching relay shall contain two actuating coils (one each for the two C&DH components) with a single set of contacts. Non-latching relays shall be redundant. For relays that switch power, one more wire than required shall be implemented to carry the power to the user. Latching relays shall have tell-tale contacts whose status are telemetered to the ground.

6.2.11. Data Recorders

There shall be two solid state recorders (SSR), cross strapped to each C&DH component; however, each recorder shall only respond to one of the C&DH components at a time. Each SSR shall have end of life capacity of 650 megabits (≥26 hours of recorded data). The design shall not preclude the use of both recorders, one for recording and one for dumping data to the ground. Simultaneous read and write capability is not required. Differential clock and data interfaces shall be used between C&DH component and recorder.

Recorder operational modes shall be:

- record
- reproduce
- set recorder pointer
- set reproduce pointer
- standby

6.3. POWER SUBSYSTEM

The power subsystem shall provide power to the observatory components starting at launch and continuing throughout the 5 year mission. The 12 A-hr battery shall support the critical components during launch and potentially until first ground contact (in the case of a solar array deployment failure). The solar arrays shall be deployed autonomously TBD seconds after the spacecraft separates from the launch vehicle. Through the remaining mission, the battery may be required (as much as its capability permits) to supplement the solar array power if the solar array cannot support the loads. On orbit, the solar array shall be wired directly to the main power bus and supply the required load power. Prior to launch, the power subsystem shall support observatory testing. A block diagram of the power subsystem is shown in Figure 6.3-1.

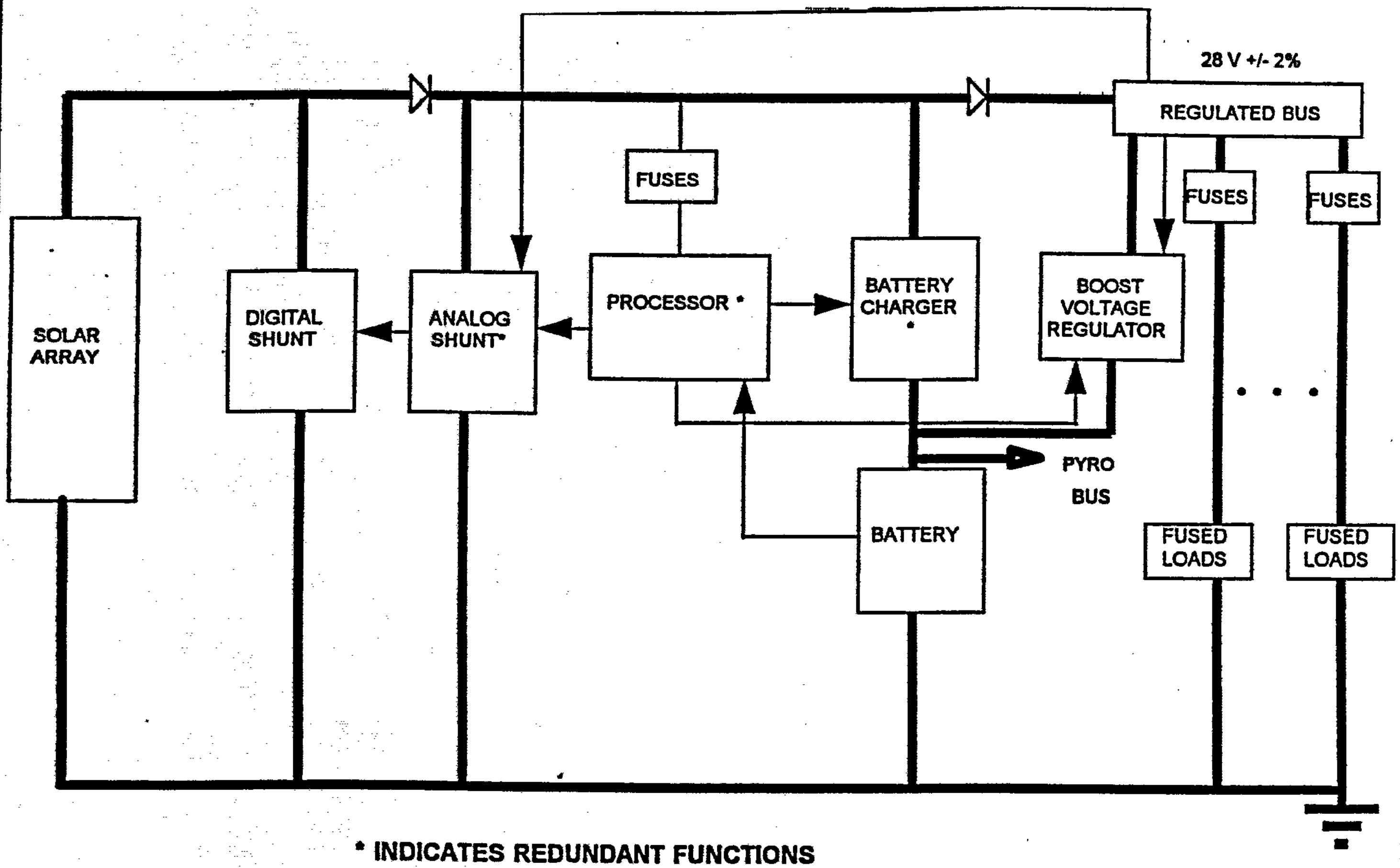
The power subsystem shall consist of the following components/functions:

- Solar array comprised of four solar panels
- 12 A-hr nickel cadmium battery
- Digital and analog shunt system
- Battery charger
- Boost voltage regulator
- Control and protection circuitry

The power subsystem shall:

- Provide a regulated output voltage of 28 $\pm$ 2% V dc for 5 years to all observatory components and heaters. Launch shall be nominally August 1997, no later than December 1997.
- Support power requirement given in table 5.1.1-2.
- Provide for a single-system primary power ground point.
- Implement twisted pairs for all DC power cables.
- Provide bus protection with three undervoltage trip points; one for main bus and two for battery.
- Provide three independent low voltage sense interfaces to C&DH subsystem.
- Provide for overvoltage protection of 30 volt max.
- Design power subsystem so that no single failure aborts the mission (reduced capability is permitted), except for battery during launch. Provide for internal redundancy for critical functions.
- Provide for battery discharging and charging to condition the battery.
- Dissipate excess array energy through shunt elements.
- Provide for receiving power from an external source (for ground testing and battery charging).
- Measure the total observatory current.

Figure 6.3-1 Block Diagram of Power Subsystem



* INDICATES REDUNDANT FUNCTIONS

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6.3.1. Spacecraft Power Buses

Two buses shall be supported:

Pyro bus (battery): • Battery bus shall be taken directly from battery and operates over 18.9 to 27 V. Loads powered during launch, that is, critical loads (see table 5.6-1) and solar array pyros shall be capable of operating down to 18.9 V.

Main Bus: • Normal operating range +28 V ± 2%.
• Operating capability over range 18.9 to 30 V (critical subsystems shall be capable of operating over this range, other subsystems shall survive this range). The main bus shall be protected from overvoltage, 30 volts max.

6.3.1.1. Bus Protection

All loads on the main bus shall be fused. Additionally C&DH autonomy limit checks loads nominally once per second and removes loads that exceed pre-defined limits. The Low Voltage Sense functions shall provide protection against large faults (e.g.short to ground).

6.3.1.1.1. Low Voltage Sense (LVS)

The power subsystem shall detect one or more of the low voltage conditions below and provide three separate low voltage sense signals to the C&DH component:

- Main bus LVS indicating a bus voltage < 26 volts.
- Battery LVS threshold #1 indicating a battery bus voltage <19.8 volts.
- Battery LVS threshold #2 indicating a battery bus voltage <18.9 volts.

6.3.2. Solar Array

The solar array shall consist of four identical 34 x 59 inch solar panels which provide the power for the observatory subsystems for post-launch-vehicle separation and on-orbit operations. Each panel contains solar cells, cover glass and adhesives, thermistors, insulation sheet, and electrical connectors. There are 5 circuits on each of the four panels and each circuit has 1 cell in parallel and 86 cells in series. Normal operation shall be for the top of the solar panels (cells) to continuously face the sun. On separation from the launch vehicle, the solar panels shall be deployed in pairs.

To minimize the solar array contribution to the observatory magnetic field, the solar array shall be backwired and its design shall follow the guidelines given in the APL environmental specification 7345-9007. The solar array design shall:

- Provide ≥443 watts for the first 2 years (required mission life) and ≥430 watts for the remaining 3 year (EOL total 5 years) at 28V ±2% (20° sun angle).
- Provide for operation over a temperature range consistent with a sun angle between 0 and 20°.
- Meet performance requirements during solar quiet as well as solar active times.
- Provide for by-pass diodes to accommodate shadowing from magnetometer boom or other sources.
- Meet performance requirements in the presence of Ultra-Pure hydrazine thruster

shunt and 20 digital shunt circuits (5 shunt circuits per panel). The analog shunt shall dissipate all the excess current up to a maximum level. When this maximum level is reached, a signal shall be sent to a sequencer that proceeds to turn on additional digital shunts. A similar procedure shall be followed for turning off digital shunts as a minimum level in the analog shunt is reached. Thereby the bus voltage shall be maintained in regulation.

6.3.4.2. Boost Voltage Regulator

The boost voltage regulator will turn on if the main bus voltage drops due to insufficient power from the solar array and boost the battery voltage to the regulated bus voltage. The boost voltage regulator shall sense the main bus voltage and compare it to the precision voltage reference from the control processor. The error signal shall feed a pulse width modulator and boost circuits to pump current from the battery and supply it to the load at the higher voltage.

6.3.4.3. Battery Charger

The battery charger shall consist of a series of voltage regulators with input from the solar array bus and output to the battery. The charge control method used shall be constant current and modified constant potential with temperature compensated voltage limits. The maximum charge current is provided to the control processor by ground command. This maximum is used as the reference voltage and compared to the charge current by the current control loop in the charger. Discharge and reconditioning of the battery, under ground control, can be effected by passing a 0 maximum charge current reference to the charger.

The battery voltage limit and temperature dependence shall be established with a family of selected V/T curves, one of which is the NASA standard V/T curve. These are programmed into the power subsystem control processor.

6.3.4.4. Processor

The power subsystem shall incorporate a processor to provide the needed control references in the system to maintain the regulation of the bus and provide for battery charging and discharging. The processor shall gather and processes telemetry data from the power subsystem telemetry points. The processor shall also be capable of resetting itself in response to a ground command or on-board problem detection shall provide for autonomy to switch to redundant power subsystem hardware in case of failure.

6.3.5. Shunts

Two types of shunts shall be used; analog and digital shunts. Analog shunts are comprised of a series of resistors mounted on a plate. There are four digital shunt packages, one for each solar panel. Each package contains one digital shunt for each of the 5 solar array circuits on each panel (total of 20 digital shunts).

6.3.6. Observatory Power Modes

The observatory power modes are given in table 6.3.6-1.

Table 6.3.6-1 Observatory Power Modes

	NOMINAL	LAUNCH	MANEUVERS		RELAY ACTIVATION	
POWER ESTIMATES			PRE	FIRE	ANT (1)	OTHER
INSTRUMENTS POWER						
NOMINAL(solar quiet; includes nominal heater power)	x					
PEAK(solar active; operational heaters; autonomously operated motors)			x	x	x	x
POWER						
BATTERY: charge (2)	x		x	x	x	x
BATTERY: discharge		x				
CONTROL ELECTRONICS	x	x	x	x	x	x
DISSIPATOR ELECTRONICS	x	x	x	x	x	x
RF COMMUNICATIONS						
ONE TRANSMITTER/PRE-MOD COND UNIT: on	x		x	x		x
BOTH RECEIVERS: always on	x	x	x	x	x	x
COAX SW: on (for approx 40 msec)					x	
AD&C						
STAR TRACKER: on	x		x	x	x	x
ONE SUN SENSOR (2 Optical heads with electronics): on	x		x	x	x	x
PROPULSION (3)						
THREE THRUSTERS: on				x		
CATALYST BED HEATER: on (approx 1 hr prior to maneuver) (4)			x			
PRESSURE TRANSDUCER: on	x		x	x	x	x
CMD & DATA HANDLING						
BOTH DATA STORAGE: on (idle, record or playback)	x		x	x	x	x
C&DH#1: COMMAND FUNCTIONS on; DH FUNCTIONS disabled		x				
C&DH#1: COMMAND FUNCTIONS on; DH FUNCTIONS on	x		x	x	x	x
C&DH#2: COMMAND FUNCTIONS on; DH FUNCTIONS disabled		x				
C&DH#2: COMMAND FUNCTIONS on; DH FUNCTIONS on	x		x	x	x	x
ONE RELAY ACTIVATED (THRUSTERS) (3,5)			x	x		
ONE RELAY ACTIVATED (ANT. SWITCH)					x	
ONE RELAY ACTIVATED (REDUNDANCY SELECTION OR TURN ON COMPONENT)						x
PSU/ORDNANCE FIRE: standby	x	x	x	x	x	x
THERMAL (heater requirement under review)						
LAUNCH HEATERS		x				
BATTERY HEATERS: on peak			x	x	x	x
nominal	x					
PROPULSION HEATERS: on peak			x	x	x	x
nominal	x					
FWD DECK HEATERS: on peak			x	x	x	x
nominal	x					
AFT DECK HEATERS: on peak			x	x	x	x
nominal	x					
ADDITIONAL HEATERS: on peak			x	x	x	x
nominal	x					
ANALOG SHUNT HEATERS: on	x		x	x	x	x
C&DH HEATERS (oscillators): on	x		x	x	x	x

General Note: Solar Array/Magnetometer boom ordnance supported by battery. Instrument cover ordnance supported by Solar Array.

Notes: (1) Transmitter must be turned off prior to switching antennas (emergency hot switching is TBR).

(2) Immediately after solar array deployment, battery recharge will start at approx 34 watts and taper to 4 watts within approx 6 hrs (instruments are off during this time). Magnetometer turned on at first ground contact .

(3) Latch valves require 41 watt for 50 msec and will be opened prior to launch. Relays to drive latch valves consume 3.1 watt for approx 40 msec.

(4) Either radial or axial heaters are turned on. 6 radials is worst case.

(5) Thruster Setup and Arm relays are latching and consume1.7 and 3.5 watts respectively and are activated sequentially. Fire relays are FETs.

6.3.7. Types of Power Subsystem Interfaces

The types of signal interfaces between the power subsystem and other observatory subsystems are as follows:

Power Subsystem Signal	Source or Destination	Characteristics
Main Bus Power	to observatory fused loads via Power Switching or Ordnance Components	+ 28 V ± 2%
Pyro Power	to solar array pyro circuits via Power Switching or Ordnance Components	+18.9 to 27 V
Data Commands	from C&DH subsystem	see table 6.2.1.2-1
Remote Relay Commands	from C&DH subsystem	see table 6.2.1.2-1
LVS Main Bus Threshold	to C&DH subsystem	dedicated interface
LVS Battery Threshold #1	to C&DH subsystem	dedicated interface
LVS Battery Threshold #2	to C&DH subsystem	dedicated interface
0-5V single-ended analog voltages Telemetry	to C&DH subsystem	see section 6.2.8.1
0-50mV differential analog voltages Telemetry	to C&DH subsystem	see section 6.2.8.1
PT 103 sensor telemetry	to C&DH subsystem	see section 6.2.8.1
AD 590 sensor telemetry	to C&DH subsystem	see section 6.2.8.1
Digital tell-tale	to C&DH subsystem	see section 6.2.8.1

6.4. ATTITUDE DETERMINATION AND CONTROL (AD&C)

The AD&C functions for the observatory shall be accomplished with the following complement of components:

- Single star tracker used in a scanning mode.
- Two sun sensors (each sun sensor shall be comprised of ~~2~~ two optical heads and one electronics package).
- Two passive nutation dampers

Figure 6.0-1 shows these components. All AD&C components shall meet and support all the requirements given in this document. For normal operations, several star measurements and one sun measurement are sufficient to monitor attitude.

6.4.1. Star Tracker

The star tracker (used in a scanning mode) shall be mounted so that it looks out radially from the spin axis. Time delay integration shall be used in the software to obtain suitable signal strength for the spacecraft spin rate. Star measurements will be used to determine the spacecraft spin axis orientation. After each attitude correction, multiple star measurements per spin will be required to unambiguously determine the spin axis attitude.

The star tracker shall meet the following requirements, while the spacecraft is spinning:

- The star tracker shall have a 99% probability of detecting at least 3 stars for any spacecraft spin axis attitude.
- FOV shall be >20x20° and <25x25°
- Sensor aperture shade shall be used to provide stray light protection for angles ≥65° off boresight for sun, and angles ≥40° off boresight for earth and moon.
- Spin rate: 30°/sec
- Total random error shall be 9 arc-minute (3 sigma) max.
- Star tracker shall be capable of tracking 5 stars within FOV and storing star centroids for the 16 brightest star images. 10 stars shall be sent to the C&DH subsystem for the downlink.
- Star tracker electronics shall only report the star's position once as it crosses the sensor's X-axis (at 5 rpm spin rate, a star will be detected multiple times as it crosses the tracker's FOV). All star data processing and storage shall be done in the star tracker electronics.
- Star tracker processor shall reduce star data and provide, to the C&DH component, a formatted data stream giving relative position, time tag, and intensity of the selected star images as well as star tracker status and health.
- Star tracker shall provide the required data to the C&DH component.

6.4.1.1. Star Tracker Interfaces

The types of signal interfaces between the star tracker and the C&DH subsystems are as follows:

Star Tracker Signal	Source or Destination	Characteristics
Main Bus Power	from power subsystem to star tracker via Power Switching Components	+ 28 V ± 2%
Data Commands	from C&DH subsystem	see table 6.2.1.2-1. Clock, enable and command data interfaces are used.
Telemetry Data	to C&DH subsystem	(see section 6.2.8.7) minor frame pulse; read out gate; clock; star tracker output data

6.4.2. Sun Sensor

The redundant 2-axis digital sun sensors shall consist of two sets of optical heads; each set with an associated electronics package. The optical heads shall be mounted on the spacecraft as follows:

- two optical heads shall be mounted on the top deck with the normal to the sensor face directed towards the sun, parallel to the spacecraft spin axis
- two optical heads shall be mounted on the side of the spacecraft, canted ~120° from the spin axis.

Each electronics package controls one top deck and on side deck optical head. The other set shall be for redundancy. For normal attitude conditions, the top deck sensor will provide the sun crossing pulse and the sun angles as the spacecraft rotates. The sensors mounted on the side of the spacecraft provide for complete spherical coverage in case of an extreme off-axis attitude.

The sun sensors shall meet the following requirements:

- FOV 128° x 128° .
- Resolution 0.5°, each axis.
- Transition accuracy ±0.25°.
- Short term repeatability of MSB: ±0.02°.
- MSB of one axis shall be made available to C&DH component for sun-pulse and spin clock generation.
- The sun sensor electronics shall determine which optical head shall be used to provide sun sensor data and shall provide an ID specifying the sensor selected. The telemetry data shall consist of 8-bit Gray Code sun angle from the illuminated optical heads and a sensor ID.

6.4.3. Nutation Dampers

The two passive nutation dampers will provide damping for perturbations in the observatory's attitude introduced by maneuvers and oscillations due to structures (booms, etc). The nutation dampers and the fluid in the propulsion tanks will help damp these perturbations.

The nutation dampers shall be rings (~18 in diameter, tube diameter 0.875 in) containing an ethylene-glycol solution. The damping time constant (1/e) for the nutation damper shall be ≤ 12 hrs.

6.5. PROPULSION SUBSYSTEM

The spacecraft propulsion subsystem will be used to correct launch vehicle dispersion errors, inject the spacecraft into the L1 halo orbit, adjust orbit, adjust spin axis pointing and maintain a 5 rpm spin rate. High Purity monopropellant hydrazine (N_2H_4), shall be used as the fuel for the spacecraft propulsion subsystem. Nitrogen propellant pressuring agent shall be used as the propulsion subsystem pressurant.

A schematic of the propulsion subsystem is shown in figure 6.5-1. The propulsion system is comprised of:

- Four fuel tanks mounted between the top and bottom decks.
- Four axial thrusters to provide velocity control along spin axis.
- Six radial thrusters to provide velocity control in spin plane and spin rate control.
- All ten thrusters to provide spin axis orientation control.
- Filters
- Pressure transducers
- Latch valves
- Fill and drain valves
- Thruster select valves
- Heaters
- Plumbing and structure

The propulsion subsystem design shall limit the loss of propellant or pressurant due to the failure of any subsystem component. Tanks and groups of thrusters shall be cross strapped to allow the use of remaining system resources in the event of a component failure. To reduce the risk of leakage, the propulsion subsystem shall be an all welded design, with no field joints. Connectors and fittings used in servicing the subsystem with fluids shall be designed to prevent cross-connecting. Surge suppression orifices shall be installed to limit peak pressure spikes at the latch valves and pressure transducers.

Control of magnetic field shall be implemented to the extent practical, including judicious routing of heaters; minimizing the use of magnetic materials; positioning the latch valves as far as practical from the magnetometers; and using twisted pairs for thruster heaters and latch valves.

6.5.1. Propellant Tanks

Figure 6.5.1-1 is an isometric view of the propulsion subsystem showing the location of the titanium tanks on the bottom deck. The propellant tanks shall provide for the following:

- Propellent capacity of 195 kg of hydrazine at a temperature of 298K. Present mass budget support tanks filled to 189 kg. The full capacity of the tanks may be utilized depending on the observatory launch mass.
- . Shall be operated in a blow-down mode
  - initial pressure 305 psia @ 70 °F (363 psia @ 122 °F)
  - final pressure 91 psia @ 70 °F (88 psia @ 50 °F)

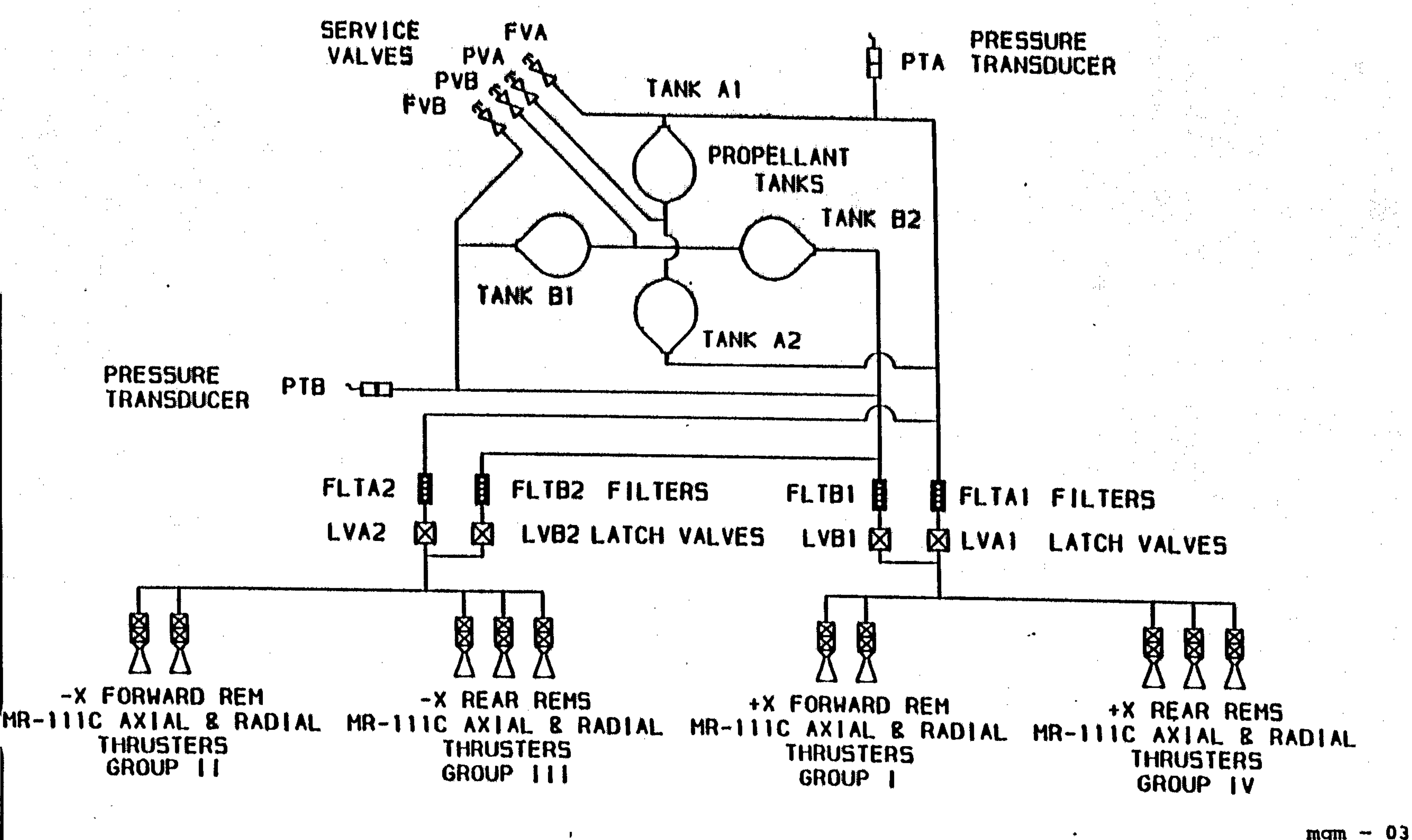

Figure 6.5-1 Propulsion Subsystem Schematic

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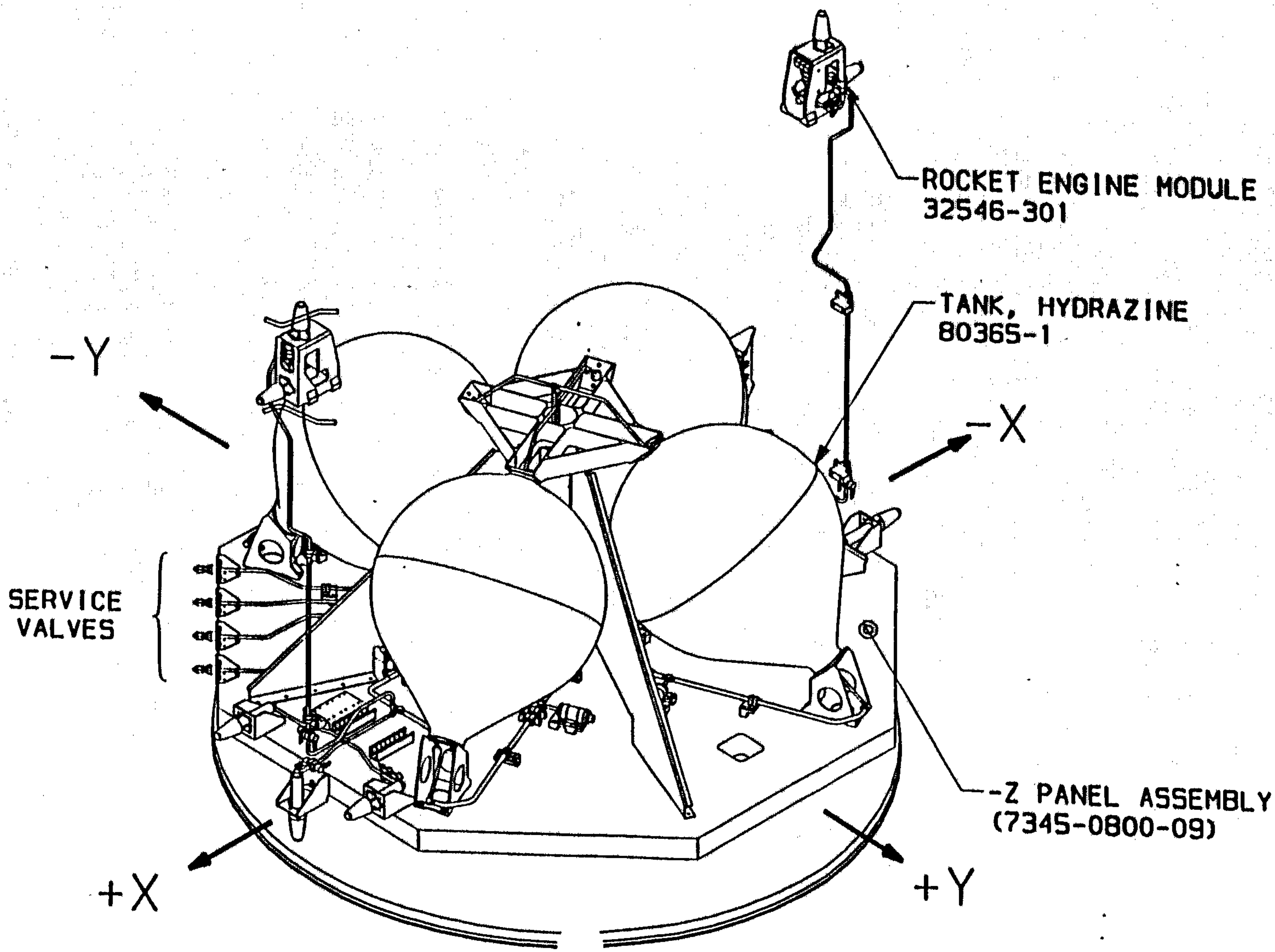

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Figure 6.5.1-1 Propulsion Subsystem Isometric View

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6.5.2. Thrusters

Thrusters shall be located as shown in Figure 6.5.2-1. Thruster functions are given in table 6.5.2-1. The thruster shall meet the following requirements.

- 1 lb_f thrusters shall be used.
- Thrusters shall be designed to deliver a mission average specific impulse of 216 - 221 s at a temperature of 50 - 70 °F.
- Thruster shall be grouped such that a failure in one bank can be isolated, with attitude and orbit control being maintained with remaining bank.
- Thrusters shall be aligned to ±0.75° (goal ±0.5°) of the spacecraft primary axis.

6.5.3. Heaters

Heaters and temperature sensors shall be incorporated in the propulsion subsystem to ensure proper operation. Catalyst bed heaters shall be operated prior to thruster use for most efficient thrusting.

6.5.4. Latch Valves

Four latch valves shall be used to isolate fuel tanks. These shall be open at launch and remain open unless there is a failure and tanks need to be isolated. These shall be controlled by the C&DH subsystem.

6.5.5. Thruster Valves

The fuel supply to each thruster shall be controlled by a fail-safe valve. These shall be controlled by the C&DH subsystem.

6.5.6. Pressure Transducers

Two pressure transducers shall be included in the propulsion subsystem design. Provision shall be made to obtain tank pressure information on the ground when the spacecraft is unpowered.

6.5.7. Emergency Drain of Fuel

The propulsion subsystem and structure design shall provide for the capability to drain fuel in an emergency.

Figure 6.5.2-1. Thruster Location

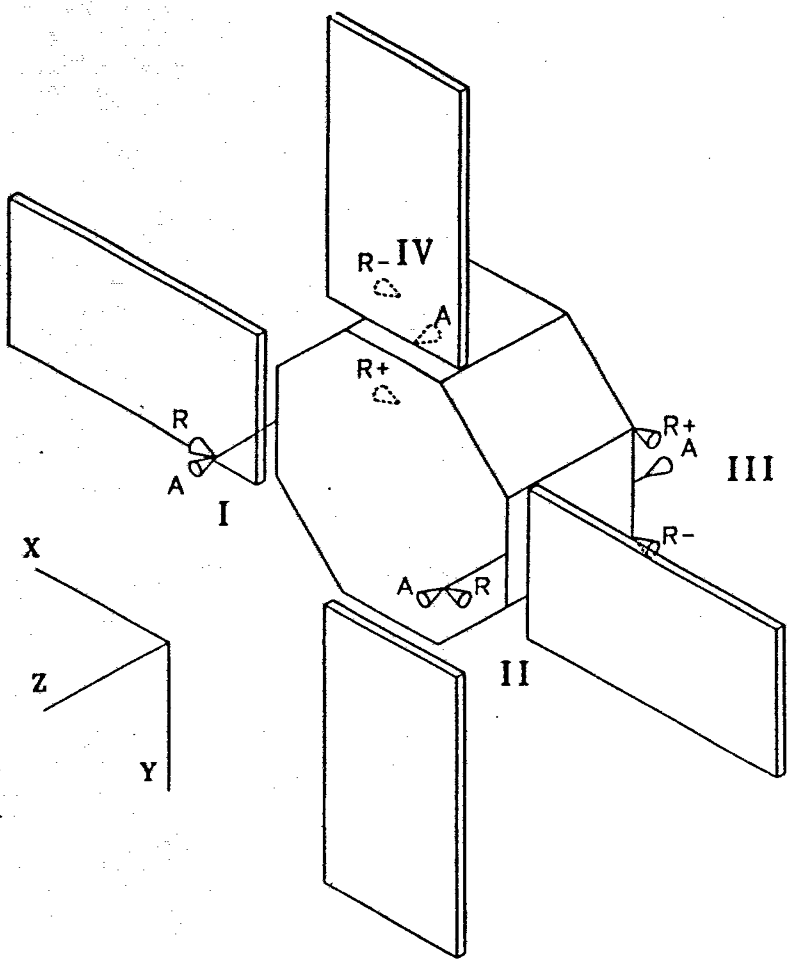

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Table 6.5.2-1. Thruster Functions

	Velocity Control +Z	Velocity Control -Z	Velocity Control in X-Y Plane	Velocity Control in X-Y Plane	Spin Axis Pointing	Spin Axis Pointing	Spin Axis Pointing	Spin Axis Pointing	Spin Rate Increase	Spin Rate Decrease
IA		X			X					
IR			X				X			
IVA	X					X				
IVR+			X					X	X	
IVR-			X					X		X
IIA		X				X				
IIR				X				X		
IIIA	X				X					
IIIR+				X			X		X	
IIIR-				X			X			X

6.5.8. Propulsion Subsystem Interfaces

The interconnection of the propulsion subsystem with the C&DH subsystem is given in table 6.5.7-1.

Table 6.5.7-1 Propulsion Subsystem Interfaces

Propulsion Subsystem Signal	Source or Destination	Characteristics
Main Bus Power to latch valves, heaters and pressure transducers	from power subsystem via Ordnance Fire Component	+ 28 V ± 2%
Remote Relay Commands	from C&DH subsystem	see table 6.2.1.2-1
Relay Commands	from C&DH subsystem	see table 6.2.1.2-1
0-5V single-ended analog voltages Telemetry	to C&DH subsystem	see section 6.2.8.1
0-50mV differential analog voltages Telemetry	to C&DH subsystem	see section 6.2.8.1
PT 103 sensor telemetry	to C&DH subsystem	see section 6.2.8.1
AD 590 sensor telemetry	to C&DH subsystem	see section 6.2.8.1
Digital tell-tale	to C&DH subsystem	see section 6.2.8.1

6.6. STRUCTURE SUBSYSTEM

The structure subsystem shall provide for mounting, accessibility, clearance, alignment, clear fields of view, MLI coverage, and environmental control for observatory instruments and spacecraft subsystems. It shall also provide a mating interface for attaching to the launch vehicle (see figure 5.3.1.1-1). The structure subsystem shall maintain structural integrity while withstanding appendage deployment, transportation, storage, ground handling, environmental testing, launch, and orbit loads as specified in the APL environmental specification 7345-9007. The structure subsystem shall minimize magnetic contamination of the observatory by following magnetic cleanliness guidelines as outlined in that document. Figure 6.6-1 shows an exploded view of the observatory. The layout shall facilitate, to the extent practical, instrument installation and removal without affecting the remaining subsystem.

6.6.1. Structure Design

The structure subsystem design shall provide for the mounting, accessibility, alignment and environmental control for the spacecraft and instrument subsystems. Subsystems shall be mounted as shown in tables 4.1.-1 and 4.2.-2. Provision shall be made for a support for the MLI blankets to cover the decks (except radiators, antennas, thrusters and sensor apertures).

The observatory shall be comprised of an octagon +Z (instrument) upper deck, an octagon -Z lower deck, and eight rectangular side decks (see also figure 5.1.6.-1). All decks shall be constructed of aluminum honeycomb material. The instrument deck shall provide mounting for nine instruments (see figure 6.6.1-1), upper deck thrusters and broad beam antennas, sun sensor optical heads and shall support four deployable solar panels. The magnetometers shall be deployed off the ends of the Y axis solar panels. The lower deck (figure 6.6.1-6), shall support the RF subsystem (including parabolic dish and two broad beam antennas) , the SLAM packages, and the observatory attach fitting. The side panels shall support the remaining spacecraft subsystems, thrusters and the CRIS and SWIMS instruments (see figures 6.6.1 - 2 to 6.6.1- 5). The propulsion subsystem shall be located internal to the volume between decks and will be supported by the upper and lower decks.

The layout shall support the mass balance requirements given in the APL environmental specification 7345-9007.

6.6.2. Instrument FOV Requirements

The instrument field of view requirements are given in tables 5.7-1 and 6.6.2-1. The requirements are from the ACE Science Requirements Document (SRD) GFSC-410-ACE-002 Rev A - Change Notice #1 dated April 18, 1994. The observatory layout shall keep the instruments' field of view clear of blockage. Where blockage is unavoidable, it shall be negotiated with the responsible instrument experimenter. The solar panels and the magnetometer booms shall not impinge on these FOVs.

6.6.3. Alignment of Optical Axis

The alignment of sensor optical axes shall be accomplished by positioning the mounting hole locations on the spacecraft, with respect to the spacecraft coordinate system. The optical axes of the instrument sensors, sun sensor optical heads and star tracker shall be related to the instrument/sensor-mounting hole pattern. The instrument experimenter shall measure the instrument optical axis alignment with respect to the mounting hole pattern to verify the validity of the optical axis alignment.

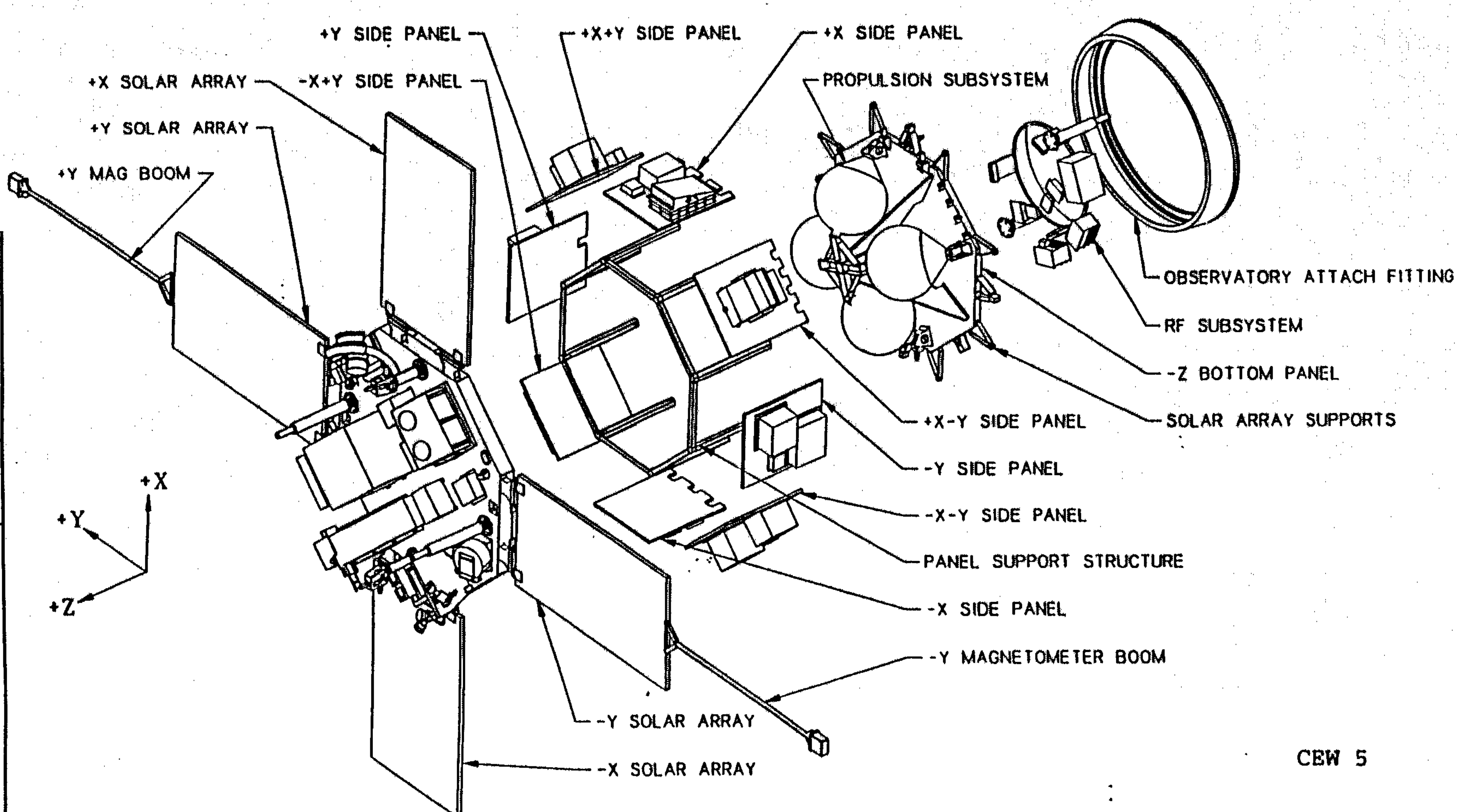

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Figure 6.6-1 Observatory - Exploded View

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Figure 6.6.1 - 1 Instrument (Upper) Deck

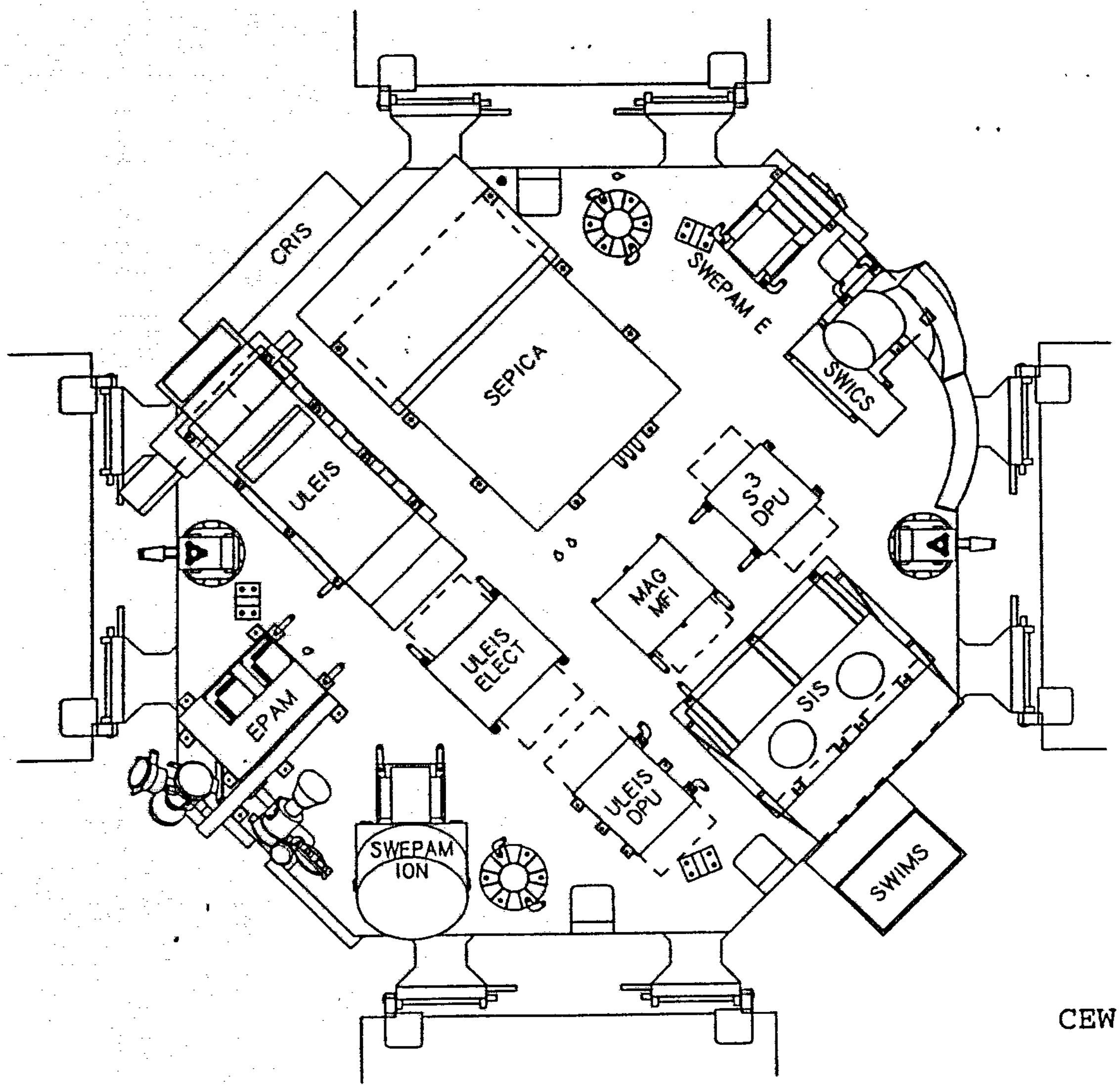

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Figure 6.6.1 - 2 Side Deck -X

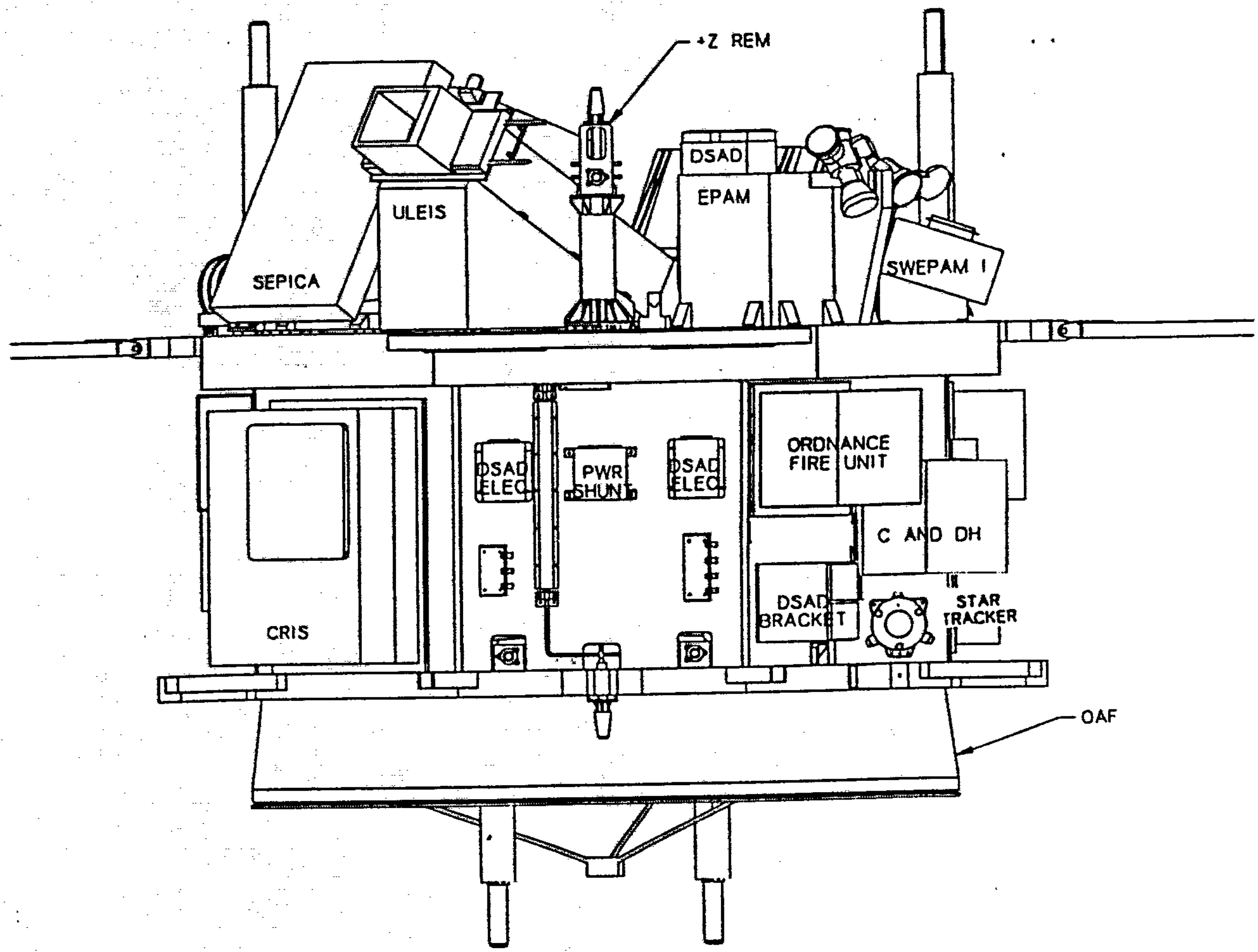

Figure 6.6.1 - 3 Side Deck +Y

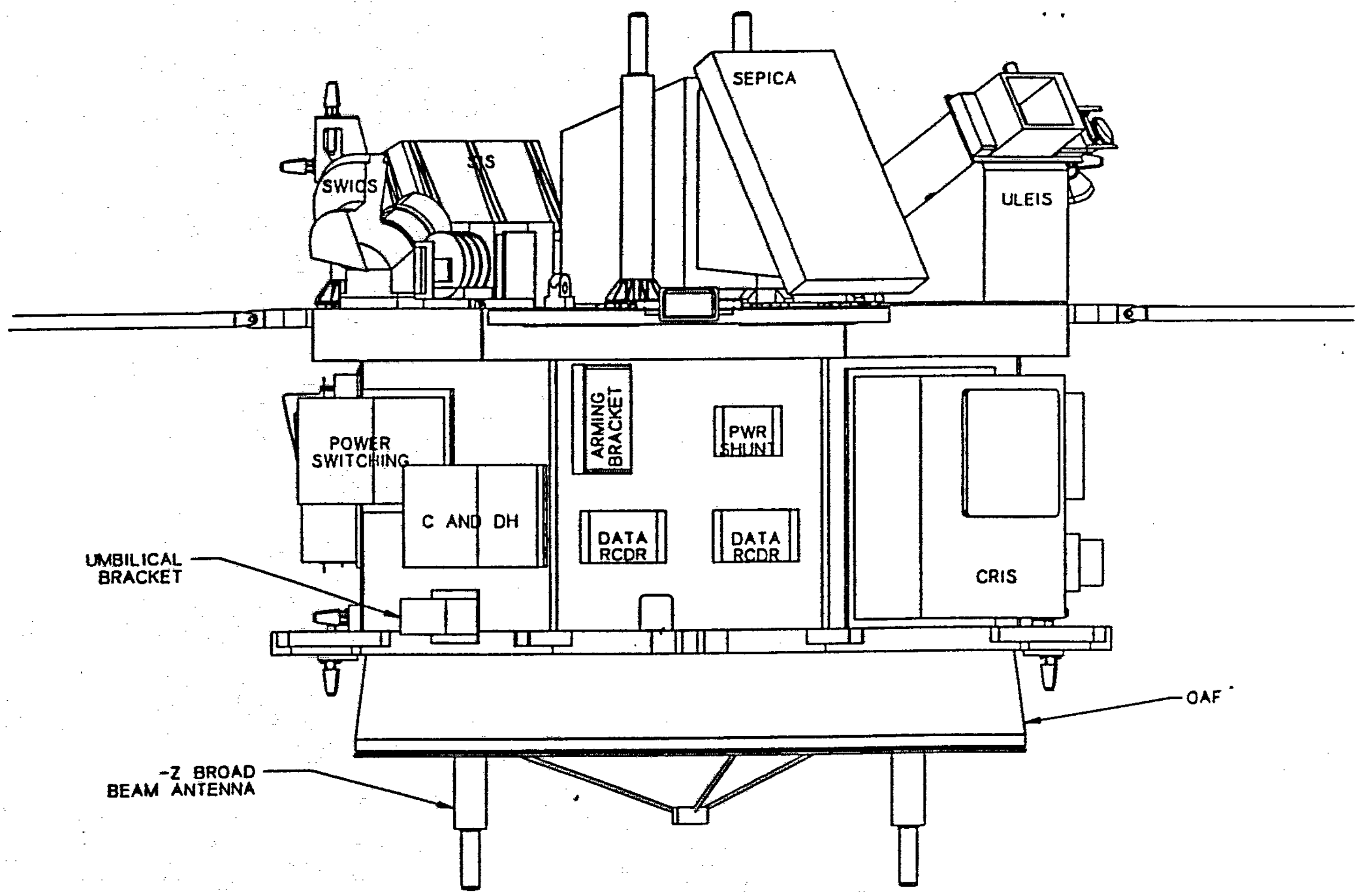

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Figure 6.6.1 - 4 Side Deck +X

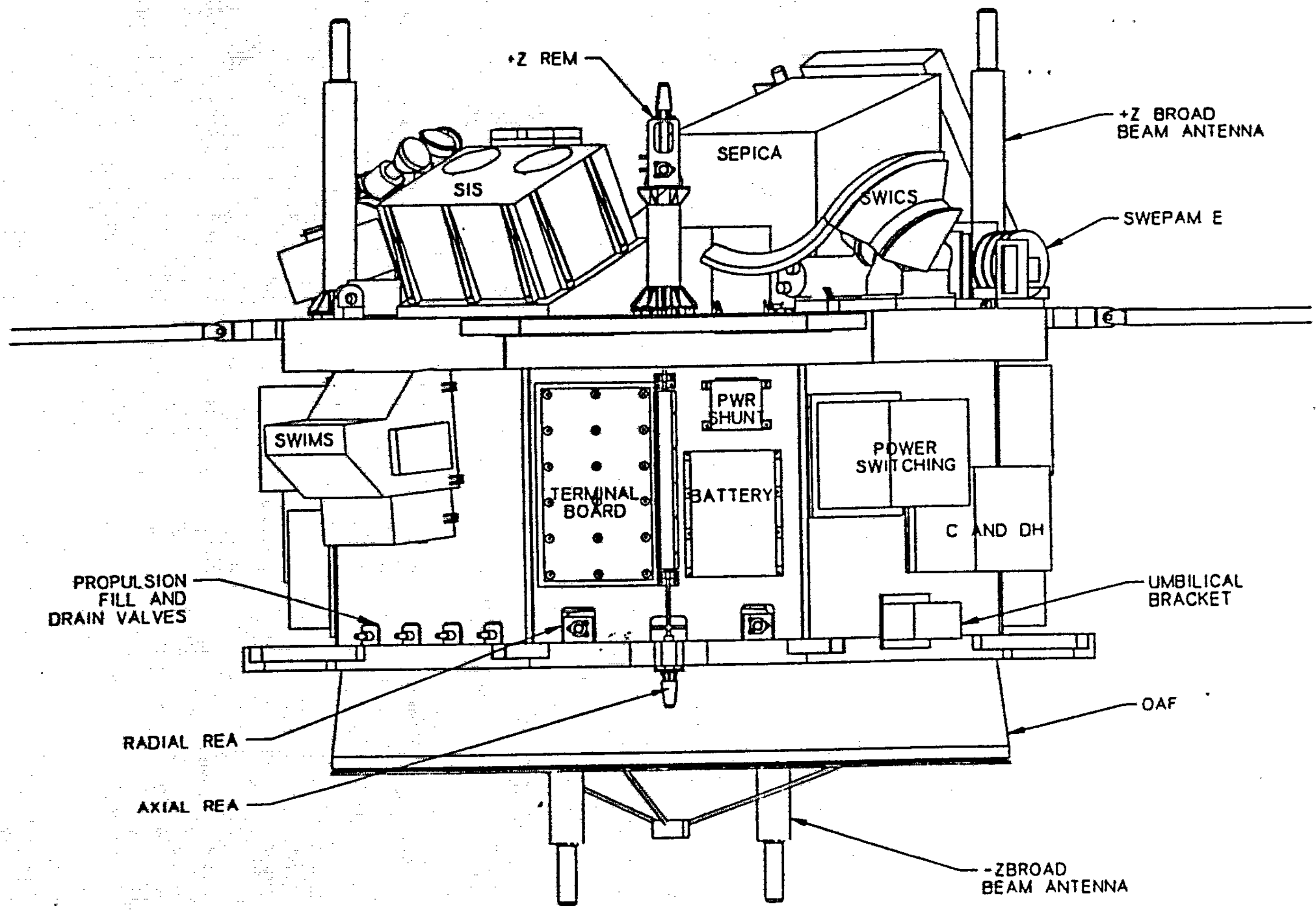

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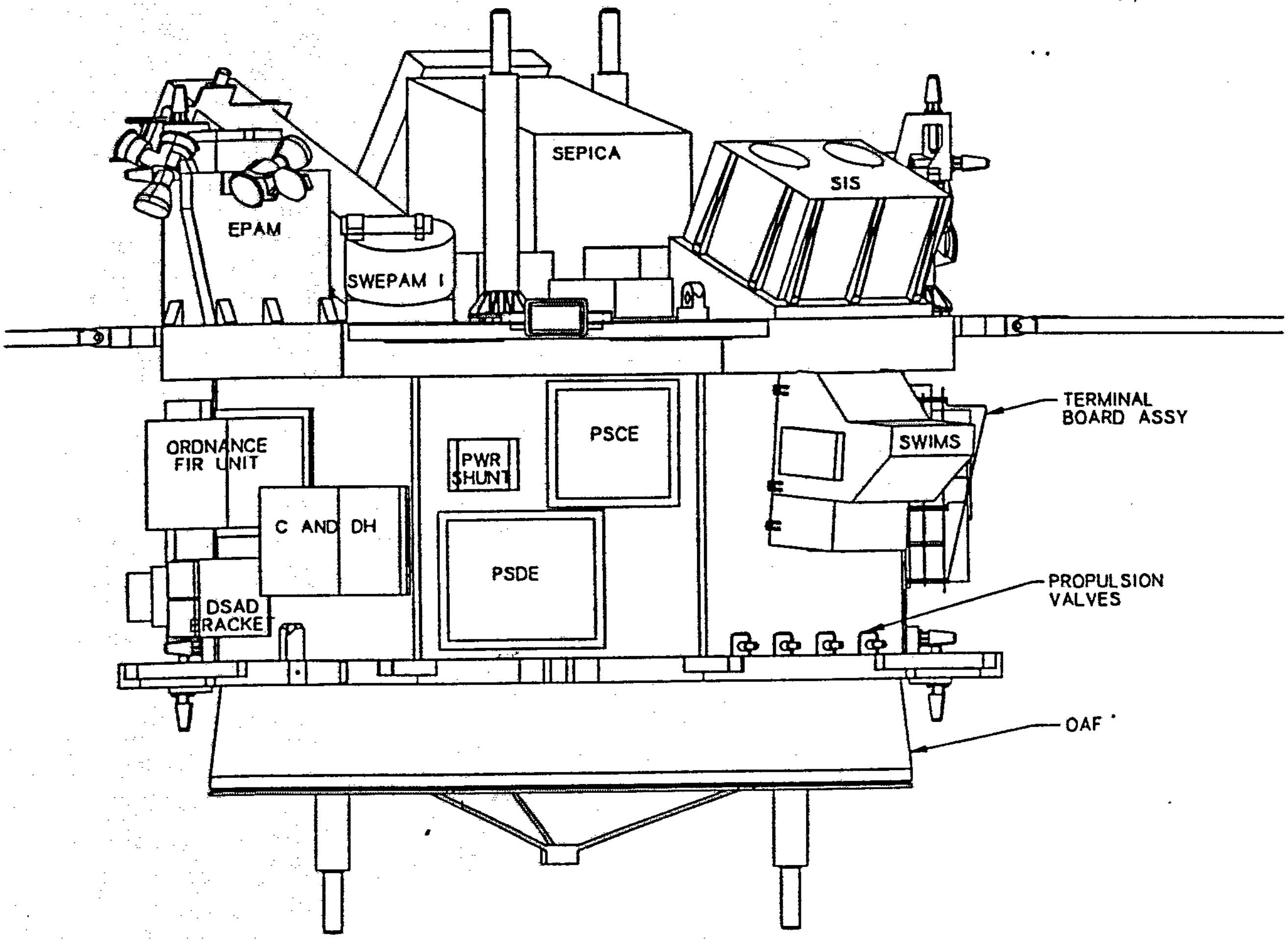
Figure 6.6.1 - 5 Side Deck - Y



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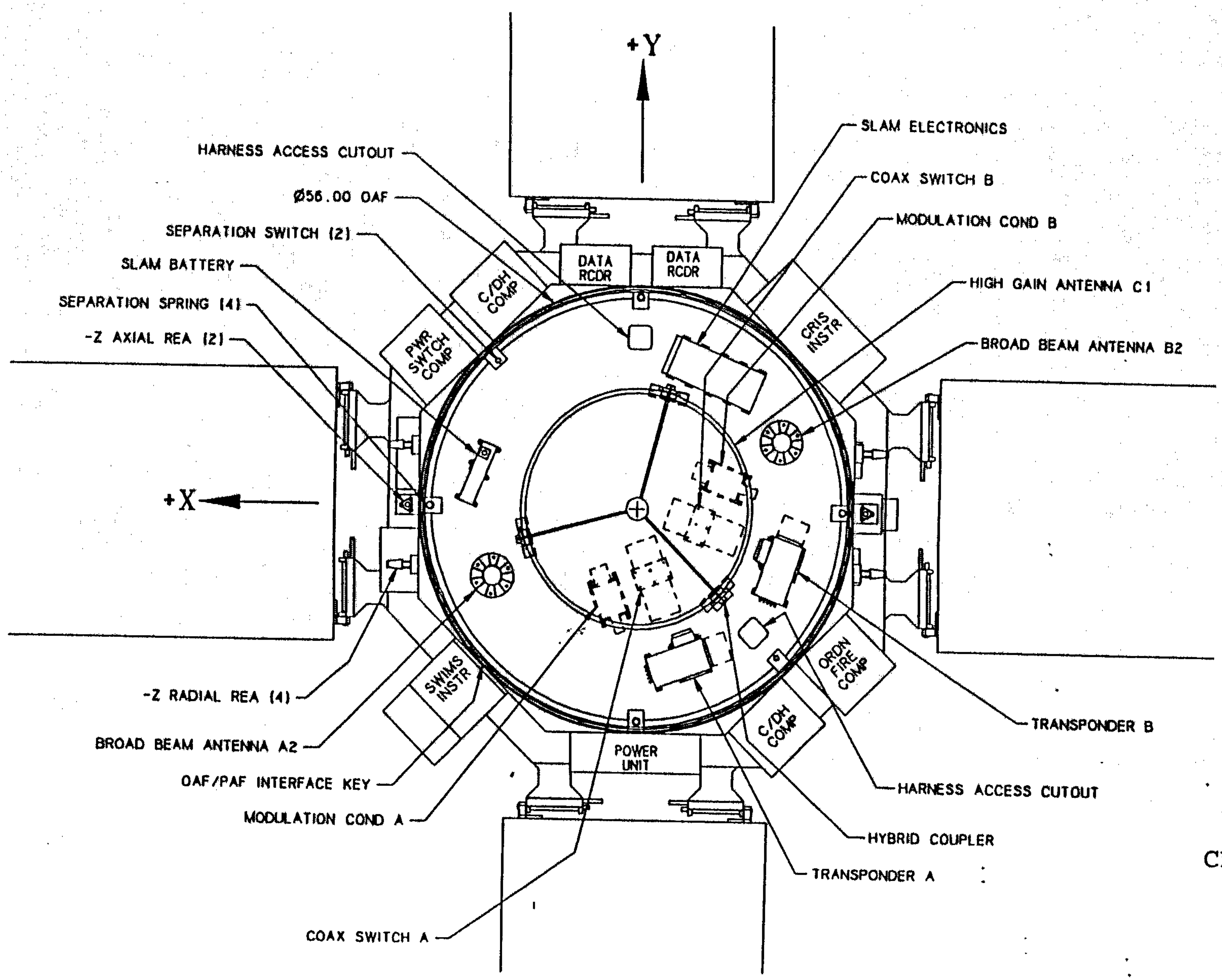

Figure 6.6.1 - 6 Lower Deck

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Table 6.6.2-1 **Instrument Pointing and Field of View**

Instrument	Primary FOV (degrees)	Desired Pointing Direction, (degrees)
CRIS	45 half cone	No Requirement
SIS	55 half cone	25+10,-5
SEPICA	80 x 35	60±2
ULEIS	24 x 20	60±1
SWIMS	62 x 60	0±1.25
SWICS	70 x 4.2	34.5 ±1.25
EPAM	51, 45, 53 53, 51, resp	30, 60, 150 60, 120 (all±1)
SWEPAM (Ions)	90x24	18.75±1
SWEPAM (Electrons)	160 x 30	90±1
MAGNETOMETER		One of three orthogonal axes parallel to S/C spin axis

NOTES:
1) Pointing angles measured from the spin axis lying in the sunward hemisphere.
2) Pointing direction knowledge requirements should be interpreted as tolerance limits, not 1σ uncertainties.

6.6.4. Magnetometer Booms

The two magnetometer sensors shall be located at the end of two titanium booms, deployed off the ends of the +Y and -Y axis solar panels. The boom length shall be 4.2 meters from the center of the spacecraft.

6.6.5. Nitrogen Purge

Low pressure nitrogen purge shall be provided for selected instruments as detailed in the GIIS and SIISs. A constant pressure supply line shall be provided for the instrument purge lines which will be an integral part of the spacecraft structure and flown with the structure.

The instrument experimenter shall be responsible for the design of the purge system within their instrument, including the determination of the purge gas flow rate as detailed in the GIIS.

6.6.6. Structural Models

Analytical structural models shall be developed for the spacecraft and the observatory. The observatory structural model will be used as input to coupling analyses for the launch vehicle/observatory interface to predict launch induced accelerations. The results of the coupling analyses may be used to further refine component and observatory level testing requirements given in the APL environmental specification 7345-9007.

6.6.7. Materials

Observatory structural materials shall be selected on the criteria contained in the JHU/APL Spacecraft Assurance Implementation Plan (7345-9100) and with consideration of the magnetic goals given in APL environmental specification 7345-9007.

For the structure subsystem, specifically, these shall be either corrosion resistant or treated to resist corrosion. Protective methods and materials for cleaning, surface treatment and application of finishes and protective coatings shall be followed as per PAIP, except where these are inconsistent with instrument requirements. They shall be chosen based on low outgassing weight loss and low volatile condensable material as per PAIP. Material strengths and other mechanical and physical properties shall be selected from authorized sources and shall be appropriate to the loading conditions, design environments, and stress states for each structural member. Where values for mechanical properties are not available, these shall be determined by approved test methods.

6.7. THERMAL SUBSYSTEM

The observatory will be thermally stable at the L_1 point since there is very little variation in the thermal environment. Observatory heat flow is shown in figure 6.7-1. The main cause of any thermal perturbations will be internal power fluctuations. Observatory deck temperatures shall be controlled with active control and passive control. The observatory thermal design support spacecraft and instrument subsystem requirements during all phases of the mission given in section 1.4.

The thermal subsystem shall maintain the temperatures of the spacecraft components and the instrument interfaces (as defined in the GIIS) at the temperatures given in the APL environmental specification 7345-9007 and the instrument SIISs.

Observatory thermal control methods are shown in Figure 6.7-2. Heat shall be removed from components by conduction to the spacecraft structure and/or radiated directly to space via localized radiators, as required. Thermal blankets shall cover all external surfaces except for radiators, antennas, thrusters and sensor apertures. Components may be either conductively tied to the deck or thermally isolated.

6.7.1. Thermal Modes of Operation

The following modes of operation shall be accommodated in the thermal design for spacecraft and instrument components:

- Survival Mode
  - the widest temperature range that each component can undergo in an unpowered state without damage or performance degradation. Survival conditions exist when all instruments and all non-critical spacecraft power loads (see section 5.6-1) are turned off. (Launch and part of cruise phase will have survival conditions.) Turn-on limits are defined as the widest temperature range within which each component must be capable of turning on without damage or performance degradation of any kind. The component is not necessarily required to operate within design specifications at the turn-on limits. However, once the instrument is within the specified operating limits, full compliance with design specifications is required. In most cases, the turn-on temperature range should be the same as the survival temperature range.

- Operational Mode
  - the temperature range within which each component must meet its operating specifications. Operational conditions include one or more instruments being turned off, but spacecraft non-critical loads are powered. For cases in which one or more instruments are powered off, while the spacecraft is still in operational mode, instrument interface temperatures will be maintained within operational limits by interface heaters located on the deck. Under these circumstances, the observatory is still defined as being in operational mode.

The spacecraft shall be responsible for the transition of interface temperatures from survival to operational mode. The instrument experimenter shall be responsible for transitioning the instrument components from survival to turn-on, given that the interface temperatures shall be within the design/test range specified in the SIIS.

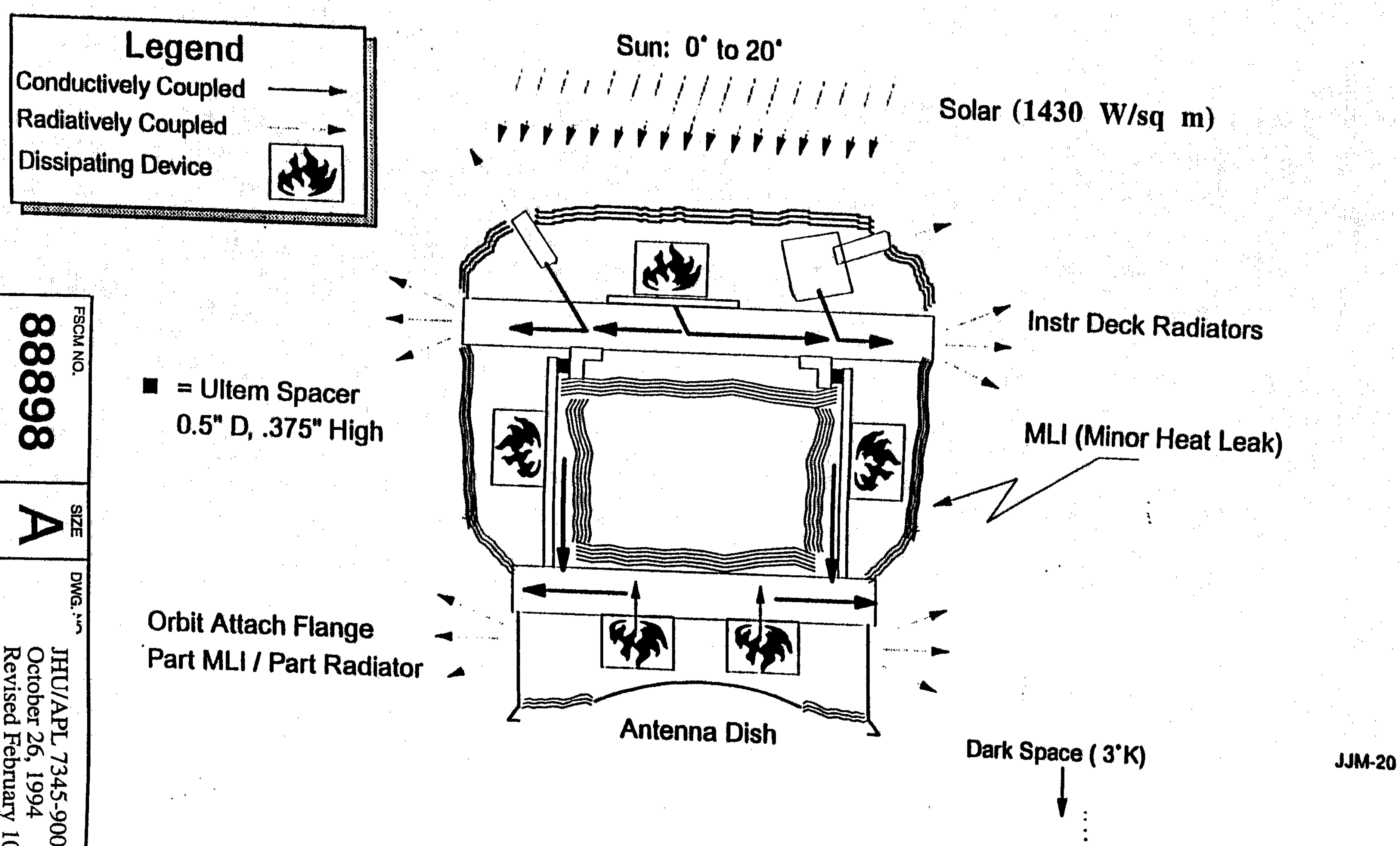

Figure 6.7-1 Observatory Heat Flow

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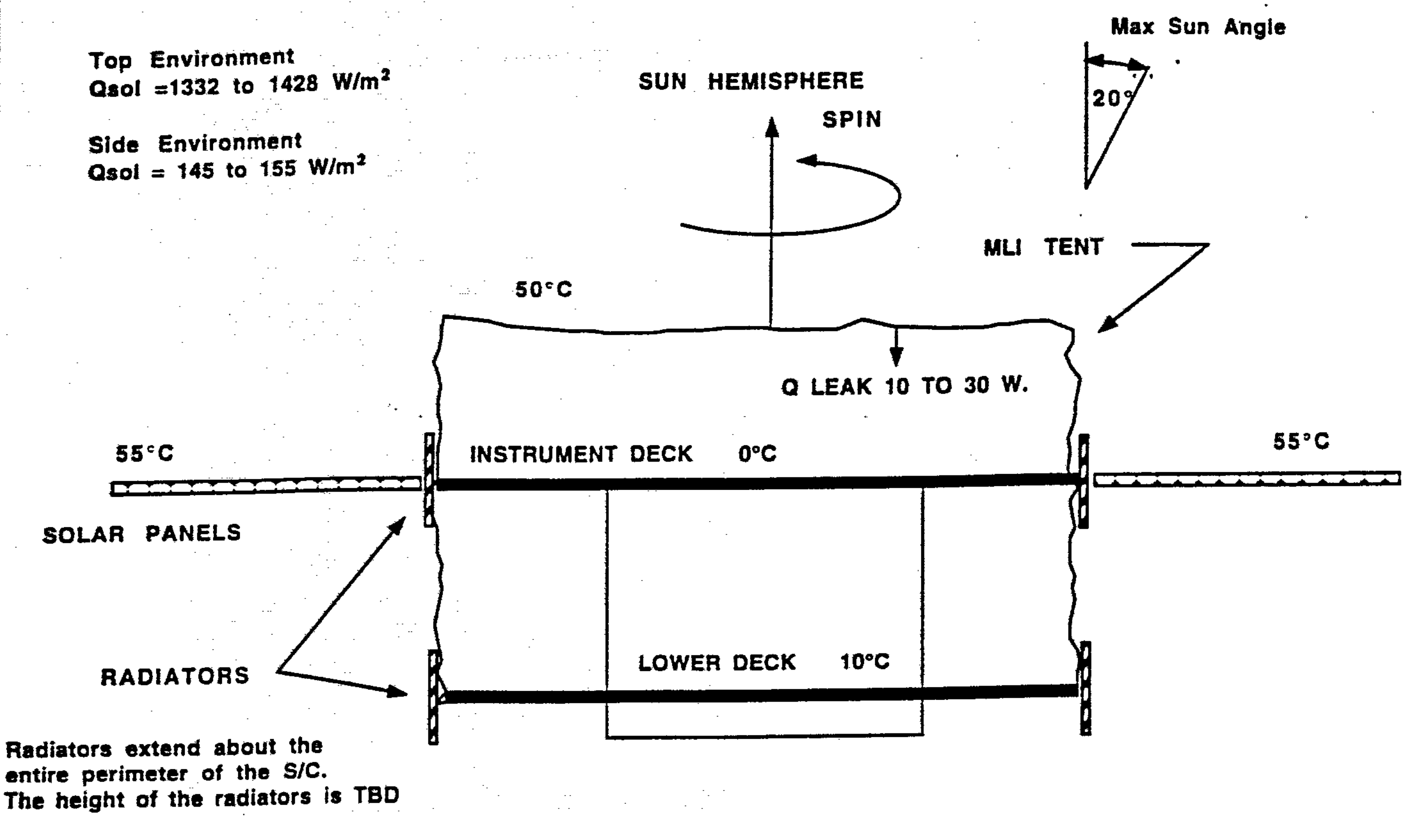
Figure 6.7-2 Observatory Thermal Control Methods



RADIATOR EFFECTIVENESS (W/m²)

S/C Temp.	Solar Array Sides	Open Sides
-20°c	108.8	150.9
0°c	196.3	221.2
+20°c	305.3	308.6

RADIATOR SIZE (TBD)

		Solar Array Sides	Open Sides
S/C Power	216.4W.	7"	8"
Instr. Power	117.4W.	4"	4"

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6.7.2. Thermal Control Methods

Thermal control will be both active (heaters and heater control electronics) and passive (thermal blankets, insulators, radiators, surface treatments). Heaters and heater control electronics will be used to control the deck temperatures for the operational and survival modes. For instruments, the choice of thermal control method shall be made by the spacecraft thermal engineer in conjunction with the instrument/sensor thermal engineer. For all methods of thermal control, the instrument thermal engineer shall take into account the allowable deck temperatures on the spacecraft side of the thermal interfaces.

- Local Thermal Control
  - For components conductively isolated from the spacecraft. Thermal dissipation is radiated directly from the external component surfaces to space. All non-radiator surfaces shall be radiatively isolated from the spacecraft and other components. A thermal interface resistance of ≥20°C/W is required (shown by analysis). Placement of local radiators, if required to remove localized heat, shall be coordinated with the spacecraft thermal engineer.

- Central Thermal Control
  - Thermal dissipation is conducted from the component to the observatory deck for eventual rejection to space. Candidates for central control include support electronics that have no apertures and which can be thermally decoupled from the sensor sections that require extended temperature ranges. Use of central thermal control shall require the approval of the spacecraft thermal engineer.

- Hybrid Thermal Control
  - Thermal dissipation is removed from the component via a combination of radiation to the environment and conduction to the observatory deck. The use of hybrid thermal control is undesirable because of analysis difficulties. Components desiring hybrid control shall demonstrate the necessity of its use and closely cooperate with spacecraft thermal engineer.

6.7.2.1. Passive Control - Radiators

The spacecraft decks shall be passively controlled as follows:

- Upper deck
  - heat shall be rejected via radiators which are attached to all eight edges of the octagon. The radiators face radially away from the axis of symmetry of the spacecraft.

- Side decks
  - heat shall be conducted along the side decks (via doublers when necessary) to the lower deck radiators. Heat exchange between individual side panels and space shall be minimized by enclosing all of the panels in thermal insulation blankets that extend from the bottom of the upper (instrument) deck radiators to the beginning of the lower deck orbital attach fitting. For the instruments mounted on the side panels, the insulation blankets will serve to radiatively isolate the instrument from the side panel. Instrument sensor, sun sensor, and Star Tracker apertures will not be covered with MLI.

- Lower deck
  - the observatory attach fitting is bolted to the lower deck and shall be used as a thermal radiator. Thermal blankets will closeout the areas around the exposed parabolic antenna.

6.7.2.2. Active Control - Heaters

Three classes of heaters shall be used, interface, operational, and survival as follows:

- Interface heaters
  - augment deck operational heaters when one or more components are turned off. The interface heaters replace some of the dissipation lost when a component is turned off, thereby reducing the overall observatory load variations. (The interface heaters are of primary importance to the instruments that utilize central thermal control. Instruments utilizing local thermal control are less affected by changes in deck temperature because of low thermal conduction in the interface.)

- Spacecraft Operational Heaters
  - support internal temperature requirements during normal component operation. The heaters shall be used to maintain the spacecraft temperatures above minimum limits and compensate for environmental changes such as variation due to sun angle, differences between BOL and EOL blanket properties and varying blanket efficiencies.

- Spacecraft survival heaters
  - support spacecraft temperatures when all instrument and non-critical spacecraft components are turned off.

- Instrument operational heaters
  - support operational heaters are intended to support internal temperature requirements and can be used during normal instrument operation.

- Instrument survival heaters
  - support internal temperatures during periods in which the instrument is unpowered.

6.7.3. Thermal Math Model

An analytical thermal model of the observatory shall be developed. Thermal models from the instruments shall be provided by the instrument experimenter for inclusion in the overall observatory model. After integration with the spacecraft thermal model, the reduced instrument thermal models will be used by the spacecraft thermal engineer to predict instrument temperatures. Therefore, the reduced models should correlate well with the detailed instrument thermal model in the areas of heat transfer across the spacecraft interface, heat transfer with the space environment, and critical component temperatures. Detailed instrument thermal models should be validated during some phase of thermal vacuum testing. If an instrument is not subjected to qualification testing in vacuum, then, the interfaces cannot be validated until the observatory level thermal vacuum test. At that point, each instrument risks the possibility that its interface will differ significantly from that of the thermal model. Since the instrument models then defines thermal control requirements of the instrument deck and surrounding surfaces, a significant error in the models could limit the thermal control available during flight.

6.7.4. Thermal Math Model Validation

The thermal math model shall be validated by a correlation of the thermal balance test results with the temperature predictions made for each configuration. The thermal balance test shall include the worst case sun aspect angles and observatory powered configurations as given in the APL environmental specification 7345-9007.

7.0 SPACECRAFT LOADS AND ACOUSTICS MEASUREMENT (SLAM)

The SLAM package is a microprocessor based hardware designed to determine the structural accelerations and vibroacoustic levels that the observatory will be subjected to during the first 5 minutes of launch. SLAM hardware will read analog signals from the 18 transducers located at strategic places on the observatory, digitize these and modulate on the S-band downlink. The same frequency as for the ACE mission will be used.

SLAM shall be implemented on a non-interference bases to the main mission. SLAM will be turned on approximately 30 seconds prior to launch through the umbilical connector and shall turn off, without ground intervention, 7 minutes after launch. The test requirements for SLAM are given in the environmental and test specification APL document 7345-9007.

The SLAM package consists of:

- 18 transducers (9 low frequency accelerometers, 6 high frequency accelerometers, 3 microphones).
- One electronics box containing the required electronics and transmitter
- Battery (silver zinc).
- RF antenna (will be pulled away when fairing separates)

Interfaces to the spacecraft are as follows and shall be detailed in the SLAM ICD.

- mechanical - location of the accelerometers
- mechanical mounting for the electronics package and the battery
- RF connector
- umbilical

8.0 SPACECRAFT GROUND SUPPORT SYSTEM (GSS)

The spacecraft GSS shall perform the functions required to test, evaluate, calibrate, measure, simulate, and other associated functions for the observatory. These functions shall be performed at the APL/JHU facility, at GSFC test site, and at the ETR during launch site operations as described in the APL environmental specification 7345-9007 and the I&T Test Plan and Procedures. The GSS will be composed of three major elements;

- RF GSE
- Power subsystem GSE (primarily the solar array simulator)
- Interface and Test Operation Control Center (ITOCC)

The GSS shall perform the following functions. Figure 8.0-1 is a block diagram of the GSS.

- Power the spacecraft prior to Power Subsystem Integration
- Command spacecraft (with and without RF subsystem and GSE)
- Provide run states (automated command files)
- Receive, decode and archive telemetry
- Monitor performance of observatory
- Provide multiple data displays
- Check housekeeping telemetry for limit violations and identify failure
- Interface to MOC for compatibility testing
- Support CRIS detector bias requirements
- Provide for off-line analysis of telemetry data
- Provide serial interfaces for inputs from instrument GSEs to synchronize instrument testing with the instrument stimulus configuration.
- Provide full spacecraft telemetry stream to each instrument GSE

The instrument GSE is the responsibility of the instrument experimenter and shall be GFE to the spacecraft. Instrument GSE requirements that relate to the spacecraft shall be defined in the SIIS. The GSE shall not interface directly with any spacecraft subsystem. Any GSE interface with the instrument shall be buffered and not have any impact on the spacecraft interfaces. Figure 8.0-2 is a block diagram of the instrument GSE interfaces.

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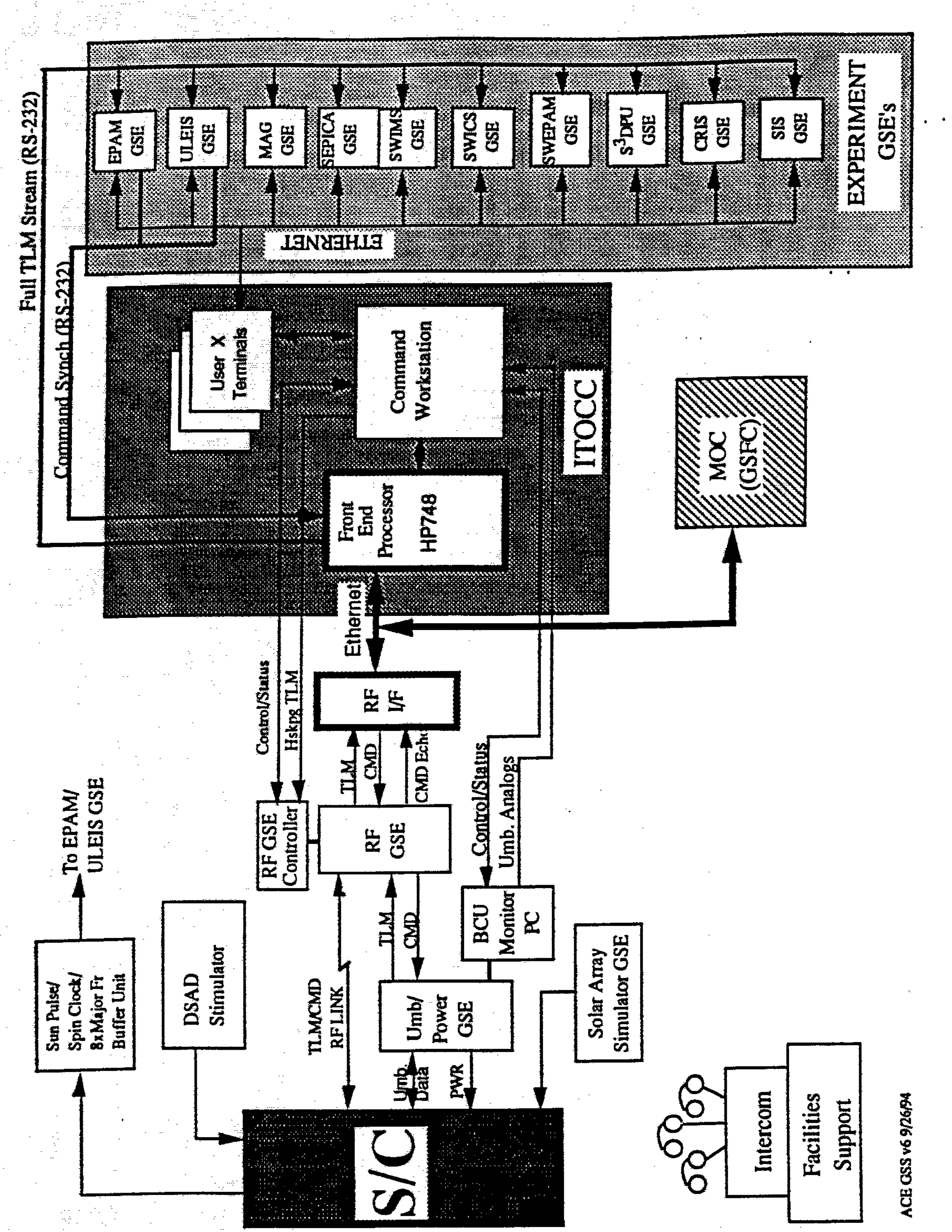

Figure 8.0 - 1 Spacecraft GSS Block Diagram

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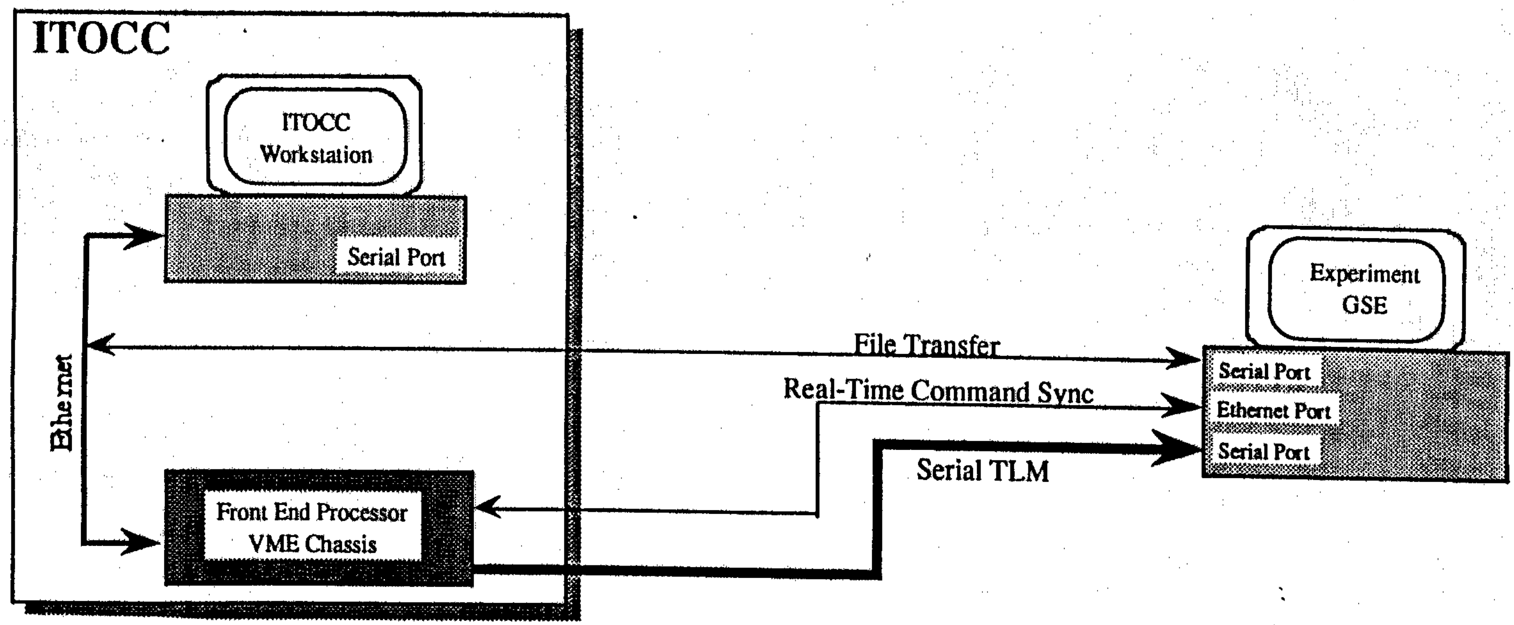

- Required: Serial Telemetry Interface - RS-232
- Optional: Real Time Command Interface - RS-232
- Optional: File Transfer - Ethernet (FTP)

Instrument GSE Interfaces

Figure 8.0 - 2

ACRONYMS LIST

ACE	Advanced Composition Explorer
ACR	Anomalous Cosmic Ray
ACS	Attitude Control System
ADC	Analog to Digital Conversion
ADS	Analysis Data Sites
AD&C	Attitude Determination and Control
A-h	Ampere-hour
AL	Anomalously Large (Solar particle events)
ALS	Airlift Solid
AM	Amplitude Modulation
AMPTE	Active Magnetosphere Particle Tracer Explorer
APL	Applied Physics Laboratory
ASC	ACE Science Center
ASG	Azimuth Signal Generator
ASOC	ACE Science Operations Center
BASD	Ball Aerospace Systems Division
BCR	Battery Charge Regulator
BCU	Blockhouse Control Unit
BNL	Brookhaven National Laboratory
BOL	Beginning of Life
BPS	Bits per Second
BSF	Back Surface Field
CA	Composition Aperture
CAMAC	Computer Automated Measurement and Control
Caltech	California Institute of Technology
CCAFS	Cape Canaveral Air Force Station
CCD	Charge Coupled Device
CCE	Charge Composition Explorer
CCSDS	Consultative Committee for Space Data Systems
CDF	Common Data Format
C&DH	Command and Data Handling
CDOS	Customer Data Operations System
CDR	Critical Design Review
CEM	Channel Electron Multipliers
CG	Center of Gravity
CHI	University of Chicago
CID	Charge Injection Device
CIR	Corotating Interaction Region
CME	Coronal Mass Ejections
CMF	Command Management Facility
CMOS	Complementary Metal-Oxide Semiconductor
CRIS	Cosmic Ray Isotope Spectrometer
CSA	Charge Sensitive Amplifier
dB	decibel
dBc	dB relative to the carrier
dBi	dB relative to isotropic
DCF	Data Capture Facility

DEC	Digital Equipment Corporation
deg	Degree
ΔV	delta Velocity Change
DEP	Dedicated Experiment Processor
DET	Direct Energy Transfer
DOD	Depth of Discharge
DPF	Data Processing Facility
DPU	Data Processing Unit
DSAD	Digital Solar Attitude Detector
DSN	Deep Space Network
EAROM	Electrically Alterable Read Only Memory
EHIC	Energetic Heavy Ion Composition
ELV	Expendable Launch Vehicle
EMC	Electromagnetic Capability
EOL	End of Life
EOM	End of Media
EPAM	Electron, Proton, and Alpha-particle Monitor
ESA	European Space Agency or Electrostatic Analyzers
ESD	Electrostatic Discharge
ESP	Energetic Storm Particles
ETR	Eastern Test Range
EXP	Experiment
F/D	Fill/Drain
FDF	Flight Dynamics Facility
FIFO	First In First Out
FIP	First Ionization Potential
FOV	Field of View
FPGA	Field Programmable Gate Array
FSS	Fine Sun Sensor
F/V	Fill/Vent
FWHM	Full Width at Half Maximum
GB	Gigabytes
GCR	Galactic Cosmic Ray
GEM	Graphite Epoxy Motor
GIIS	General Instument Interface Specification
GKS	Graphics Kernal System
GN_2	Gaseous Nitrogen
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
H	Hydrogen
HK	Housekeeping
HMRS	High-Mass Resolution Spectrometer
HVPS	High Voltage Power Supply
ICD	Interface Control Document/Drawing
I/F	Interface
IFC	In-Flight Calibrator
IMP	Interplanetary Monitoring Platform
ISEE	International Sun-Earth Explorer
ISM	Interstellar Medium
ISO/OSI	International Standards Organization/Open System Interconnect
ITOCC	Integration and Test Operations Control Center

IUS	Inertial Upper Stage
I-V	Current-Voltage
JHU/APL	The Johns Hopkins University/Applied Physics Laboratory
JPL	Jet Propulsion Laboratory
kbps	Kilobits per Second
keV	Kilo-electronvolt
kg	Kilogram
kHz	Kilohertz
L_1	Sun-Earth Libration Point
LANL	Los Alamos National Laboratory
LBL	Lawrence Berkeley Laboratory
LC	Launch Complex
lb_f	Pound force
LEFS	Low Energy Foil Spectrometer
LHC	Left Hand Circular Polarization
LISM	Local Interstellar Medium
LiD	Lithium-Drifted
LNA	Low Noise Amplifier
LRT	Last Resort Timer
LV	Launch Vehicle
LVBS	Launch Vehicle Booster Separation
LVPS	Low Voltage Power Supply
LZP	Level Zero Processing
MAG	Magnetometer
MB	Megabytes
MCP	Micro Channel Plate
MECO	Main Engine Cut Off
MHz	Megahertz
MeV	Mega-electronvolt
MLI	Multi-Layer Insulation
MLV	Multi-purpose Launch Vehicle
MΩ	Megohms
MOC	Mission Operations Center
MOM	Mission Operations Manager
MPE	Max-Planck-Institut fur Physik und Astrophysik
MRD	Mission Requirements Document
MSOCC	Multi-Satellite Operations Control Center
N_2H_4	Hydrazine
NAR	Non-Advocate Review
NASCOM	NASA Communication Network
NBS	National Bureau of Standards
NOAA	National Oceanographic and Atmospheric Administration
NOCC	Network Operations Control Center
NASA	National Aeronautics and Space Administration
NRL	Naval Research Laboratory
NRZ	Non-Return to Zero
NSCL	National Superconducting Cyclotron Laboratory
NSN	NASA Space Network
NSPAR	Non-Standard Part Approval Request
NSSDC	National Space Science Data Center
OAF	Observatory Attach Fitting

OBC	On-Board Computer
OCZ	Outer Convective Zone
OLS	Orbital Launch Sciences
Ω	Ohms
PAF	Payload Attach Fitting
PAIP	Product Assurance Implementation Plan
PAPS	Power Acceleration Power Supply
P/B	Playback
PC	Personal Computer
PCM	Pulse-Code Modulated
PCU	Power Control Unit
PDC	Production Data Center
PDF	Production Data Facility
PDR	Preliminary Design Review
PHA	Pulse-Height Analyzer
PI	Principal Investigator
PM	Proton Monitor or Phase Modulation
POCC	Project or Payload Operations Control Center
PRD	Program Requirements Document
PROM	Programmable Read Only Memory
PSD	Positive Sensitive Detectors
PSK	Phase Shift Keyed
PTM	Proof/Test Model
RAM	Random Access Memory
RDM	Radiation Design Margin
R_E	Earth Radius
RF	Radio Frequency
RHC	Right Hand Circularly Polarized
RMS	Root-Mean Square
ROM	Read Only Memory
rpm	Rotations Per Minute
R/T	Real Time
RTSW	Real Time Solar Wind
S/A	Solar Array
S/C	Spacecraft
SDPF	Sensor Data Processing Facility
SECO	Second Engine Cut Off
SEDA	Secondary Electron Detector Assembly
SEP	Solar Energetic Particle or Separation (as for Launch Vehicle)
SEPICA	Solar Energetic Particle Ionic Charge Analyzer
SEU	Single Event Upset
SFDU	Standard Formatted Data Units
SFU	Solar Flux Unit
SIIS	Specific Instrument Interface Specification
SIS	Solar Isotope Spectrometer
SRD	Science Requirements Document
SSD	Solid State Detector
SSR	Solid State Recorder
STE	Special Test Equipment
STS	Shuttle Transportation System
SW	Solar Wind

S^3DPU	Solar Wind Data Processing Unit
SWEPAM	Solar Wind Electron, Proton and Alpha Monitor
SWICS	Solar Wind Ion Composition Spectrometer
SWIMS	Solar Wind Ion Mass Spectrometer
SWT	Science Working Team
TAC	Time to Amplitude Converters
TBD	To Be Determined
TBR	To Be Reviewed
TLM	Telemetry
TOF	Time of Flight
TPF	Telemetry Processing Facility
T/R	Tape Recorder
TR	Transition Region
TV	Thermal Vacuum
U BERN	University of Bern
UH	Ultra Heavy
ULEIS	Ultra Low Energy Isotope Spectrometer
UMD	University of Maryland
UNH	University of New Hampshire
UV	Ultraviolet
VCTF	Virtual Channel Transfer Frame
VDC	Volts (Direct Current)
VLISM	Very Local Interstellar Medium
VSWR	Voltage Standing Wave Ratio
WAVE	Wide Angle Variable Energy
WG	Waveguide
WSA	Wedge and Strip Anode
XMTR	Transmitter
XPNDR	Transponder