Earth Radiation Budget Satellite

On-Orbit Reliability Investigation Final Report

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Earth Radiation Budget Satellite (ERBS) On-Orbit Reliability Investigation

Executive Summary

The research described in this report is the continuation of the *On-orbit Reliability Investigation* begun in 1993. This earlier investigation focused on determining the feasibility of developing a Goddard Space Flight Center (GSFC) reliability data base that would facilitate reliability analysis based on actual satellite operational experience. It resulted in a significant amount of detailed information related to a large number of spacecraft. However, the investigation was conducted in a very short period of time, and more data was acquired on some spacecraft than on others. Also, some key information had to be determined by deductive methods and engineering judgment, resulting in a greater chance for error. Therefore, the next logical step in this research was to demonstrate the feasibility of acquiring data with increased ***depth*** and ***fidelity***.

The goals of this continued research were to:

- Demonstrate the ***depth*** to which satellite data can be acquired for a single spacecraft.
- Demonstrate the ***fidelity*** to which data can be acquired for a single spacecraft.
- Provide a foundation and a model on which to refine the data for other spacecraft.
- Determine the feasibility of acquiring and analyzing data by manufacturer.
- Determine the feasibility of calculating on-orbit failure rates at the piece part level.

All of the goals of this research have been achieved. In-depth ERBS data was acquired, increasing the reliability data records by 629%. Fidelity in the characterization of the ERBS component relationships, redundancies and duty cycles was demonstrated in 44 pages of detailed functional descriptions, and reliability characterization. Manufacturers were provided for all of ERBS major components, providing a valuable link between mission failures and component providers. Finally, the feasibility of acquiring part-level on-orbit data was explored and found to be easily achievable. This research has provided a model for future in-depth reliability data acquisition for other Goddard spacecraft.

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20. Drawing number 70759, *Schematic Diagram, Battery Monitor, ERBS PMU*, Prepared by Ball Aerospace Systems Division.

21. *Mission Operations Document, Rev B*, May 1985, Prepared by Ball Aerospace Systems Division.

22. Numerous e-mail "interviews" with Goddard Space Flight Center and Ball Aerospace Employees.

Acronyms

Abbreviation	Meaning
AHMU	Amp-Hour Meter Unit
ACDS	Attitude Control and Determination Subsystem
ACE	Attitude Control Electronics
C/D	Charge/Discharge
CDR	Charge/Discharge Ratio
C&DH	Command and Data Handling
CDU	Command Decoder Unit
CSM	Command Storage Memory
CSB	Control System Bus
DTU	Digital Telemetry Unit
DAC	Digital to Analog Converter
DC	Duty Cycle
ERBS	Earth Radiation Budget Satellite
ESSA	Electronically Steerable Array Antenna
ENG	Engineering
GSFC	Goddard Space Flight Center
IRU	Interial Reference Unit
MCE	Magnetic Control Electronics
MCS	Magnetic Control System
MMS	Modular Multimission Spacecraft
MW	Momentum Wheel
NEB	Non-Essential Bus
NEI	Non-Explosive Intiator
NPM	Number of Components Provided
NRN	Number of Components Required
OAPS	Orbital Adjustment Propulsion System
ORU	Orbital Replacement Unit
OIB	Orbiter Interface Box
PA	Power Amplifier
PCU	Power Control Unit
PMU	Power Monitor Unit
PSA	Power Supply Assembly
PWA	Printed Wiring Assembly
PWB	Printed Wiring Board
RF	Radio Frequency
RCS	Reaction Control System
RMS	Remote Manipulating System
SW	Scanwheels
STS	Space Transportation System
SOAR	Spacecraft Orbital Anomaly Report
SPRU	Standard Power Regulator Unit
SSP	Standard Switch Panel
TRU	Tape Recorder Unit

Abbreviation	Meaning
TLM	Telemetry
TDU	Telemetry Distribution Unit
TXCO	Temperature Controlled Crystal Oscillators
TDRSS	Tracking and Data Relay Satellite System
XMTR	Transmitter
VT	Voltage/Temperature

Introduction

Background

The research described in this report is the continuation of the *On-orbit Reliability Investigation* begun in 1993. This earlier investigation focused on determining the feasibility of developing a Goddard Space Flight Center (GSFC) reliability data base that would facilitate reliability analysis based on actual satellite operational experience. The rationale for creating this data base was twofold:

- Military Handbook 217 failure rates do not reflect GSFC on-orbit experience, and
- there *is* sufficient operational data available to document on-orbit operational experience.

A report[1] issued by Hernandez Engineering in 1993 proved conclusively that this goal was feasible and achievable, albeit requiring a large amount of research time. This initial investigation yielded a data base of 1,239 records relating to Goddard spacecraft and pertinent reliability data, such as redundancy levels and duty cycles. Operational time was summed for the years the spacecraft components had been in operation and failure data were then used to calculate failure rates.

This investigation resulted in a significant amount of detailed information related to a large number of spacecraft. However, the investigation was conducted in a very short period of time, and more data was acquired on some spacecraft than on others. Also, some key information had to be determined by deductive methods and engineering judgment, resulting in a greater chance for error. Therefore, the next logical step in this research is to demonstrate the feasibility of acquiring data with increased ***depth*** and ***fidelity***.

Scope and Focus

For this demonstration, we have chosen the Earth Radiation Budget Satellite (ERBS). This spacecraft was chosen primarily because of our immediate access to a large volume of engineering and operations documentation. Although this research is limited exclusively to ERBS, it is viewed as the first step in a process to acquire commensurate data for all Goddard spacecraft. With the achievement of this goal, credible reliability data will be easily available to serve as a resource for engineering analyses and spacecraft management.

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[1] See On-orbit Reliability Investigation Final Report, August 15, 1993 for a complete discussion and example data base.

Objectives

The goals of this continued research are to:

- Demonstrate the ***depth*** to which satellite data can be acquired for a single spacecraft.
- Demonstrate the ***fidelity*** to which data can be acquired for a single spacecraft.
- Provide a foundation and a model on which to refine the data for other spacecraft.
- Determine the feasibility of acquiring and analyzing data by manufacturer.
- Determine the feasibility of calculating on-orbit failure rates at the piece part level.

To accomplish these goals, we have compiled a large amount of ERBS data and developed a data base to manage and report failure rates.

Benefits

The evolving on-orbit reliability data system offers significant programmatic benefits to NASA. It will enable GSFC management to make better-informed decisions resulting in more cost-effective designs. Moreover, it will foster greater confidence in reliability predictions and analyses. Because the data is based on actual operational experience, rather than a more generic prediction, such as MIL-HDBK-217, the predictions will be more acceptable to Goddard managers. This data can be used for the following activities:

- Designing and assessing redundancy requirements.
- Deciding how to reconfigure failed orbital systems.
- Performing engineering trade studies.
- Troubleshooting on-orbit problems.
- Predicting ground spare parts required.
- Deciding design characteristics for Orbital Replacement Units (ORU's).
- Determining hardware failure modes and failure frequency.
- Refining unique GSFC reliability guidelines.
- Basing technology decisions on similar-technology analyses.

Technical Approach

General

The data acquisition order of precedence for this task was from the top down. ERBS subsystem and component data was compiled with the focus on failure rate generation at the component level. The term “component” is used here to include both components and subassemblies. This choice simplifies the data base while allowing operating time and failures to be grouped and analyzed with a higher fidelity.

Normal ERBS Operations

There are nine phases of ERBS operations: Prelaunch Configuration, Launch Phase, Bay Door Opening Phase, In-bay C/O Phase, Development Phase, Initial Freeflight Phase, Orbit Transfer Phase, On-orbit Initialization Phase, and Normal Operations. For the purpose of this research, we have focused primarily on the ERBS Normal Operations phase.

Part-Level On-Orbit Reliability

A goal of this ERBS research was to determine the feasibility of compiling part-level data, with the intent of calculating on-orbit part failure rates. We have determined that this goal is definitely feasible. Although, part-level data was not extracted from the documents that were reviewed for this task (beyond scope of task), we found this data easily available and could be compiled and analyzed using the same methodology we have devised for component data.

However, because anomaly data is usually not charged to the piece part level, the data would be mainly operating time, without failure data points; therefore statistical measures would necessarily be less precise, but still could provide some complementary data for in-depth analyses.

Moreover, part-level data is important to the goal of on-orbit reliability determination because part-level data can suggest the technology vintage and assist in characterizing the components for similarity analyses. This information then allows for precision in grouping similar technology for reliability calculations. Future research should support the compiling of all spacecraft data available, including part data.

Manufacturer/Supplier Failure Rates

Another goal of this research was to determine the feasibility of obtaining manufacturers for ERBS components. We were able to obtain manufacturer data on most of the major ERBS components. As this data is acquired for other spacecraft, failure rates can be assigned by manufacturer or supplier to support comparative analyses. It will provide managers and designers with a valuable tool to support prudent choices regarding spacecraft component selection, resulting, ultimately, in improved reliability for future missions. Table 1 below provides the manufacturers for major ERBS components.

Data Sources

The data sources for the original 1993 reliability investigation were the SOAR data base, and reports found primarily in the archived records at the Federal Records Retirement Center, Suitland, Maryland. Sufficient data were quickly acquired to develop a fully functional reliability data system for Goddard spacecraft. Spacecraft Failure Analysis Reports, Failure Review Board reports, project-specific anomaly reports, among other documentation, were used to compile 15 years of data for 29 spacecraft. These reports have proven to benefit Goddard managers, designers, and engineers for many years; and the data base developed from this documentation offers significant promise for simplifying and enhancing reliability analysis for GSFC.

Table 1. Component Manufacturers

Major Components	Manufacturer
Louver Assembly	Fairchild
Scanwheels	Ithaco
Magnetometer Probe	Schonstedt
Magnetic Control Electronics	Ithaco
Torqrods	Ithaco
Momentum Wheel	Bendix
Attitude Control Electronics	BASD
Sun Sensor	Adcole
Inertial Reference Unit	Northrop
Solar Panel	Spectrolab
Power Monitor Unit (PMU)	BASD
Stand Power Regulator (SPRU)	Gulton
Power Control Unit (PCU)	McDonnell
Amp Hour Meter	McDonnell
28V Regulator	Gulton
Main Battery, 50 Amp Hour	McDonnell
Transponder	Motorola
S-Band Amplifier	Loral
ESSA Antenna	BASD
ESSA Controller	BASD
Command Decoder Unit (CDU)	Gulton
Digital Telemetry Unit (DTU)	Gulton
Tape Recorder Unit (TRU)	Odetics
Telemetry Distribution Unit (TDU)	BASD
Oscillator	Gulton
Orbiter Interface Box	BASD
Propellant Pressure Tank	PSI
OAPS Components	Rocket Research
OAPS Assembly	Rocket Research

The sources used in this report (relating to ERBS data) are the SOAR data base, a large number of drawings, operations documents, and e-mail interviews with Goddard and Ball Aerospace employees.

Ground Rules

The following ground rules were used to perform this task:

- ERBS was the ***only*** spacecraft considered.
- Analysis was restricted to component/subassembly level.
- On-orbit part level prediction feasibility was assessed.
- Feasibility of acquiring manufacturer-related failure rates was investigated.
- Failure at the component level was defined as the inability of the component to perform 100% of its intended functions, even if the component is being operated in a degraded state.
- SOAR component codes were assigned to facilitate compiling aggregate component time.

Procedures

The procedures to accomplish this task were to:

- Design ERBS data base relational tables.
- Describe ERBS functions.
- Diagram ERBS subsystems to identify functional and reliability relationships.
- Review all available ERBS documents and drawings to populate the data base.

Statistics and Analysis Methods

The analysis methodology and statistics for this study effort are identical to the methods used for the earlier 1993 investigation. These methods are described below. Because ERBS is only one spacecraft, and as such, would provide an insufficient sample size for failure rate calculations, there is no intent to perform an independent analysis of ERBS, or to update the remaining spacecraft with this proof-of-concept data base.

The intent here is to demonstrate a capability to improve the ***depth*** and ***fidelity*** of data acquisition for <u>one</u> spacecraft to be used as a beginning for improving the spacecraft data for the entire reliability data system. With that in mind, a review of the Statistics and Analysis methods employed for the entire data system (all spacecraft and aggregate component time) is provided below.

Component Similarity. In the original plan[2] to accomplish the goal of acquiring spacecraft hardware failure rates, the Modular Multimission Spacecraft (MMS) was viewed as a standard model around which similar technology failure data could be aggregated. The plan's author envisioned that MMS-similar components could be aggregated, to provide sufficient component sample sizes and numbers of failures for meaningful statistical analyses. In this scheme, a component was deemed MMS-similar, without further qualification, if it was a standard MMS component. Non-MMS standard components were to be defined as "similar" when analysis revealed complexity, technology, and reliability levels "comparable" to the appropriate MMS component. Understanding technology similarity is important because we want to be certain we are grouping components that are of similar technology. As an extreme example, we wouldn't want to group 1950's vacuum tubes with 1980's integrated circuits.

In the actual performance of the 1993 investigation, we found that the similarity was much more subjective. Because of the limited research time allocated for conducting the research, adequate circuit descriptions and piece part data were not found to facilitate confident engineering judgment regarding technology similarity. As a result, grouping of like component data was achieved by assigning component codes, based on standard SOAR component codes. See Appendix 1 for a list of standard SOAR component codes.

However, the results of the ERBS reliability investigation does clearly demonstrate that, with detailed circuit and piece part information, technology vintage and complexity can be determined. As additional spacecraft are analyzed to the same degree that we have analyzed ERBS, we can improve the fidelity of the grouping of like technologies and, thereby, improve the confidence in the data.

ERBS is not an MMS-standard spacecraft; however, the technology is similar to MMS and can be grouped with standard MMS spacecraft components, as well as with "MMS-similar" components. Because other spacecraft technologies have *not* been analyzed to the degree to which we have analyzed ERBS, the SOAR standard component grouping will not be abandoned at this time. As more spacecraft are analyzed and technology is accurately characterized, we will transition to the MMS-similarity method of grouping components.

Component Active Operating Time. Component active operating time must be accrued for each component in order to determine mean *time* between failure, or its inverse, the failure rate. ***Active*** redundant component operating time is included in summing operating time. ***Standby*** redundant component time is not summed in calculating operating time. Figure 1 below offers a simplified example of how operating time is calculated.

[2] Plan for the Development of GSFC Spacecraft Hardware Failure Rates, July 1992, Hernandez Engineering, Inc.

Record Types: Failure Record, Standby Redundancy Operating Record (S) and Active Redundancy Operating Record (A).

- **For a <u>failure record</u>: TERMDATE - ANOMDATE or date of the analysis (record date) for orbiting spacecraft (entered as a negative number for active, 0 for standby)**
- **For an <u>operating record</u> (S): TERMDATE - LAUNCHDATE = spacecraft operating time; operating time * subsystem duty cycle * number required * component duty cycle =component operating time.**
- **For an <u>operating record</u> (A): TERMDATE - LAUNCHDATE = spacecraft operating time; operating time * subsystem duty cycle * number provided * component duty cycle= component operating.**

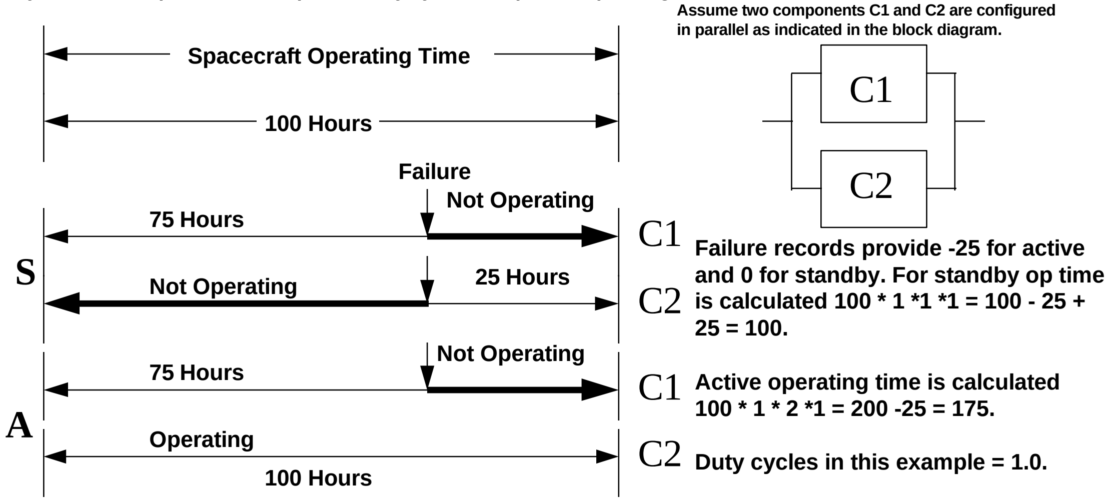

Figure 1. Example Operating Time Calculation

Essentially, the data base tables contain three types of records: Failure records; operating records (with ***Standby*** redundancy) and operating records (with ***Active*** redundancy). The failure record is used in the component operating time summation for a given component number as follows: The termination date (or the date of the report for currently orbiting spacecraft) is subtracted from the anomaly date. For ***actively*** redundant components (coded A in Typered field), the result of this calculation is entered as a negative number in the component operating time field of the failure record. Because the number of components provided is the same as the number that are ***active***, the "number of components provided" is used as a multiplier on the operating record.

For example, if the spacecraft operating time is 100 hours and there are two actively redundant components, as in the example in Figure 1 above, then 100 x 2 components will equal 200 operating hours for that component. We are assuming, in this example, that the subsystem and component duty cycles equal 1.0. If there is a failure of one of the components at 75 hours, then 25 hours must be subtracted to reflect the accurate component operating time summation. That is the reason for using a negative number in the failure record component operating time field; i.e., in the calculation of component operating time, the time from anomaly (failure) date to termination or report date is subtracted—when component operating time is calculated.

But the standby redundancy configuration is different. As you can see from the diagram in Figure 1, the time prior to failure does not result in operating time for the redundant component. The actual operating time is only the spacecraft operating time multiplied by 1—because there is never more than one component operating at a time. Again, we are assuming a duty cycle of 1.0. The way this condition is accounted for, in our computation methodology, is by entering a 0 for any record coded S (standby) instead of a negative number—as with active redundancy—since the time from anomaly date to termination date has only one component operating. And only one component was operating prior to that date.

After the component operating time has been calculated, a simple program is run to perform the statistical calculations and generate a report. A constant failure rate is assumed for all reliability calculations. The program calculates a point estimate by dividing the total number of random failures into the total active operating time for a given component classification. The Chi-square estimator is also used to calculate an estimated mean time to failure for component operating time. [3]

Operating hours was calculated for ERBS by using October 30, 1996 as the "termination" date. The operating hours were 105072. See appendix 6.

Report. An example of a typical report generated by this data system is found at Appendix 2 of this report. This report was based on a developmental data system and the results should be used for demonstration purposes only. As other spacecraft are subjected to the rigorous analysis that we have conducted for ERBS, the results will become more credible. Also, as spacecraft "projects" are retired, and operating and engineering data samples are increased, statistical analyses will continue to provide finer results.

Component Duty Cycles. The duty cycle is the ratio of operating time to on-orbit spacecraft time. This ratio is used to discriminate component active operating time from standby time. This factor is used to in calculating the operating time to be used in failure rate calculations. It is extremely important to obtaining credible failure rate results. However, it is quite often the most elusive data element to acquire. Such factors as orbit depth or solar flare activity can effect the density of the atmosphere and, therefore, affect the need for drag makeup maneuvers, and the commensurate duty cycles of ACDS components. The orbit depth band required by the principal investigators can increase or decrease the need for station-keeping actions.

Due to the complexity and dynamism of engineering and operations strategies, including undocumented functional redundancies, innovative operational work-arounds, and mission-unique science requirements, it impossible to achieve total

[3] See *Plan for the Development of GSFC Spacecraft Hardware Failures Rates*, July 1992 and *On-Orbit Reliability Investigation Final Report*, August 1993, both by Hernandez Engineering, Inc. for more in-depth discussions of analysis techniques.

accuracy. However, in the original 1993 investigation, as well as our investigation of ERBS, we have carefully considered the mission operations for each spacecraft and, when possible, have confirmed our interpretations with experienced designers and operators.

Primarily, we believe general duty cycle estimations based on nominal mission operations can provide relative accuracy for credible failure rate calculations. An example of the duty-cycle estimation logic applied to ERBS is provided in Table 2. Duty cycle is estimated by examining the frequency of routine events associated with ERBS, and then defining the major components that would be energized to perform the activity. Active components (energized for ERBS lifetime) will always have a duty cycle of 1.0. It is recognized that there are many unplanned events that also affect the duty cycle. However, this methodology offers an improvement over simply assuming a 1.0 duty cycle for all components. Note that the largest duty cycle is used as the primary factor for operating time calculations.

Anomalies. ERBS anomalies have involved the following components:

- Solar Arrays
- Sun Sensors
- Telemetry Encoders
- Transponders
- Command Decoders
- Electrical Motors
- Memory
- Earth Sensor
- Gyros
- IRU-1
- Batteries
- Command Storage Memory

A table reflecting the details related to these anomalies is provided at Appendix 3.

ERBS Data Base. The new ERBS reliability data base, developed for this task, is provided at Appendix 4. We were able to increase the ERBS component records from 24 in the original data base to 175 records in this new data base, a 629% increase. This achievement, combined with the **in-depth** functional description of ERBS spacecraft systems clearly demonstrates that precise reliability data can be acquired for Goddard spacecraft.

Table 2. Duty Cycle Estimation Methodology

Event[4]	Frequency	Components Affected	Duty Cycle Factors
SAGE-II Events DTU-2 Mode change to all experiment.	Twice per orbit for 6 minutes	DTU-2, CSM, TDU	1.0 for CSM 1.0 for one DTU-2 0.0 for one DTU-1
Load and Dump CSM's	Every 12 Hours for 10 minutes	Operational DTU[5], CSM	1.0 for one DTU 0.0 for one DTU
TRU-to-Transponder data out switching	Eight times per day	TRU's -1, -2, -3, -4	.25 for each TRU 1.0 for Transponder
Advance CSM loads.	12 hour memory loads for continuous CSM command executions	CSM	1.0 for CSM
Tape Recorder Dump	After four orbit record cycle = nominally 8 per day for 22 minutes each with potential 8 redumps.	Tape Recorders	1.0 for TRUs
ERBE Data Collection	Continuous Recording Required TRU-1 and TRU-2 for four orbits (6.5 hours each)	ERBE TRU's	1.0 for TRU's
ERBS Transponder Power Up when main ERBS bus enabled	Nominally powered throughout the mission.	Transponder Receivers	1.0 for Transponder Receivers.
ERBS Transponder Transmitter inhibit relays via NEB power.	Nominally powered throughout the mission.	Transponder Transmitters	1.0 for Transponder Transmitters
ESSA Antenna Communications	7 to 8 25-minute TDRSS MA events and one 15-minute	RF components ESSA antenna components.	(8 x 25 minute = 200 + 15 = 215 min./60/24=

[4] These events were extracted from Mission Operations documents.
[5] The backup configuration for the DTU allows all mission requirements to be met with one DTU.

Event[4]	Frequency	Components Affected	Duty Cycle Factors
	TDRSS MA event per day.		0.15 for RF and ESSA components.
Horizon Scanners Initiated	Normally on.	Horizon Scanners	1.0 for Horizon Scanners
ESSA antenna controller/driver selection	Nominally selected throughout the mission.	String-1	1.0 for string-1 0.0 for string-2
Orbit and Attitude maneuvers	As required by Principal Investigator.	ESSA Antenna not used for this. Transponder-1	0.0 for ESSA 1.0 for Transponder-1
Sun Sensor Functions	Normally on.	Sun Sensor	1.0 for Sun Sensors
Yaw Turn	Every 30 days	CSM Thrusters	1/30 day = .03 for Thrusters.
Power Amplifier powered	Always	Power Amplifier	1.0 for PA's
MCE Power On.	Always	MCE	1.0 for MCE
ACE Power On.	Always	ACE	1.0 for ACE
Temperature Control	All heaters enabled throughout mission except Catalyst Bed Heaters, which are enabled 2 hours prior to maneuvers[6]	All heaters	.30 for Catalyst Bed Heaters (average 30% dual element duty cycle), 1.0 for all other heaters
Pointing Control	Continuous	MCS components (primary system)	1.0 for MCS
Delta V	Every 3 to 6 months	RCS components	2-4 times per year
+X/-X & -X/+X Yaw Maneuvers	Every 30 days @ Beta=90 degrees	RCS Components	12 times per year-- combined = 14 times per year. Use 14 days as conservative estimate. ~14/365 days= .04 for RCS components.
Torqrods Enabled.	Always.	Torqrods	1.0 for Torqrods

[6] Reference Ball Aerospace Drawing # 63928, Sheet 2.

Earth Radiation Budget Satellite

General Description

Our first step in investigating reliability data for ERBS was to understand the design characteristics and functionality of the system. This understanding is demonstrated through the functional description provided at appendix 5 of this report.

The primary mission objectives of ERBS are:

- To determine the monthly average Earth radiation budget on regional, zonal, and global scales for a minimum of one year.
- To determine the equator-to-pole energy transport gradient.
- To determine the diurnal variation in the regional radiation budget on a monthly basis.
- To map vertical profiles of stratospheric aerosols, NO_2, O_3, from 70 degrees S to 70 degrees N latitude.
- To determine the seasonal variations in stratospheric aerosols, NO_2, and O_3.
- To identify sources of stratospheric aerosols, NO_2, and O_3.
- To define a baseline for investigating the effects of anthropogenetically and naturally induced changes on climate and environmental quality.

The ERBS spacecraft is a free-flying system designed to support instrument and mission requirements in space. Electrical power for the observatory, provided by a combination of solar arrays and batteries, is supplied to each instrument by means of commandable relays. Power is generated by solar arrays and stored in batteries. Propulsion to raise the observatory orbit from the Shuttle orbit to mission orbit was provided by hydrazine propulsion subsystem. Three-axis attitude control for the nadir-pointing mission operations, and for orbit adjust operations is provided by a momentum biased system coupled with electromagnetic torqrods. Thermal control for the spacecraft is provided by appropriate use of coatings, blankets, louvers, and heaters. Command and Data Handling (C&DH) with standard logic elements, timing and control, and data buffer storage techniques interface with Tracking and Data Relay Satellite System (TDRSS), and GSTDN, and conforms to the standards for those links. It requires a standard TDRSS transponder and a medium-gain steerable antenna. ERBS is composed of the following subsystems: Attitude Control and Determination, Communications and Data Handling, Power, Propulsion, Thermal, and Instruments.

Appendix 5 provides a thorough, detailed functional description of the ERBS subsystems. This level of detail is provided because it allows for a full understanding of the system functional interrelationships, imperative to an in-depth characterization of the reliability relationships inherent in the system. One element of data regarding the frequency of the occurrence of various operational functions has been extracted to provide duty cycle factors, as presented in table 2, earlier in this report. The functional description included in appendix 5 provides further context for evaluating duty cycle and redundancy assessments.

Conclusion and Recommendations

All of the goals of this research have been achieved. In-depth ERBS data was acquired increasing the reliability data records by 629%. Fidelity in the characterization of the ERBS component relationships, redundancies and duty cycles was demonstrated in 44 pages of detailed functional descriptions and reliability characterization. Manufacturers were provided for all of ERBS major components, providing a valuable link between mission failures and component providers. Finally, the feasibility of acquiring part-level on-orbit data was explored and found to be easily achievable.

This research has provided a model for future in-depth reliability data acquisition for other Goddard spacecraft. We recommend that the same disciplined approach, that we have applied to characterizing ERBS, be applied to all spacecraft that can be included in our reliability data system. It is not enough to acquire clear, accurate, and thorough data for only one spacecraft. The remaining inaccuracies in the other spacecraft could diminish the fidelity of the results of any aggregate calculations. We also recommend that part level data be pursued in any future research. With that goal in mind, we recommend that any anomaly or failure analyses that can charge failures to the part-level be included in the SOAR data system. This evolving research can provide significant benefits it is continued in a highly disciplined manner.

Appendix 1. SOAR Generic Component Codes

Component ID	Component Generic Name
100	**TC&C [C&DH]**
101	Accumulators
102	Command Decoders
103	Command Distribution Units
104	Computers
105	Demodulators
106	Memory
107	Oscillators
108	Receivers
109	Signal Conditioners
110	Software
111	Timers and Clocks
112	Voltage Controlled Oscillators
113	Digital Equipment
114	X-Ray Drive Release
200	**TLM & DH [C&DH]**
201	A/D, D/A Converters
202	Antenna Components
203	Commutators
204	Data Formatters
205	Data Handling Units
206	Diplexers
207	Filter Networks
208	Magnetic Tape Units
209	Multiplexers
210	Phase Modulators
211	Programmers
212	Subcarrier Oscillators
213	Telemetry Encoders
214	Transmitters, VHF
215	Transmitters, Doppler
216	Transmitters, FM
217	Transmitters, S-Band
218	Transmitters, Special Purpose
219	Transmitters, Tracking
220	Transmitters, Video
221	Transmitters, Wideband
222	Transmitters, Other
223	Transponders
224	Hybrid
225	Isolator
226	Yo-Yo System

Component ID	Component Generic Name
227	PCM CTL
228	STACC/STINT
229	RF Switch
300	**Thermal**
301	Heat Exchangers
302	Heat Pipes
303	Heaters
304	Louver Components
305	Thermal Control Electronics
306	Thermistors/Thermostats
400	**ACS**
401	Accelerometers
402	Attitude Control Components
403	Control Gas Components
404	Control Switching Components
405	Earth Sensor Components
406	Gyro Assembly Units
407	Gyros
408	Horizon Sensors
409	Infrared Scanners
410	Magnetometers
411	Momentum Wheel/Reaction Wheel Components
412	Nutation Dampers
413	Regulators, Pressure
414	Star Trackers
415	Sun Sensors
416	Torquing Coils
417	Valves
418	IRU
419	Computers
420	Gyro Power Supply
500	**Power**
501	Amplifiers, Power
502	Amplifiers, Other
503	Array Drive Electronics
504	Battery Charge/Discharge Control Units
505	Battery Packs
506	DC/AC Converters
507	DC/DC Converters
508	Fuel Cell Modules
509	Motors, Electrical
510	Power Distribution Units
511	Regulators, Voltage
512	Solar Arrays
513	Undervoltage Detectors/Control Circuits

Component ID	Component Generic Name
514	Intercon Harness
600	**Propulsion**
601	Apogee Boost/Kick Motors
602	Propulsion Module
603	Regulators, Pressure
604	Tanks
605	Thrusters, Cold Gas
606	Thrusters, Hydrazine
607	Thrusters, Bi Propellant
608	Valves
609	Filters
610	Line Heaters
700	**Instruments (unique)**
800	Structure
801	Primary
802	Secondary
900	**Other**
901	Bolometer Components
902	Detectors
903	Gear Trains
904	Magnetic Sensing Devices
905	Pneumatic Components
906	Radiometers
907	Sequencers
908	Vidicon Cameras

Appendix 2. Example Reliability Report

Component ID	Component Name	Sample Size	Total Hours	Total Failures	Point Estimate Failure Rate (Failures/10 6 Hours)	70% Chi Square Estimate (Failures/106 Hours)
102	Command Decoders	29	1212016	0	0	0.99421
103	Command Distribution Units	34	1623960	2	1.2316	2.22604
104	Computers	37	1383960	4	2.8903	4.2631
105	Demodulators	5	104568	0	0	11.5236
106	Memory	23	554016	0	0	2.17503
107	Oscillators	10	272400	2	7.3421	13.27093
108	Receivers	23	730872	1	1.3682	3.3385
109	Signal Conditioners	59	1754527	1	0.57	1.3907
111	Timers and Clocks	14	457080	0	0	2.6363
112	Voltage Controlled Oscillators	7	271272	0	0	4.44204
113	Digital Equipment	3	127536	0	0	9.44831
114	X-Ray Drive Release	2	56952	0	0	21.15817
200	TLM&DH	1	121368	0	0	9.92848
201	A/D, D/A Converters	1	114864	0	0	10.49067
202	Antenna Components	70	2895277	2	0.6908	1.24859
204	Data Formatters	5	227040	0	0	5.30743
205	Data Handling Units	29	1000248	2	1.9995	3.6141
206	Diplexers	20	667809	0	0	1.80441
207	Filter Networks	3	222696	0	0	5.41096

Component ID	Component Name	Sample Size	Total Hours	Total Failures	Point Estimate Failure Rate (Failures/10 6 Hours)	70% Chi Square Estimate (Failures/106 Hours)
208	Magnetic Tape Units	59	914266	7	7.6564	10.0627
209	Multiplexers	11	681864	0	0	1.76721
210	Phase Modulators	1	43032	0	0	28.00242
211	Programmers	3	111720	0	0	10.78589
212	Subcarrier Oscillators	1	62712	0	0	19.21482
213	Telemetry Encoders	51996	7104	10	10.029	12.4862
214	Transmitter, Beacon	7	270672	0	0	4.45188
217	Transmitters, S-Band	25	858381	1	1.165	2.8426
218	Transmitters, Special Purpose	5	207552	6	28.9084	39.0264
221	Transmitters, Wideband	2	164400	1	0.6893	1.682
222	Transmitters, Other	41	1450656	1	0.6893	1.682
223	Transponders	31	13876096	3	2.1644	3.4341
224	Hybrid	3	229488	0	0	5.25082
225	Isolator	6	229488	0	0	5.25082
226	Yo-Yo System	1	43032	0	0	28.00242
227	PCM CTL	2	174456	0	0	6.90719
228	Stacc/Stint	6	300288	0	0	4.01281
229	RF Switch	6	300288	0	0	4.01281
230	Amplifiers	2	92280	0	0	13.05808
305	Thermal Control Electronics	1	24096	0	0	50.0083
400	ACS	1	43032	0	0	28.00242

Component ID	Component Name	Sample Size	Total Hours	Total Failures	Point Estimate Failure Rate (Failures/10 6 Hours)	70% Chi Square Estimate (Failures/106 Hours)
401	Accelerometers	11	429120	0	0	2.80807
402	Attitude Control Components	116	3283788	3	0.9136	1.4495
403	Control Gas Components	18	789192	0	0	1.52688
404	Control Switching Components	21	881928	2	2.2678	4.09897
405	Earth Sensor Components	35	1323096	4	3.0232	4.4592
406	Gyro Assembly Units	23	425131	1	2.3522	5.7394
407	Gyros	33	1754952	14	7.9774	0
408	Horizon Sensors	8	395832	1	2.5263	6.1642
410	Magnetometers	32	1299000	1	0.7698	1.8784
411	Momentum/Reaction Wheel Comps.	50	1913808	3	1.5676	2.4872
412	Nutation Dampers	21	1187520	0	0	1.01472
414	Star Trackers	9	237288	2	8.4286	15.23465
415	Sun Sensors	85	2817804	2	0.7098	1.28291
416	Torquing Coils	37	1720968	0	0	0.70019
417	Valves	5	135144	0	0	8.91642
418	RIU	10	450432	0	0	2.67521
419	Gyro Power Supply	3	29832	1	33.5211	81.7914
501	Amplifiers, Power	1	20832	0	0	57.8437
502	Amplifiers, Other	8	433680	0	0	2.77855
503	Array Drive Electronics	36	1135512	7	6.1646	8.1021

Component ID	Component Name	Sample Size	Total Hours	Total Failures	Point Estimate Failure Rate (Failures/106 Hours)	70% Chi Square Estimate (Failures/106 Hours)
504	Batt Charge/Discharge Control	101	3682392	3	0.8147	1.2926
505	Battery Packs	70	2418552	3	0.8147	1.2926
506	DC/AC Converters	18	490176	0	0	2.4583
507	dc/dc converters	16	497424	0	0	2.42248
509	Motors, Electrical	5	211056	0	0	5.70939
510	Power Distribution Units	25	1182936	1	0.8454	2.0627
511	Regulators, Voltage	50	1065048	2	1.8778	3.39421
512	Solar Arrays	107	3263760	4	1.2256	1.8077
513	Undervoltage Detection/Cont Circuit	62	1534176	0	0	0.78544
514	Intercom Harness	3	127536	0	0	9.44831
601	Apogee Boost/Kick Motors	29	1165152	0	0	1.0342
602	Propulsion Module	10	393403	0	0	3.06302
603	Regulators, Pressure	22	713880	0	0	1.68796
604	Tanks	38	921352	0	0	1.30786
605	Thrusters, Cold Gas	89	979147	1	1.0213	2.492
606	Thrusters, Hydrazine	61	1075920	1	0.9294	2.2678
607	Valves	105	2419729	1	0.4133	1.0084
700	Instruments	47	1664688	2	1.2014	2.17158
900	Other	12	664632	0	0	1.81303
902	Detectors	1	56952	0	0	21.15817

Appendix 3. ERBS Anomalies

SOAR_NO	SPACECRAFT	SUBSYSTEM	COMPONENT	ANOM_DESCR	CAUSE_ANOM	CORRECT_ACTN
A00896	ERBS	Power	Solar Arrays	-Y solar array failed to deploy for 29 minutes before ERBS was released from STS. Normal array deployment is about 15 seconds.	Unknown.	None.
A00897	ERBS	ACS	Sun Sensors	Incorrect alpha angles from sun sensor # 2. Eight lsb telemetry bits are inverted. The ninth bit is incorrect.	The spacecraft sun sensor # 2 was wired incorrectly, i.e., harness from sun sensor # 2 to the electronics box was miswired (two wires reversed).	Flight Dynamics (Code 581) changed their ground calibrations to fully correct for this error in the spacecraft.
A00898	ERBS	TLM&DH	Telemetry Encoders	Approximately 6 TLM parameters from the ERBE instrument read incorrectly in the high (12.8 kb) data rate. Occurs in both digital TLM units.	Charge retention by and stray capacitance between switching circuits in the digital TLM units.	None required or possible.
A00899	ERBS	TLM&DH	Transponders	Telemetry dropouts sometimes when real-time commands were being uplinked from GSTDN stations.	This is a design characteristic of the transponder. This is not a failure. This was an unexpected and undocumented feature of the NASA STDN transponder.	When initiating commands make sure range tones are present if the ranging channel is enabled.
A00900	ERBS	TC&C	Command Decoders	Bit changes in block (delta-time) section of both command memories. The	Problem in circuit design. Block and normal memories are both enabled	ERBS is attempting to reduce the number of memory transfers by

SOAR_NO	SPACECRAFT	SUBSYSTEM	COMPONENT	ANOM_DESCR	CAUSE_ANOM	CORRECT_ACTN
				number of hits to date are 142 memory locations and bit changes over 14 days. Most bit changes have been from 1 to 0.	for a short period of time when transferring to and from block memory. This results in bit flips.	shortening delta times of CMDS in block memory. Also reducing number of 1's in time field of each command.
A00901	ERBS	Instruments	Motors, Electrical	Intermittent sticking (20-30 seconds each) of ERBE-Scanner instrument over a 6-hour period, and a slight increase in temperature. No sticking observed since.	Suspected increase in friction in scan bearing.	None.
A00903	ERBS	Instruments	Memory	Two SAGE sunrise events lost because of incorrect azimuth updates in the sunrise register. Analysis showed noise on azimuth angle readout and high motor currents.	Unknown and no recurrence.	More tests before launch.
A00904	ERBS	Instruments	Earth Sensor	Approximately 1% noise readout of elevation angle of ERBE-Scanner Instrument from 12/21/84 to 01/11/85 and 01/21/85 and continued. The slice-2 and 3 temps increased approximately 5 degrees C at the same time.	Mechanical friction in scanner bearings caused scan motor to stay in hi-power mode. Intermittent sticking occurred a few months later: see SOAR 00901.	None. Se anomaly report NASA TM 87636 (Langley) for recommendations for future spacecraft.
A00906	ERBS	ACS	Gyros	Intermittent noise from Y-GYRO rate. Occasionally spikes above yellow operating limit of 0.02 degrees/second, about 2X normal. Red operating limit	Mechanical wear on bearings. Same thing occurred in X-GYRO of IRU-1 18 months ago. (SOAR A00913). The X-GYRO failed about 6	GYRO used only in YAW turn and attitude determination by the experimenters. Contingency procedures are being developed.

SOAR_NO	SPACECRAFT	SUBSYSTEM	COMPONENT	ANOM_DESCR	CAUSE_ANOM	CORRECT_ACTN
				is 0.05 degrees/second. The frequency of noise has been slowly increasing for last 90 days.	months later, on 08/18/86.	
A00910	ERBS	ACS	IRU-1 (Gyro)	GYRO-Y in IRU-1 stopped after 1140 days of operation. Other gyros in IRU-1 and -2 have failed previously. X and Z gyros in IRU-2 remain operational. Gyros only used in spacecraft YAW turns every 4-5 weeks, and for defining attitude determination for the principal investigator.	Wearout of mechanical bearings in gyros. Signal noise increased significantly 6-12 months before failure. Expected lifetime was 2 years.	Switched to IRU-2. Developed contingency procedures for YAW turn. No significant impact of attitude determination for principal investigators.
A01048	ERBS	ACS	GYRO	Intermittent noise from X-GYRO rate occasionally spike above yellow operating limit of 0.01 degree/sec, i.e., about 2x normal. Red operating limit is 0.05 degree/sec. Frequency has been slowly increasing over past 90 days, while magnitude remains constant.	Normal wear. GYRO failed, i.e., stopped spinning, 6 months later on 08/18/86. Approximately 10% of these gyros expected to fail after 700 days (time ERBS has been in orbit).	None. Switched to backup GYRO on 08/20/86.
A01102	ERBS	ACS	IRU-2 (Gyro)	X-GYRO temp suddenly increased from 42 degrees C to 52 degrees C, and the YAW angle error signal suddenly increased 3 degrees. The X-GYRO noise had increased in the	GYRO bearing wearout. Three other gyros have failed in a similar way over the last five years.	None. The design lifetime for these gyros was 2 years. GYRO signal disconnected from spacecraft Attitude Determination System.

SOAR_NO	SPACECRAFT	SUBSYSTEM	COMPONENT	ANOM_DESCR	CAUSE_ANOM	CORRECT_ACTN
				last year. YRO output signals has been confirmed to be invalid.		
A01106	ERBS	Instruments	ERBE Scanner	This instrument abruptly stopped scanning.	This is not a mechanical failure since the instrument also does not respond to any of its mode commands. Identical instruments are aboard NOAA-9 and -10. ERBS and NOAA failures are thought to be caused by a failure in the ROM addressing circuitry.	Numerous commands have been transmitted and analysis performed without success. Apparently no corrective action is possible.
A01108	ERBS	ACS	Sun Sensors	Intermittent dropout of least significant bit of gray code in the coarse sun sensor reticle signal, increasing with time and more bits beginning to drop out.	Radiation damage of photo diodes in the 2 degree reticle. Other degradation possible.	None.
A01109	ERBS	ACS	IRU-2 (Gyro)	Gyro-Y in IRU-2 stopped after 692 days of operation. Gyro-X in IRU-1 previously failed on 081786 after 682 days of operation. The spacecraft attitude is unaffected. Gyros only used in spacecraft YAW turns once a month, and for attitude determination by instrumenter.	Wearout of mechanical bearings in gyro. Signal noise increased significantly 3-4 months prior to failure.	Switched to IRU-1, which has no X-gyro, for safest spacecraft YAW turn contingency. Spacecraft YAW turn performed on 072888.
A01177	ERBS	ACS	Gyro	X-GYRO stopped spinning.	Normal wear.	Switched to backup

SOAR_NO	SPACECRAFT	SUBSYSTEM	COMPONENT	ANOM_DESCR	CAUSE_ANOM	CORRECT_ACTN
				Temperatures increased 10 degrees in approximately 10 minutes. Output signal suddenly flat, i.e., not following motion of the spacecraft.	Approximately 2X noise on output signals started 6 months earlier. (SOAR A01048)	IRU/GYRO on 08/20/86.
B00172	ERBS	Power	Battery	Rapid increase in battery-1 temperature, cell balance, and cell pressure at time of cell failure. Batteries showed divergence and battery-1 overcharging two weeks before failure.	Single battery charger for two batteries, which are at different rates. Batteries are 8 years old and wearout is to be expected.	Lower battery charge level. Disconnect, and then reconnect battery-1 for each orbit, and change charge method each orbit to help balance charge between the two batteries.
B00173	ERBS	ACS	IRU-2 (Gyro)	GYRO-Z of IRU-1[sic] stopped operating. Increase of temperatures and currents, and loss of signal occurred at time of failure. Gyros only used for ACS control at times of spacecraft YAW turns every 4-6 weeks.	Gyro wearout after 1710 days of operation.	Only power up backup GYRO-Z in IRU-1 a few days before spacecraft YAW turn, and then power gyro off after a YAW turn.
B00174	ERBS	Power	Battery	Rapid increase in battery-1 temperature, cell balance, and cell pressure at time of cell failure.	Unable to balance charge to two different and unequal batteries. There was a cell failure one month earlier in this battery. Wearout of battery after 8 years of operation.	Disconnect battery-1, and attempt to operate mission on single remaining battery. Vary charge level and spacecraft loads in accordance with varying sunlight conditions/durations over the orbit.
B00188	ERBS	Power	Battery	Assumed cell shortage (one cell failed). Heat rise and half battery voltage	Normal use and possible overcharge due to previous redundant battery #1	Reduce charging level from V/T 6 to V/T 3. Initially suspend SAGE II data

SOAR_NO	SPACECRAFT	SUBSYSTEM	COMPONENT	ANOM_DESCR	CAUSE_ANOM	CORRECT_ACTN
				difference show dramatic increase. Two cells failed in battery # 1 last year. Spacecraft has been operating on one battery (# 2) since.	failure.	collection. Resumed normal operations after 3 days.
B00189	ERBS	Power	Battery	Assumed cell shortage (second cell failed). Immediate one-half battery voltage difference increase--delayed temperature increase. First cell failed a month ago.	Normal use and possible overcharge due to condition of the battery.	Immediate power reduction through powering off instruments. Attempts to reduce overcharge results in V/T 1 setting. Attempted to use other damaged battery unsuccessful.
B00190	ERBS	TC&C	Command Storage Memory	Time tag on commands are affected by anomalous changes in memory chips. Testing showed anomaly continuing. CSM # 2 determined to be unreliable. (It is used only for backup help, and it managed heater commands for spacecraft.	Investigation of similar anomaly on CSM #1 revealed noise susceptibility of memory chips. It is unknown why there is an increase in number of occurrences.	Discontinue use of CSM # 2. (All heaters on the spacecraft have been turned off).

Appendix 4. ERBS Reliability Data Base

Component ID	Component	Spacecraft ID	Subsystem	NPM	NRN	Typered	Comp_DC
102	Command Decoder Unit	1	C&DH	2	1	A	1
103	Telemetry Distribution Unit	1	C&DH	2	1	A	1
103	Transformer Assy, TDU	1	C&DH	2	1	A	1
104	Command Storage Memory	1	C&DH	2	1	A	1
108	Transponder Receiver	1	C&DH	2	1	A	1
109	Limiter	1	C&DH	2	1	A	1
109	Signal Processor	1	C&DH	2	2	A	1
111	Spacecraft Clock	1	C&DH	1	1	A	1
202	ESSA Assy, Mast	1	C&DH	1	1	A	1
202	ESSA Boom Assy	1	C&DH	1	1	A	1
202	ESSA Caging Assy	1	C&DH	1	1	A	1
202	ESSA Assy	1	C&DH	1	1	A	1
202	OMNI Zenith Antenna Assy	1	C&DH	1	1	A	1
202	Transformer Assy, (T1) ESSA Contr.	1	C&DH	2	1	A	1
202	ESSA Microprocessor Controller	1	C&DH	2	1	S	1
202	OMNI Nadir Antenna Assy	1	C&DH	1	1	A	1
202	ESSA Switching Pwr. Divider	1	C&DH	1	1	A	1
202	ESSA Antenna Drivers	1	C&DH	2	1	S	1
202	ESSA Antenna NEI	1	C&DH	1	1	A	1
202	ESSA Antenna Sensor	1	C&DH	2	1	A	1
202	ESSA Steering Logic Circuits	1	C&DH	1	1	A	1
205	Digital Telemetry Unit	1	C&DH	2	1	A	1
205	TDU Inductor Assy	1	C&DH	1	1	A	1
205	TDU Electrical Assy	1	C&DH	2	1	A	1

Component ID	Component	Spacecraft ID	Subsystem	NPM	NRN	Typered	Comp_DC
206	Diplexer	1	C&DH	1	1	A	1
208	Tape Recorder Unit	1	C&DH	4	2	S	1
208	Tape Recorder Relay	1	C&DH	4	2	A	1
209	TDU Multiplexer CBA Assy	1	C&DH	1	1	A	1
209	TDU Multiplexer	1	C&DH	1	1	A	1
210	Frequency Standard	1	C&DH	2	1	S	1
210	Phase Demodulator	1	C&DH	2	1	A	1
212	RF Isolator Assy	1	C&DH	2	1	A	1
212	Reference Oscillator	1	C&DH	2	1	A	1
217	RF Power Amplifier	1	C&DH	2	1	A	1
221	Transponder Transmitter	1	C&DH	2	1	A	1
223	Transponder Board	1	C&DH	2	1	A	1
223	NASA Standard Transponder	1	C&DH	2	1	A	1
301	SAGE II Elect. Thermal Radiator	1	Thermal	1	1	A	1
301	Spacecraft Thermal Radiator	1	Thermal	1	1	A	1
303	Deck Heater	1	Thermal	6	3	A	1
303	ERBE Survival Heaters (S, NS)	1	Thermal	4	2	A	1
303	Propellant Valve Heater	1	Thermal	16	8	S	1
303	Cat. Bed Heaters (1,2,3,4,A,B,C,D)	1	Thermal	16	8	S	1
303	Prop. Mounting Plate Heater	1	Thermal	14	7	A	1
303	ERBE Pulse Load Heater (S, HS)	1	Thermal	2	1	A	1
303	SAGE II Survival Heater	1	Thermal	2	1	A	1
303	Battery Heater	1	Thermal	4	2	A	1
303	ERBE Head Radiator	1	Thermal	1	1	A	1
303	ERBE Pedestal Radiator	1	Thermal	1	1	A	1

Component ID	Component	Spacecraft ID	Subsystem	NPM	NRN	Typered	Comp_DC
304	Thermal Louver	1	Thermal	1	1	A	1
304	MLI Kapton Blanket	1	Thermal	1	1	A	1
305	Heater Status Monitor	1	Thermal	1	1	A	1
306	Thermistor	1	Thermal	6	6	A	1
306	Thermistor A-338 PWB	1	Thermal	1	1	A	1
306	Thermostat	1	Thermal	16	8	S	1
400	MCS Power Supply	1	ACS	3	3	A	1
401	Accelerometer	1	ACS	4	4	A	1
402	Gyro Rate Handling Logic Circuits	1	ACS	1	1	A	1
402	Thruster Control Logic Circuits	1	ACS	1	1	A	0.04
402	ACE Electronics Assy	1	ACS	1	1	A	1
402	Thruster Selection Logic Circuits	1	ACS	1	1	A	0.04
402	Attitude Reference Logic Circuits	1	ACS	1	1	A	1
402	PWA, (A1) Power Supply	1	ACS	1	1	A	1
402	ACE Transformer Assembly	1	ACS	1	1	A	0
402	PWB, ACE Roll-YAW	1	ACS	1	1	A	1
402	PWA, ACE Aux	1	ACS	1	1	A	1
402	PWB ACE (9)	1	ACS	1	1	A	1
402	Power-up Reset Logic Circuits	1	ACS	1	1	A	1
402	PWB, A342	1	ACS	1	1	A	1
402	PWA, A325	1	ACS	1	1	A	1
402	PWA, A338	1	ACS	1	1	A	1
402	Thruster Shutdown Circuits	1	ACS	1	1	A	0.04
402	ACE Phase Plane Controller	1	ACS	3	3	A	1
402	ACE Power Supply	1	ACS	1	1	A	1
402	ACE Monitor	1	ACS	1	1	A	1

Component ID	Component	Spacecraft ID	Subsystem	NPM	NRN	Typered	Comp_DC
402	ACE IRU Temp Monitor	1	ACS	2	2	A	1
402	ACE 16 Bit Shift Registers	1	ACS	10	8	A	1
402	ACE Data Latch Circuits	1	ACS	1	1	A	1
402	ACE DAC	1	ACS	1	1	A	1
402	ACE 8 Channel Mux	1	ACS	1	1	A	1
402	ACE Signal Return & Thermal Board	1	ACS	1	1	A	1
402	ACE Roll Attitude Detector	1	ACS	1	1	A	1
402	ACE Yaw Gyro Integrator	1	ACS	2	1	A	1
402	ACE Inverting Level Detector	1	ACS	2	1	S	1
402	ACE Configuration Decoder	1	ACS	1	1	A	1
402	IRU Electronics	1	ACS	2	1	S	1
402	MCS Pre-amps	1	ACS	2	1	A	1
402	MCE Signal Processor	1	ACS	2	1	A	1
402	MCE Attitude Computer	1	ACS	2	1	A	1
404	Thruster Valve Magnetic Relays	1	ACS	3	3	A	0.04
404	Thruster Valve Latch Relays	1	ACS	2	2	A	0.04
404	Thruster Circuit Assy	1	ACS	1	1	A	0.04
404	Thruster Select Logic	1	ACS	1	1	A	0.04
404	ACE Frequency to Voltage Converter	1	ACS	2	1	A	1
406	IRU Assy	1	ACS	2	1	S	1
407	Gyrocompass	1	ACS	1	1	A	1
408	Scan Wheel (w/ HS)	1	ACS	2	2	A	1
410	Magnetometer Probe	1	ACS	1	1	A	1
411	Momentum Control Electronics	1	ACS	1	1	A	1
411	Momentum Wheel	1	ACS	1	1	A	1

Component ID	Component	Spacecraft ID	Subsystem	NPM	NRN	Typered	Comp_DC
415	Sun Sensor	1	ACS	2	1	A	1
415	Sun Sensor Electronics	1	ACS	1	1	A	1
415	Sun Sensor Cube Assy	1	ACS	3	3	A	1
415	Sun Sensor Selector	1	ACS	1	1	A	1
415	Coarse Sun Sensor Head	1	ACS	1	1	A	1
415	Fine Sun Sensor Head	1	ACS	1	1	A	1
416	Torqrod	1	ACS	4	4	A	1
417	ACE Thruster Valve Drivers	1	ACS	8	4	S	0.04
502	Current Sensing Amp	1	Power	4	4	A	1
504	PMU Battery Charger	1	Power	1	1	A	1
504	Battery Charger Relay	1	Power	1	1	A	1
505	Battery, 50 AMP Hour	1	Power	2	1	A	1
510	Power Monitor Unit	1	Power	1	1	A	1
510	PMU Battery Monitor	1	Power	1	1	A	1
510	Spacecraft Essential Bus	1	Power	1	1	A	1
510	Spacecraft Cont. Sys. Bus	1	Power	1	1	A	1
510	Spacecraft Main Bus	1	Power	1	1	A	1
510	Trans/Pri/NEI Bus	1	Power	1	1	A	1
510	Prop/Backup/NEI Bus	1	Power	1	1	A	1
510	Solar Array Disconnect Relay	1	Power	2	2	A	1
510	NEB/Sig. Terminal Board	1	Power	2	2	A	1
510	Battery Disconnect Switch	1	Power	2	1	A	1
510	Power Module On/Off Control	1	Power	3	1	S	1
510	SAGE-II Power Relay	1	Power	2	1	A	1
510	Bias and Logic Control Circuits	1	Power	1	1	A	1
510	Prec/MB Lockout Relay	1	Power	1	1	A	1
510	Battery Shunt Relay	1	Power	1	1	A	1

Component ID	Component	Spacecraft ID	Subsystem	NPM	NRN	Typered	Comp_DC
510	Battery 1 Cell Relay	1	Power	1	1	A	1
510	Battery 2 Cell Relay	1	Power	1	1	A	1
510	Power Control Unit	1	Power	1	1	A	1
510	Control System Bus	1	Power	1	1	A	1
510	Spacecraft Non-essential Bus	1	Power	1	1	A	1
510	OIB 28 V Heater Power	1	Power	1	1	A	0.01
511	Power Relay Switch	1	Power	5	5	A	1
511	SPRU Current Sensor	1	Power	1	1	A	1
511	PMU 15V Regulator	1	Power	1	1	A	1
511	28V Instrument Regulator	1	Power	1	1	A	1
511	Current Limiter	1	Power	2	1	A	1
511	Inverter/Low Voltage Supply	1	Power	1	1	A	1
511	Amp-Hour Meter Unit	1	Power	2	1	A	1
511	Standard Power Regulating Unit	1	Power	1	1	A	1
512	Solar Array Panel Assy	1	Power	2	1	A	1
512	Solar Cell	1	Power	11232	5616	A	1
512	Solar Array PWB Assy	1	Power	1	1	A	1
512	SA Drive Assy	1	Power	1	1	A	1
512	Solar Panel Gaging Assy	1	Power	2	2	A	1
512	Solar Array Caging Shield	1	Power	1	1	A	1
512	+Y Solar Array NEI	1	Power	2	1	A	1
512	-Y Solar Array NEI	1	Power	2	1	A	1
512	Solar Array Sensor	1	Power	4	2	A	1
601	Prop. Thruster Valve Driver	1	Propulsion	8	4	S	0.04
603	Thruster System Filter (25 Micron)	1	Propulsion	1	1	A	0.04
603	OAPS Transducer	1	Propulsion	2	1	S	0.04

Component ID	Component	Spacecraft ID	Subsystem	NPM	NRN	Typered	Comp_DC
604	APU Propellant Tanks	1	Propulsion	2	2	A	0.04
604	Thruster	1	Propulsion	8	6	S	0.04
607	Fill and Drain Valve	1	Propulsion	1	1	A	0.04
608	Thruster Latch Valve	1	Propulsion	4	4	A	0.04
608	Thruster Valve (Dual Series)	1	Propulsion	16	8	S	0.04
608	Fill and Vent Valve	1	Propulsion	2	2	A	0.04
608	Latch Valve and Filter Assy	1	Propulsion	4	4	A	0.04
700	ERBE Non-Scanner	1	Instruments	1	1	A	1
700	SAGE-II	1	Instruments	1	1	A	1
700	ERBE-Scanner	1	Instruments	1	1	A	1
700	ERBE-S Bi-level TLM Driver	1	Instruments	1	1	A	1
700	ERBE-S Pulse Cmd Receiver	1	Instruments	1	1	A	1
700	ERBE-S High Speed Clock Receiver	1	Instruments	1	1	A	1
700	ERBE-S Analog TLM Driver	1	Instruments	1	1	A	1
700	ERBE-S Discrete CMD Receiver	1	Instruments	1	1	A	1
801	Power Supply Frame Assembly	1	Structure	1	1	A	1
802	Sup. Instl. Fine Sun Sensor	1	Structure	1	1	A	1
802	Keel Assy	1	Structure	1	1	A	1
901	Bolometers (Horizon Scanners)	1	Other	2	1	S	1
904	Magnetic Control Electronics	1	Other	1	1	A	1
904	Magnetic Current Sensor	1	Other	1	1	A	1
904	Control Magnets	1	Other	4	4	A	1

Appendix 5. Earth Radiation Budget Satellite Description

Attitude Control and Determination Subsystem (ACDS)

The control system for ERBS is called the ACDS, and can be functionally divided into the attitude control subsystem and the attitude determination subsystem. Each subsystem requires periodic performance checks and ground supplied biases. In addition, infrared (IR) horizon scanner sun interference prediction and Omni Antenna contact prediction support is required. The ACDS also consists of the following components:

- Inertial Reference Units
- Gyros
- Sun Sensors
- Attitude Control Electronics (ACE)
- Magnetometers
- Momentum Wheels
- Magnetic Control Electronics (MCE)
- Scan Wheels
- Torqrods
- Thruster Valve Drivers
- Gyrocompass

ERBS attitude control requirements consist primarily of fine-tuning the ERBS onboard attitude control system by uplinking various control system biases. There are no requirements to uplink magnetic coil commands or thruster commands for attitude trimming during routine operations. Normally, support of any attitude maneuvers consists solely of a monitoring function. However, the ACDS has the backup capability to support the 180-degree yaw maneuvers by providing ground-determined thruster commands (thruster ON/OFF times) should an in-flight anomaly render the onboard Reaction Control System (RCS) ineffective.

Normal operations consist of bias determination activities and a daily check of the control system pointing performance. Attitude telemetry data to support these operations is available in the form of full orbit playbacks and real-time data transmissions.

Four types of biases are required by the onboard attitude determination and control system which must be determined and checked by the ACDS. These biases are the residual dipole bias, scanner bias, wheel speed differential and gyro bias.

Residual Dipole Bias. The residual magnetic dipole moment is less than 2600 pole centimeters along any of the three axes. Compensation for the dipole moment is limited, due to the coarse resolution of residual dipole bias commands. However, the ACDS analyzes flight data and provides recommendations, and subsequent evaluations of the use of this bias. Compensation for the residual magnetic dipole will likely consist of only a 1-3 count electromagnetic bias command about any of the three spacecraft axes.

Wheel Speed Differential. The wheel speed differential will give the spacecraft a component of momentum along the yaw (nadir-pointing) axis. This bias reduces the effect of gravity gradient torques on spacecraft pointing control. The boresights of the dual horizon scanners, which provide 30 percent of the spacecraft's total momentum, are tilted out of the roll/pitch plane. Thus, by varying the difference in scanwheel speeds, the yaw component of momentum can be affected.

Horizon Scanner Bias. Roll and pitch biases are required to correct the IR horizon scanner data. Although scanner mode and pitch dependent, ground determination of scanner biases is nominally for the dual scanner, 0 degree pitch orientation (both 0 and 180 degree yaw). These biases are updated throughout the mission. Biases for single scanner, pitch 0 degrees are determined only if sufficient flight data in that mode can be collected.

Figures 2 and 3 below provide functional and reliability diagrams for the ERBS ACDS system. Figure 4 depicts the ACE Thruster Valve Drivers reliability relationships.

The Attitude Control and Determination Subsystem provides the following for the ERBS mission:

- Pointing control of the ERBS within +/- 1 degree in the X and Y axes, and within +/- 2 degrees in the Z axis, using either the Magnetic Control System (MCS) or Reaction Control System (RCS).
- Orbit adjust thruster control (raises orbit and controls ERBS to within +/- 3 degrees).
- Attitude maneuver capabilities to any one of eight orientations (four pitch positions in two yaw references).
- Attitude determination (on-board) angle results to within +/- 0.25 degree for X and Y, and to within +/- 0.25 degree for Z day-time results and +/- 1 degree for Z night-time results.

These activities use a combination of the MCS and RCS for normal operations with backup modes available. Table 3 provides a detailed listing of RCS and MCS utilization.

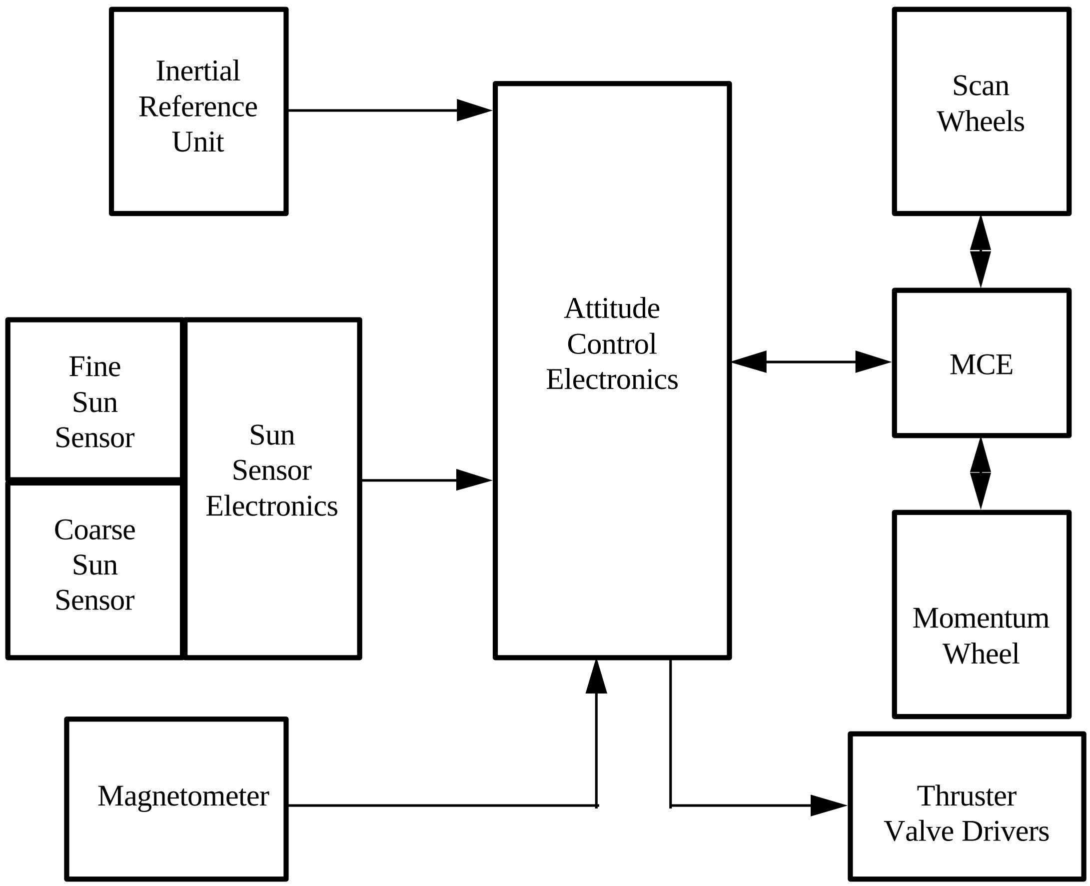

Figure 2. ACDS Subsystem Functional Diagram (Simplified)

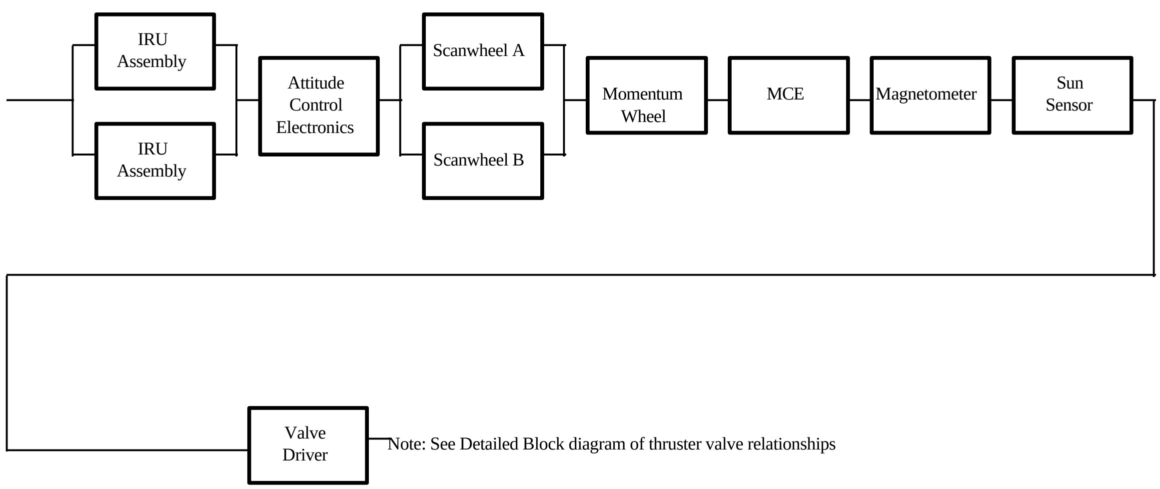

Figure 3. ACDS Subsystem Reliability Block Diagram

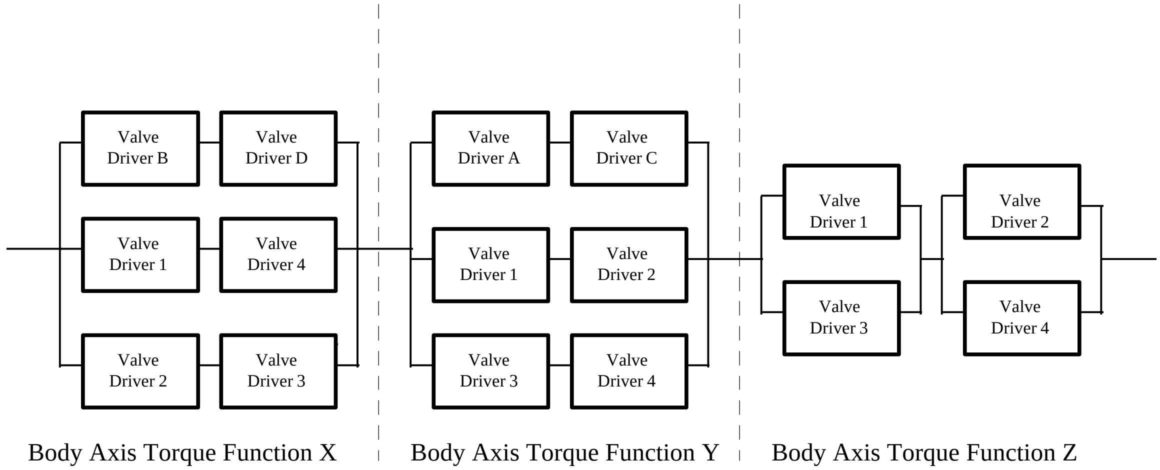

Figure 4. Thruster Valve Drivers Reliability Block Diagram

Magnetic Control System. The Magnetic Control System (MCS) is comprised of the following components:

- Magnetic Control Electronics (MCE)
- Scanwheels (SW) (2, with horizon scanners)
- Momentum Wheel (MW)
- Torqrods (4 magnetic dipole control on 3 axes)
- Magnetometer

These components act as an integrated system, using sensor data to determine spacecraft position and maintain control on all three axes in the reference position. The horizon scanners and the magnetometer provide sensor input to the MCE which computes attitude errors. The MCE controls the scanwheels, the momentum wheel, and the torqrods in order to respond to errors, and correct the spacecraft position.

For the ERBS mission, the MCS is used as follows:

- Normal pointing control at all times when above 400 Km;
- Pitch 90/270 maneuvers for orbit adjust operations; and,
- Pitch 180 maneuvers for instrument calibrations.

In addition to normal pointing control and maneuver activities, the MCS has an attitude safe-hold mode termed the reacquisition mode. This mode is a magnetic alignment of the ERBS with the earth's magnetic field, and results in a slow, controlled tumble (2X pitch tumble per orbit).

Table 3. ERBS ACDS Utilization Summary

ALTITUDE	ACTIVITY	PRIME METHOD	BACKUP METHOD	FREQUENCY	REMARKS
Below 400 Km	Pointing Control	RCS	MCS	Approx. first 2 days of mission	MCS unstable below 400 Km
Below 400 Km	Pitch 90/270	RCS	MCS	Nominally 1 maneuver before test/cal burn	Pitch 90 for +X forward, pitch 270 for -X forward
Below 400 Km	Delta V	RCS	Manual Thrusters	Test burn, calibration burn	Orbit transfer
Above 400 Km	Pointing Control	MCS	RCS	Continuous	
Above 400 Km	Pitch 90/270	MCS	RCS	Once every 3 to 6 months	Pitch 90 for +X, pitch 270 for -X for delta V burns
Above 400 Km	Pitch 180	MCS	RCS	Approx. 6 times in 2 years	For instrument calibration
Above 400 Km	+X/-X & -X/+X Yaw	RCS	Hybrid	Every 30 days @ Beta=90 degree	Hybrid in MCS and manual thruster firings
Above 400 Km	Delta V	RCS	Manual Thrusters	Every 3 to 6 months	Orbit transfer and adjustments

<u>Magnetic Control Electronics (MCE)</u> The Magnetic Control Electronics (MCE) is the primary component of the MCS and serves as the controller of all other MCS components.

MCS Control Modes. The MCE serves as the MCS controller and has associated with it three control modes. They are:

- Normal Mode

- Pitch-Only Mode
- Reacquisition Mode

Normal Mode. The Normal Mode, as its name implies, is the primary control mode of the MCS, and is used almost continuously throughout the mission to control ERBS pointing. The Normal Mode can stabilize the ERBS in any one of eight positions (pitch 0, 90, 270, and 180, in both a +X forward and -X forward flight direction) to within +/- 1 degree in roll and pitch, and +/- 2 degrees in yaw. The Normal Mode is active in the MCE when the control loops are closed, all three power supplies are on, any two of the three wheels are on, and the torqrods are enabled. Commanding the MCS to the normal control mode is accomplished as follows:

- power supplies A, B, and C on,
- scanwheels A and B on,
- horizon scanners A and B on,
- momentum wheel on,
- pitch reference set to 0 degree pitch (nadir earth pointing),
- auto-reacquisition mode disabled,
- attitude computer B selected, and
- control loops closed (enables control of all three ERBS axes).

Commands also establish the scanwheel, momentum wheel, and MCE polarities for either a +X forward or -X forward flight direction.

Pitch-Only Mode. The pitch-only mode of the MCE involves active MCS control of the spacecraft pitch loop only, leaving roll and yaw control of the spacecraft unattended. When in the pitch-only mode, the momentum wheel and X/Z torqrods are used to control the pitch axis to within +/- 1 degree, using horizon sensor data as the primary sensor input for position and error determination.

For the pitch-only mode to operate properly, the momentum wheel must be on, the bolometers (horizon scanners) must be on, and the torqrods must be enabled. When in this mode, no roll or yaw control of the spacecraft is provided. The scanwheel speed will drop to approximately 1250 rpm each, the Y-torqrod dipole will be near 0, and the control loop status will indicate open. Use of the MCE pitch-only mode is not planned for the ERBS mission, unless an anomaly occurs which predicates the need for it.

Reacquisition Mode. The third MCE mode is the Reacquisition Mode. This mode uses the magnetometer input for sensing the earth’s magnetic dipole and locks the ERBS to the field, using a strong set of torqrod control laws. It does not use the wheels, the horizon sensors, the signal processor, or the attitude computer in its operation. It is an emergency attitude control mode only. The reacquisition mode can be initiated by command or automatically by the MCE, when the roll error meets or exceeds 10 degrees (when the auto-reacquisition mode is enabled). When in this mode, the torqrod bi-level will indicate a disabled state, even though, in reality they will not be disabled. The momentum wheel speed drops to approximately 1250 rpms, and the scanwheel speeds drops to approximately 2000 rpms (with +/- 1/2 speed bias). The spacecraft will begin a magnetic capture operation which should lock the roll, pitch, and yaw axes into a position following the magnetic field direction, producing a slow, controlled tumble.

MCE Signal Processors. The MCE contains two signal processors (A and B) which receive the horizon scanner signals. It conditions these signals for use by the MCE attitude computers. The signal processors are redundant, with both processors sending conditioned signals to attitude computers A and B.

MCE Attitude Computers. The MCE contains two attitude computers (A and B) which receive the conditioned horizon scanner signals from the signal processors and compute roll and pitch errors. In essence, attitude computers A and B are redundant units (there is one exception, the ACE interface).The use of the attitude computers is as follows:

- Attitude computer B will be used by the MCE at all times.
- Attitude computer A will be a backup to attitude computer B (in the event of a failure) for the MCE.

A command can select attitude computer B for use by the MCE. A switch to attitude computer A can be performed by executing the command, and can also occur automatically within the MCE, if power to attitude computer B is lost (power supply B is turned off). When an attitude computer is selected, its roll and pitch error signals are used by the MCE in controlling the ERBS (when in a normal or pitch-only control mode). The attitude computer that is not selected will continue to operate, and will generate independent roll and pitch errors. However, these signals will not be used in controlling the ERBS.

Telemetry provided by each attitude computer indicates the roll and pitch errors of the ERBS (with respect to the pitch reference) as determined by the MCE. This telemetry is perhaps the most important attitude control telemetry, and is used often to verify nominal MCS control.

The attitude computers have a commandable relay which enables and disables use of the horizon scanner A and B signals from the signal processor. These relays will only be used in the event of:

- a component failure (power supply, signal processor, or horizon scanner), and
- sun or moon interference of the horizon scanner.

The MCE attitude computers can not only be commanded to disable use of the horizon scanner signals, but can automatically disable use of the scanner signals. If the horizon scanner signals become erroneous as determined by the attitude computer, the signals will not be used and the telemetry parameter will indicate the disabled state. This automatic decision protects the system from scanner failures.

Another function of the attitude computers is to receive uplinked roll and pitch angle bias commands. The attitude computers then condition the roll and pitch attitude error results, accordingly.

Nutation Damping Control. One function of the MCE is to provide nutation damping control. This is performed by circuitry which interprets the roll errors, and uses the scanwheels and the Y-torqrods to damp the nutation.

Precession Control. The MCE also contains circuitry which corrects for precession errors. Roll errors are evaluated by the circuitry and Y-torqrod action is initiated in response.

Momentum Wheel/Scanwheel Interface and Control. The MCE is responsible for:

- Momentum Wheel Speed Control
- Momentum Wheel Use
- Scanwheel Speed Control
- Scanwheel Use
- Momentum Wheel/Scanwheel Speed Telemetry
- Scanwheel/Momentum Wheel Speed (Momentum) Dumping

Normal MCE operations will entail the use of the Momentum Wheel (MW) and both Scanwheels (SW) for attitude control. If an anomaly arises which dictates disabling of either of the three wheels, commands are available to disable each wheel individually. Commanding MCE use of the MW or the SW's is not desired, and not planned, barring any anomalies. Therefore the MCE maintains use of the wheels for roll and pitch control of the ERBS (when in a normal or pitch-only control mode).

The MCE, in controlling and using the MW and SW's, will maintain various wheel speeds, as needed to respond to MCE control modes, roll and pitch errors, and scanwheel speed biases. If the wheels are enabled for use by the MCE, speed control occurs. In controlling the momentum wheel and scanwheel speeds, the MCE contains momentum dumping logic, which uses the X and Z torqrods to apply torques to allow wheel spin up and down.

Torqrod Control/Magnetometer Interface. The MCE drives the torqrods in response to nutation and precession errors, momentum unloading needs, and the reacquisition mode. In driving the torqrods, magnetometer data is evaluated (X, Y, and Z axes magnetic field) to determine the torqrod to be excited, as well as the dipole of the torqrod. In addition, the torqrods are used to correct for spacecraft residual magnetic dipole, through the use of torqrod biases.

MCE Biases. The MCE uses six biases for pitch, scanwheel, roll and torqrod control. They are:

- Pitch angle bias: primarily used to correct for alignment errors in the momentum wheel.
- Roll bias: primarily used to correct for alignment errors in the scanwheels.
- Scanwheel speed bias: used to correct for X-axis gravity torques by setting in a different speed for Scanwheels A and B.
- Torqrod X, Y, and Z biases which are used to correct for spacecraft residual magnetic dipole.

All of these biases are uplinked to the ERBS. The pitch bias conditions the pitch signal in the attitude computers. The roll bias conditions the roll signal in the attitude computers. The scanwheel speed bias sets the nominal speed difference between A and B. The torqrod biases condition torqrod dipole at the torqrod driver in the MCE.

MCE Interface with the ACE. The interface from the MCE to the ACE entails two voltage signals: output from the MCE attitude computer A only, and input to the ACE gyrocompass. One voltage signal represents the roll error, as computed by attitude computer A. The other represents the pitch error, also a result of attitude computer A. These signals are used by the ACE gyrocompass to refine the gyrocompass angle and rate results.

MCE-to-ACE Interface Function. Briefly, the function of this interface is to provide accurate roll and pitch error signals (derived from horizon scanner data) to the ACE gyrocompass to improve the overall accuracy of the gyrocompass. Without it, the gyrocompass can only integrate rates using gyro data, which is characteristically noisy and causes the ACE to drift over long periods of time. The advantages of having and operating this interface are:

- improved attitude determination results; and,
- improved RCS control, when active.

ACE-to-MCE Interface. The ACE supplies two analog voltages to the MCE, one for pitch bias, and one for scanwheel speed bias. These are carried by separate lines, and the pitch bias splits and goes to both the A and B attitude computers.

ACE-to-MCE Interface Function. The ACE converts serial digital commands into analog voltage commands, which is the form of command required by the MCE. This is done in the ACE because the C&DH Command Decoder Unit (CDU) is not capable of providing analog voltage commands.

Momentum Wheel Operations. The momentum wheel is controlled by the MCE. It provides MCS pitch control of the ERBS, and induces a gyroscopic stiffness into the ERBS body to aid in maintaining pointing control. The gyroscopic stiffness effect is a function of the wheel speed, while pitch control is provided by the spin direction and acceleration/deceleration torquing effects. The MW speed is controlled by the MCE application of a motor drive voltage, which varies widely due to spinup, spindown, and speed maintenance requirements in the control loops. This drive voltage is downlinked in the telemetry data. Normal operating conditions result in an application of 8 to 15 volts to maintain speed, and 15 to 30 volts to drive the wheel speed up or down. Rarely is the drive voltage dropped to 0 volts, and certainly not for any significant period of time. In the event of a drive voltage failure, the momentum wheel speed will drop to low values within minutes, and the MW speed limit will be reached.

Another major activity concerning momentum wheel operations deals with wheel reversals. The momentum wheel spin direction is dependent upon the flight direction. When the spacecraft is yaw’ed around 180 degrees every 30 days (for solar array illumination), the momentum wheel must be reversed and spun-up in the yaw maneuver command modules. This condition occurs as an integral part of the maneuver. The wheel reversal process is a function of the NEB bus voltage. But variations are slight with respect to reversal time.

Another aspect of momentum wheel operations deals with the friction bias. The bias is built into the system and cannot be changed. It is established for friction characteristics inherent in the MW. However, as the MW bearings wear, the friction will change, and the built-in bias may be incorrect.

Scanwheel/Horizon Scanner Operations . Provided on the ERBS, are two scanwheel units with each scanwheel having an infrared earth horizon scanner. Scanwheel B is located on the +Y (sun) side of the spacecraft, mounted on the instrument module. Scanwheel A is mounted opposite and slightly above scanwheel B on the instrument module -Y (anti-sun) side. Both scanwheels are aligned with the Y axis, with each wheel canted down from the X-Y plane by 10 degrees to provide a yaw component of the torque created by the wheel accelerations and decelerations.

Scanwheel operations can be divided into two functional groups:

- Scanwheel Rotation Operations
- Horizon Scanner Operations

Scanwheel Rotation Operations. One of the two functions of the scanwheels is to provide momentum and torques to the ERBS body to maintain yaw stability and roll

control. Pitch control can also be provided by the scanwheels, should the momentum wheel fail (automatically occurs in the MCE when the MW is turned off). The scanwheel provides these control functions via wheel rotations and associated accelerations and decelerations.
Under normal conditions, the scanwheel spin function will always be on. However, they can be separately commanded off. The SW speed will vary slightly, around a 2000 rpms nominal speed under normal control conditions.

The scanwheels can be commanded to offset the speeds between the two wheels by a commandable range of +/- 850 rpms. This speed offset is referred to as the scanwheel bias or scanwheel speed bias. It is uplinked to the ERBS. The scanwheel bias can be a negative or positive value with the polarity of the value dictating which of the two wheels will spinup and which one will slow down to arrive at the commanded speed delta.

Drive voltage is the signal provided by the MCE to the wheel motor drive to maintain, increase, and decrease the speed. Normal speed maintenance by the MCE entails 7 to 15 volts. When the MCE is required to drive the wheel speed up or down, 15 to 30 volts will be seen (just after a new speed bias is uplinked, when roll rates are large, and when the wheel is commanded to reverse).

Horizon Scanner Operations. The horizon scanners will normally remain on. Each horizon scanner uses the pitch reference to measure the field of vision arc distance from the pitch reference to the trailing earth's horizon. The horizon scanner will be affected by sun and full moon interference when these bodies are at or near the horizon (20 degree). This interference results in a stretched earth pulse which equates to roll and pitch errors. To avoid corruption of the MCE control and the ACE attitude determination that results from this interference, the MCS is switched to a single scanner mode.

<u>Torqrod/Magnetometer Operations.</u> The MCS uses the torqrods (1 X-axis, 1 Z-axis, and 2 Y-axes) for spacecraft control and wheel momentum unloading. Torqrod activities are directly related to the magnetometer sensing of the earth's magnetic field. Magnetometer power is automatically applied when the MCE is powered up, and can be turned off by turning off power supply C. The Y torqrods (2) are used by the MCE nutation damping and precession control loops.

The X and Y torqrods (1 per axis) are used by the MCE exclusively for momentum unloading. The combination of the scanwheels (2) and the momentum wheel spin action results in momentum along the Z-axis. However, because the momentum wheel and scanwheels are dynamically used by the MCE for roll and pitch control, their speeds will vary. To vary the speeds and still maintain the Z-axis momentum and pointing, the torqrods are used to increase or decrease momentum as needed, to allow wheel speed accelerations and decelerations.

Although they can be disabled torqrods will only be disabled in the event of a failure or emergency switch to the RCS, but are not disabled for any normal operations.

The torqrods are also used to correct for residual spacecraft magnetic dipole via the uplink of torqrod biases (one bias for each axis). The bias values are uplinked as needed. To stop all drives to the torqrods, the torqrod drivers can be turned off, or the biases can be zero'ed.

<u>Reaction Control System.</u> The RCS will be used for:

- Yaw Maneuvers (every 30 days),
- Pitch Maneuvers (when below 400 Km),
- Delta V Orbit Adjust Thruster Firings, and
- Backup to the MCS for Pitch Maneuver and Pointing Control, if the MCS fails.

Components of the RCS include:

- Attitude Control Electronics (ACE);
- Inertial Reference Units (IRUs); and,
- Propulsion Interface.

Attitude Control Electronics (ACE) Operations. The Attitude Control Electronics is the central component of the RCS, and also performs the task of on-board attitude determination. It contains many subcomponents which have unique functions and operating procedures. The subcomponents are:

- Gyrocompass
- Gyro Rate Handling
- Thruster Control Logic
- Thruster Shutdown Logic
- Thruster Drivers
- Thruster Selection Logic
- Attitude Reference Logic
- Power Up Reset States

Gyrocompass Operations. The gyrocompass in the ACE is an integrator which receives spacecraft rate and roll/pitch angle error signals and produces attitude results. It is important at all times because it provides attitude determination results for science data processing, and is used by the RCS in maneuvering and controlling the ERBS. The gyrocompass operates in one of two modes:

- scanner/rate gyro mode (GCOMP), or

- rate gyro only mode (GRATE).

Scanner/Rate Gyro Mode. The scanner/rate gyro mode combines roll and pitch signals from the MCE (horizon scanner), with the rate gyro data from the IRUs, to provide a more accurate rate and angle error result. This scanner/rate gyro mode is used at all times, except for:

- Attitude maneuvers,
- Switching to or from a single scanner mode, and
- MCE failure, which would render the roll and pitch signals useless (scanner failure, attitude computer A failure, etc.).

Rate Gyro Mode. The rate gyro mode is used at times when the MCE roll and pitch signals are known to be corrupted, invalid, or saturated. Occurrences such as these arise during attitude maneuvers, scanner mode switches, and certain MCE failures. When the rate gyro only mode is required, it is commanded automatically by Command Storage Memory (CSM) commands, usually as an integrated step in a maneuver or scanner interference command module. When in the rate gyro only mode, the selected IRU must be powered on and operating nominally.

Gyro Rate Handling. The ACE has relays and circuitry which selects and converts IRU gyro rate signals to voltage levels for use by the gyrocompass. The ERBS design contains two redundant IRUs. Only one is used at a time, with IRU-1 selected as the prime, and IRU-2 selected as the backup. IRU power is provided by the non-essential bus, and is turned on and off via commands. IRU selection is via a relay in the ACE, which responds to the commands. Once selected, the gyro signals are passed to the frequency-to-voltage converter for signal conditioning. The conditioned signals are then provided in the telemetry, and to the gyrocompass.

Thruster Control Logic. The ACE contains circuitry which controls thruster firings for RCS control and delta V operations, when enabled. A single relay in the ACE enables and disables use of the thrusters by the RCS. On-orbit operations occur primarily via CSM commands and transfer control to, and from, the MCS for yaw and delta V operations. The thrusters will remain disabled for a majority of the mission while the MCS controls the ERBS. However, they will be enabled once every 30 days to perform the yaw maneuver, and as needed to perform orbit adjust delta V operations.

The RCS thruster control logic can operate in one of two modes:

- RCS Control Mode, or
- Delta V Control Mode.

RCS Control Mode. The RCS control mode can be described as an "off, pulsed on" thruster-use mode. The RCS will maintain the spacecraft attitude to within +/- 0.85

degree of the nominal 0 degree reference (as determined by the attitude reference bit configuration).

Delta V Control Mode. The delta V control mode can be described as follows:

- The delta V thrusters (A, B, C, and D) are used in an "on, pulsed off" mode.
- The yaw control thrusters (1,2,3, and 4) are used in an "off, pulsed on" mode.

The delta V mode enables orbit-raising by accelerating the ERBS in the direction of the velocity vector, yet maintains spacecraft control to within +/- 3 degrees. The delta V mode requires that the thrusters be enabled, the latch valves be open, and the catalyst bed heaters be on. The delta V mode is only used during special operations when the ERBS orbit needs adjusting. The commands to activate the delta V mode will be executed by CSM at burn start time.

The delta V mode is terminated via command, or automatically by the thruster shutdown logic. The command is executed out of burn stop time. When the command is sent, it reverts the RCS back to the normal RCS control mode.

One final item involving the operations of the thruster control logic deals with a thruster rate gain selection. The thruster control logic can amplify the error signal used by the thruster control logic to result in a more efficient control of the ERBS, and to reduce fuel consumption. This high gain setting is the predominant setting for most of the RCS operations. The low gain setting is only used for:

- low attitude RCS control (below 400 Km);
- yaw maneuvers;
- delta V operations; and,
- RCS pitch maneuvers.

When the low gain setting is required, the command is executed by the CSM as part of the integrated command module, and will be reset to high gain upon termination of the activity.

Thruster Shutdown Logic. The ACE contains an automatic thruster shutdown capability to terminate thruster use if errors meet or exceed 5 degrees. The shutdown logic contains two modes of operation, and an enable/disable switch for each mode. It will remain enabled at all times except for special operations, at which time it is commanded to the disable state, as part of the integrated command module executing out of the CSM. For normal operations, the spacecraft check procedure will verify the enable option is selected.

Thruster Drivers. The ACE contains eight thruster drivers (one for each thruster) which provide the power to open the thruster valve and allow hydrazine to pass through the valve. Each driver receives 28 volts from the non-essential bus, and is activated upon execution of the thruster enable command. The drivers are controlled by the thruster control logic which determines when a thruster valve should be opened. The thruster drivers are absent of any telemetry or commands (other than the thruster enable command). These ACE thruster drivers are completely independent of the Propulsion thruster valve drivers, and the Propulsion thruster valve bus. The latch valves must be OPEN in order to receive hydrazine at the thrusters (they normally remain open). The catalyst bed heaters for the yaw and delta V thrusters must be on prior to thruster use, and at, or above, 93 degrees C.

Thruster Selection. The ACE has several options for thruster selection. These options revolve around a nominal, preferred selection, and one or more backup selections. Selections are made based on body coordinate axes, as follows:

ROLL AXIS -- Normal setting is thrusters B (positive torque) and D (negative torque). Backup setting is thrusters 1 and 4 (positive torque) and 2 and 3 (negative torque).

PITCH AXIS -- Normal setting is thrusters A (positive torque) and C (negative torque). Backup setting is thrusters 1 and 2 (positive torque) and 3 and 4 (negative torque).

YAW AXIS -- Normal setting is thrusters 1 and 3 (positive torque) and thrusters 2 and 4 (negative torque).

Only the normal settings are expected to be used. If, however, a thruster failure occurs, the backup settings may be used.

Attitude Reference Logic. The ACE is setup to support one of eight possible orientations (pitch 0, 90, 270, and 180 in a +X forward orientation, and a -X forward orientation). In order to achieve these configurations, and to compute attitude determination results correctly, the ACE attitude reference bits must be carefully managed. Three attitude reference bits are reconfigured for the different flight orientations.

Managing attitude reference bit configuration is handled by the CSM command modules developed for attitude maneuvers. Orientation changes are performed by the attitude maneuver command modules. In the case of the RCS yaw and pitch maneuvers, the reference bit set commands are critically important, and actually control the maneuver.

ACE Power Up. The ACE was powered up shortly after timer expiration, during the Shuttle deployment phase of the ERBS mission, and is expected to remain powered at all times.

Inertial Reference Unit. The ERBS design includes two Inertial Reference Units (IRU's) which provide an important aspect of the ACE attitude determination and RCS control functions. Each IRU contains three gyros (one on each axis) which provide spacecraft rate of motion data to the ACE. The IRU's are powered by the Non-Essential Bus upon command execution. Both IRU's can be powered at one time, but only one can be used by the ACE. IRU-1 was initially selected for use by the ACE throughout the life of the mission unless it failed. [7] Therefore, it will remain powered on, while IRU-2 will remain powered off. Upon power up, the gyros inside the IRU are spun up to approximately 64,000 rpms. During the IRU spin up, the bus current will be slightly higher than normal, but will settle down within 45 seconds. If IRU-1 fails, (see footnote 1) it must be powered off, IRU-2 powered up, and IRU-2 selected in the ACE.

IRU-1 must remain powered and selected, and IRU-2 must remain off (except for periodic "on" cycles, performed once every 3 months). This is accomplished via the spacecraft check procedure, which is executed several times daily.

The IRU's are very sensitive devices that can measure very small motions. Because of their sensitive nature, and high spin rate, the gyro signals have an element of “noise” associated with them. These noise characteristics change with time and operating temperature. In order to correct for the noise characteristics, the gyro rate signals can be biased (each axis can be separately biased). This noise equates to a specification of 0.6 degrees per hour rate. Although bias commanding is intended to null out most of the noise over long periods of time, the noise still creates some drift in the gyrocompass. To correct this drift, the gyrocompass (GCOMP mode) is used. This mode uses the horizon scanner signals from the MCE to refine and validate gyro rate data. However, it is not always feasible to use the GCOMP mode.

Sun Sensors. The ERBS contains a sun sensor unit which houses two heads and an electronics assembly. The sun sensor is ***not*** used in the control loops of the ERBS, but instead enables an independent ground check of the spacecraft attitude. It is primarily used for on-board attitude determination validation checks, but does have some applications in mission operations. The sun sensor receives power from the Control System Bus (CSB) upon command execution. The sun sensor is expected to remain powered throughout the life of the mission. [8]

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[7] GYRO-Y in IRU-1 stopped after 1140 days (02/20/86) of operation. Other gyros in IRU-1 and -2 had failed previously. X and Z gyros in IRU-2 remained operational. Gyros only used in spacecraft YAW turns every 4-5 weeks, and for defining attitude determination for the principal investigator.

[8] On 10/08/84 there was an incorrect alpha angles from sun sensor # 2. The spacecraft sun sensor # 2 was wired incorrectly, i.e., harness from sun sensor # 2 to the electronics box was miswired (two wires reversed). Flight Dynamics (Code 581) changed their ground calibrations to fully correct for this error in the spacecraft. Eight 1 sb telemetry bits are inverted. The ninth bit is incorrect. On 04/22/89 there was an intermittent dropout of least significant bit of gray code in the coarse sun sensor reticle signal,

In addition to providing independent ground checks of the spacecraft attitude, the sun sensor can be used during attitude maneuvers as a coarse indication of the spacecraft attitude, and how the attitude is changing. During most attitude maneuvers the error telemetry saturates, and remains saturated for some time. The sun sensor provides a method of validating attitude and maneuver characteristics. However, it is only applicable when the ERBS is in view of the sun.

Thruster Operations. Although the thrusters are a Propulsion subsystem component, a brief discussion is presented here, since the RCS is active (thrusters are enabled), and are used to raise the ERBS orbit to achieve and maintain a 594 to 610 Km near circular orbit. The RCS controls thruster firings, based on gyrocompass angles and rates. To enable the use of the thrusters by the RCS, the following is required:

- The Propulsion latch valves must be open. This allows hydrazine to pass from the tanks to the thruster valve.
- The Propulsion mounting plate, propellant valve, and catalyst bed heaters must be on. This configuration ensures the hydrazine, the valves, and the catalyst bed in the thrusters are all at optimum temperatures. The result is more efficient thrust and minimized erosion of the catalyst bed.
- The catalyst bed temperatures must be at or above 93 degrees C.
- The ACE thruster control, thruster shutdown, and thruster selection logic must be configured for the desired operations.

In the case of the latch valves, the mounting plate, and the propellant valve heaters, the normal flight configuration calls for them to be configured for thruster use at all times. The catalyst bed heaters do not remain on at all times, but instead are cycled on and off prior to and after thruster use. The ACDS command modules, which control virtually all planned operations, contain the catalyst bed heater commands.

Propulsion Interface. The RCS uses the Propulsion subsystem thrusters in performing pointing, delta V, and maneuver operations. In addition to RCS use, the thrusters can be operated directly (by command) via the Propulsion subsystem. The interface between the Propulsion subsystem and the RCS must be called out and properly configured to allow use of the thrusters. The manual thruster commands are executed via the thruster valve bus, which has an enable/disable gate. The ACE thruster valve drivers bypass the thruster valve bus and interface directly with the valves. All that is required for RCS use of the thrusters (besides ACE internal configurations) is that the latch valves be open.

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increasing with time and more bits beginning to drop out. The cause was radiation damage of photo diodes in the 2 degree reticle.

Pointing Control Operations. The previous sections presented functional information for each component of the MCS and RCS, and indicated various uses in operations of the components and capabilities. This section attempts to integrate all characteristics of the MCS and RCS, describing the functions required of the ACDS. First, this section will present the normal flight operations of the ACDS, which entails nadir pointing control and on-board attitude determination. In particular, the following are addressed:

- ACDS normal operations configuration, and
- Control and configuration monitoring.

Normal Pointing Operations. The ACDS configuration for normal pointing control can be summarized as follows:

- MCS on, in a normal mode, with all wheels, torqrods, and sensors active.
- RCS configured for use, but thrusters are disabled.
- Attitude determination ongoing in gyrocompass, using the GCOMP gyrocompass mode.

The configuration is expected to remain stable with the exception of:

- Bias values,
- Periodic single horizon scanner modes (due to interference), and
- Monthly polarity and wheel spin direction changes (for yaw inversion to maintain sun on the arrays).

Table 4 provides an expanded description of the ACDS configuration and associated status.

<u>Attitude Maneuver Operations.</u> The ERBS normally operates in a nadir earth pointing attitude (pitch 0) for routine science data collection. However, attitude maneuvers are required periodically due to:

- Orbit adjust requirement (594 to 610 Km near circular orbit).
- Solar array illumination (maintain good incidence angle between the sun and the ERBS solar array).
- Instrument calibrations (primarily for the ERBE instruments).

Maneuver Command Modules. Two options exist. Burns can be executed with the acceleration directed along the velocity vector (to raise the orbit), or can be executed with the acceleration directed along the anti-velocity vector (to lower the orbit). Anti-velocity burns are not anticipated, but can be accommodated, if needed.

Pitch Maneuvers. The command module, which pitches the ERBS from a normal pitch 0 nadir pointing attitude to a pitch 90 (or 270) attitude, performs a variety of pre-pitch activities required to safe the instruments, configure the C&DH subsystem, configure the RF subsystem, and initiate Tape Recorder Unit (TRU) recording. The following occurs:

- SAGE instrument azimuth slew to 180 degrees (contamination purposes), then powers down.
- ERBE Scanner and Non-scanner instruments are placed in the stow position.
- The zenith omni antenna is selected, and the transponders are configured for low rate commanding and coherent enabled.

Table 4. ACDS Configuration and Status

Configuration Item	Status
MCE Power On	Remains on at all times
ACE Power On	Remains on at all times
MCS Normal Mode	3-axis MCS Control
MCS Control Loops Closed	Closed at all times
Pitch Reference in MCS set to 0 degrees	0 degrees
SW, MW, On	Always on
SW Horizon Scanners On	Will occasionally revert to a single scanner mode
Torqrods enabled	Remains enabled
MCE polarities set to flight orientation.	Varies monthly, depending on beta angles
Thrusters disabled	RCS off
IRU-1 selected	Selected
Attitude Reference bits set to 0 pitch +X and -X forward	Varies monthly, depending on beta angles
Auto-reacquisition mode enabled	At mission altitude only
Momentum wheel direction set to +X or -X	Varies monthly, based on beta angles
Power Supply C on	On at all times
Delta V logic disabled	Disabled
Delta V Burn Stop	Stop
Attitude Computer B Select	Selected
Sun Sensors on	Always on
IRU-1 Power on IRU-2 Power off	on/off
Gyrocompass mode set to GCOMP (scanner/rate gyro)	Periodically switch to GRATE mode for single scanner mode reconfiguration
Yaw bias gain set to low	Must remain low for normal operations
All three gyro signals on	Always on
Thruster rate gain set to high	Remains configured for RCS use, if needed
Thruster auto-shutdown logic enabled	Remains configured for RCS use, if needed
Thruster selections set to normal mode	Remains configured for RCS use, if needed
Biases Installed: • Roll • Pitch Angle	Bias values are defined and uplinked.

Configuration Item	Status
• Speed (SW's) • Gyros • Torqrods	

- The SAGE TRU not in use (TRU-3 or TRU-4) is configured for recording 1.6 kbps CSM/ENG data on DTU-2.
- The ERBE Non-scanner standby heaters are turned on.
- The transponder I, Q, and GSTDN-2 ports are configured.

Delta V Orbit Adjust. Periodic maintenance of the ERBS orbit will be performed through the execution of short (15 minutes or less) thruster burns termed "delta V maneuvers." CSM command modules are used to perform delta V maneuvers under memory control. The maneuver modules are structured to perform the following:

- Turns on the catalyst bed heaters one hour in advance of the burn.
- Switches attitude control from the MCS to the RCS.
- Configures the RCS for the burn (auto-shutdown thruster logic enabled, and the gyrocompass set to the GCOMP mode).
- Starts and stops the delta V burn at the specified time.
- Switches back to MCS control after the burn and turns off the catalyst bed heaters.

Pitch 180 Degree Maneuvers. The ERBS will be pitched 180 degrees to a -Z(sub-E) earth pointing attitude for instrument calibrations. This occurs periodically in the mission upon special request of the principal investigator.

Yaw Maneuvers. One of the most important spacecraft maneuvers occurs routinely every 30 days as the sun passes through the orbit plane. This maneuver requires a yaw turn of the spacecraft 180 degrees in magnitude to re-orient the solar arrays to the sun. This is required to maintain effective sun illumination of the solar arrays (to maintain power) and also to maintain thermal control of the ERBS. The maneuver entails a re-orientation of the spacecraft velocity vector direction from +X forward to -X forward, or vice versa. For beta angles of 90 to 180 degrees, a +X forward flight direction is required. Conversely, for beta angles of 0 to 90 degrees, a -X forward flight direction is required.

The optimum time to yaw the ERBS to maintain solar array illumination and thermal control is at a beta angle of 90 degrees. However, the solar arrays will continue to generate power up to 15 degrees beyond beta 90, and thermal control of the ERBS can be maintained for angles up to 25 degrees beyond 90.

Yaw maneuvers are essentially controlled by CSM commands, and the use of the RCS (thrusters) to nominally perform the turn. Special MCS activities are also required (wheel reversal and polarity changes). In order to accomplish these activities, special planning is required to ensure STDN support, avoid conflicts with other spacecraft events, and to generate the CSM load.

Yaw Maneuver Activities - TDRSS. Yaw maneuvers, controlled by the CSM, include the following activities:

- The spacecraft is switched from MCS-to-RCS control.

- The Scanwheels are reversed, requiring up to 15 minutes.

- The reversal process is a function of the non-essential bus voltage, and will vary accordingly.

- At 28 minutes into the event, the momentum wheel will be commanded to reverse, requiring up to 15 minutes to execute.

- Following momentum wheel reversal, there is a switch from RCS back to MCS control.

- After the switch and a brief settling period post maneuver activities begin.

Post maneuver activities include:

- Setting the ESSA controller up for the new flight direction.
- Turning off the catalyst bed heaters.
- Returning the ERBE-scanner to a normal scan mode.
- Reloading the SAGE-II azimuth registers.
- Returning the C&DH and RF subsystems to a normal operations configuration.

Backup Maneuver Scenarios. Various failures and performance problems may require that maneuvers be performed using other than the nominal methods. Various methods for each maneuver have been developed to accommodate many types of failures. Following are the backup methods for a three failure situations.

RCS Pitch 90, 270, 180 Maneuvers. All pitch maneuvers are performed using the MCS (except at altitudes below 400 km, which occurred during the early days of the mission only). However, if an MCS failure occurs, the RCS can be used. Command modules have been developed to pitch the spacecraft in 90 degree steps to the pitch 90, pitch 270, or pitch 180 positions for both the +X and -X forward flight direction. The command modules entail the following basic approach:

- Perform all pre-maneuver activities for instrument safing, TRU recording, and configuration.
- Use the MCE roll and pitch signals to drive the ACE gyrocompass during the maneuver.
- Include antenna handovers where applicable.
- Perform all the necessary post-maneuver activities such as instrument commanding, bias resets, etc.

These command modules assume the RCS is active, and rely upon the MCE roll and pitch signals to execute the maneuver. The implications of using the MCE signal are:

- Attitude Computer A failure voids use of these modules.
- The horizon scanners must be operating properly.
- The MCE must be configured properly for the flight direction (+X/-X) and the pitch reference (0, 90, 270, or 180).

Failure to meet any of these three conditions renders the RCS pitch maneuver command modules useless.

Hybrid Yaw Maneuvers. As a backup to the nominal RCS yaw maneuvers, a hybrid yaw maneuver method has been established. The term hybrid refers to the combined use of manual thruster firings and the MCS reacquisition mode. The method behind the hybrid maneuver involves:

- Using pulsed thruster firings to dump Y-axis momentum by scanwheel and momentum wheel spin down to idle and reversal activities,
- Commanding the CSM to the reacquisition mode once the wheels have been reversed and spun to idle,
- Waiting up to five orbits for the spacecraft to capture magnetically, and
- Recovering using the reacquisition mode procedure.

This method is somewhat time-consuming, and is difficult to recover from. But it does the job when the RCS cannot. Command modules are used to perform the maneuver.

Failed Thruster Modifications. If a thruster fails on-orbit, a backup thruster configuration will be resorted to.

The changes involve:

- Commanding the ACE to a backup thruster configuration, after power up.

- Modifying the command module execution times according to the type of failure.

In the case of pitch maneuvers, a backup thruster configuration actually results in a faster pitch. For yaw maneuvers, the opposite occurs, resulting in a longer yaw maneuver.

MCS Control Contingencies. The majority of the ERBS mission entails use of the MCS for normal nadir earth pointing (pitch 0) control. Failure to maintain this attitude to within specifications will disrupt science data processing, RF communications, thermal stability, and power collection. Therefore, the most important contingency methods developed for the ACDS involve controlled responses to limit violations on the MCE coarse/fine roll and pitch telemetry, and the gyrocompass rates and angle telemetry. These telemetry parameters indicate the attitude of the ERBS, and will be used as the primary indicators of MCS control performance. Contingency procedures which respond to attitude parameter limits have been developed and are summarized as follows:

- Attitude Computer A (off-line, backup MCE computer) fine roll and pitch error of 3, and 10 degrees, respectively, warrants use of the MCS Control Alarm Contingency procedure.

- Attitude Computer B fine roll and pitch errors and pitch errors of 3 degrees warrants use of the MCS Control Alarm Contingency procedure.

- Attitude Computer B Coarse roll errors of 7 degrees (3 degrees from auto-reacquisition mode initiation); coarse pitch errors of 10 degrees warrant emergency action (switching to RCS), and the use of the MCS Control Emergency procedure.

- A magnetometer failure warrants emergency action, and use of the MCS Control Emergency Contingency Procedure.

- Errors of 3 degrees and rates of 0.05 degree per second warrant use of the Gyrocompass Error Contingency procedure, which compares ACE results and takes appropriate action.

Communications and Data Handling Subsystem.

<u>General Description.</u> The C&DH subsystem is essentially a two-string redundant system with various cross-strap and dual operations capabilities. Figure 5 illustrates, in a simplified fashion, the C&DH capabilities and instrument interfaces. Figure 6 provides a view of the reliability relationships of the C&DH subsystem. The RF functions are sometimes referred to as a separate subsystem in order to group components and discuss interfaces.

The C&DH Subsystem provides:

- timing pulse generation, division, and distribution to spacecraft and instrument components,
- telemetry data sampling and formatting,
- command frame reception, validity checking, and execution,
- tape recorders for data storage and playback,
- switching of routine data to various components,
- signal conditioning,
- command storage memory capabilities,
- instrument interfaces, and

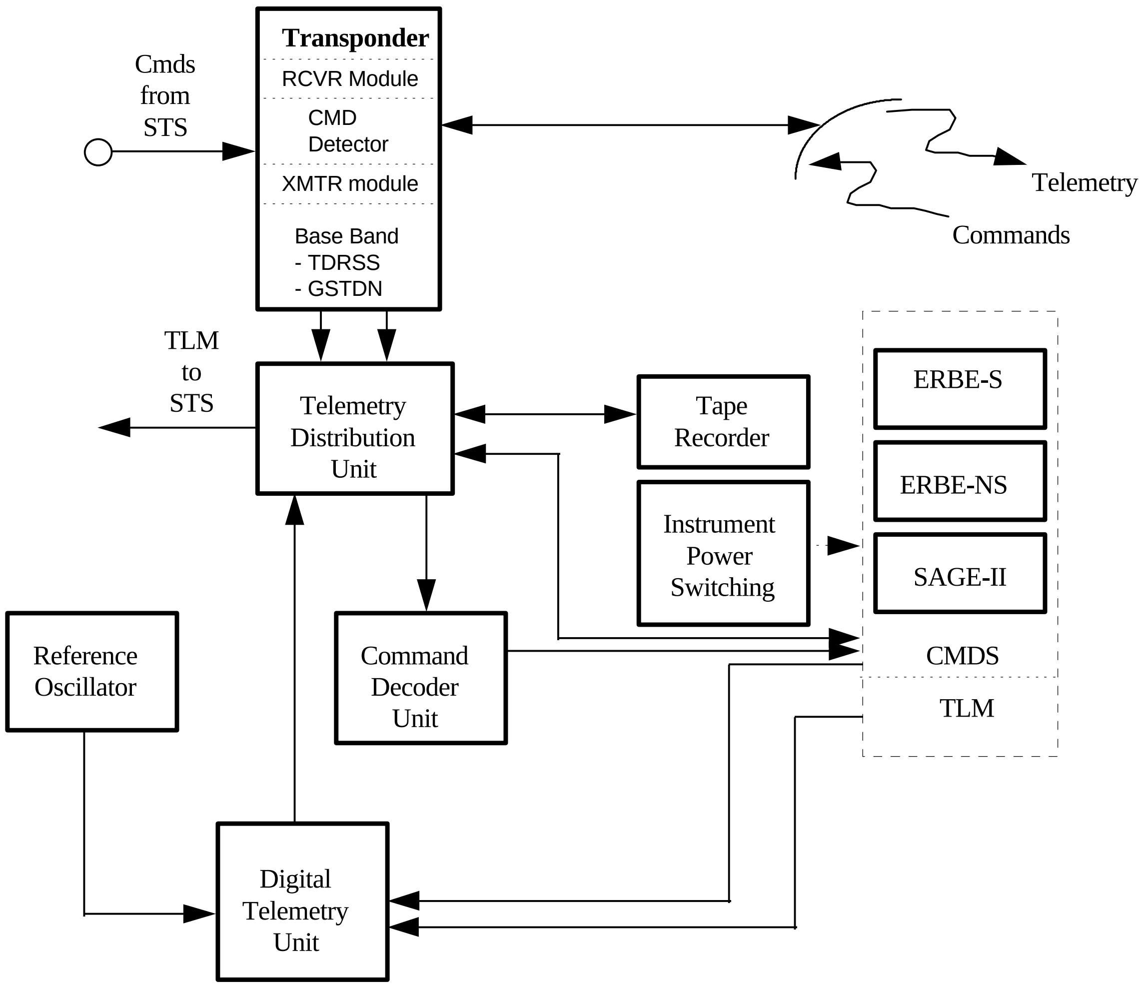

Figure 5. C&DH Subsystem Functional Diagram (Simplified)

- RF communications subsystem interfaces.

Normal DTU Functions. The C&DH design includes two Digital Telemetry Units (DTU's) which remain powered (active redundancy) and in use by operations. The DTU's provide:

- Sampling and analog-to-digital conversion for telemetry parameters from the spacecraft components and instruments.
- Telemetry data formatting into one of four formats, with some formats having rate selections.

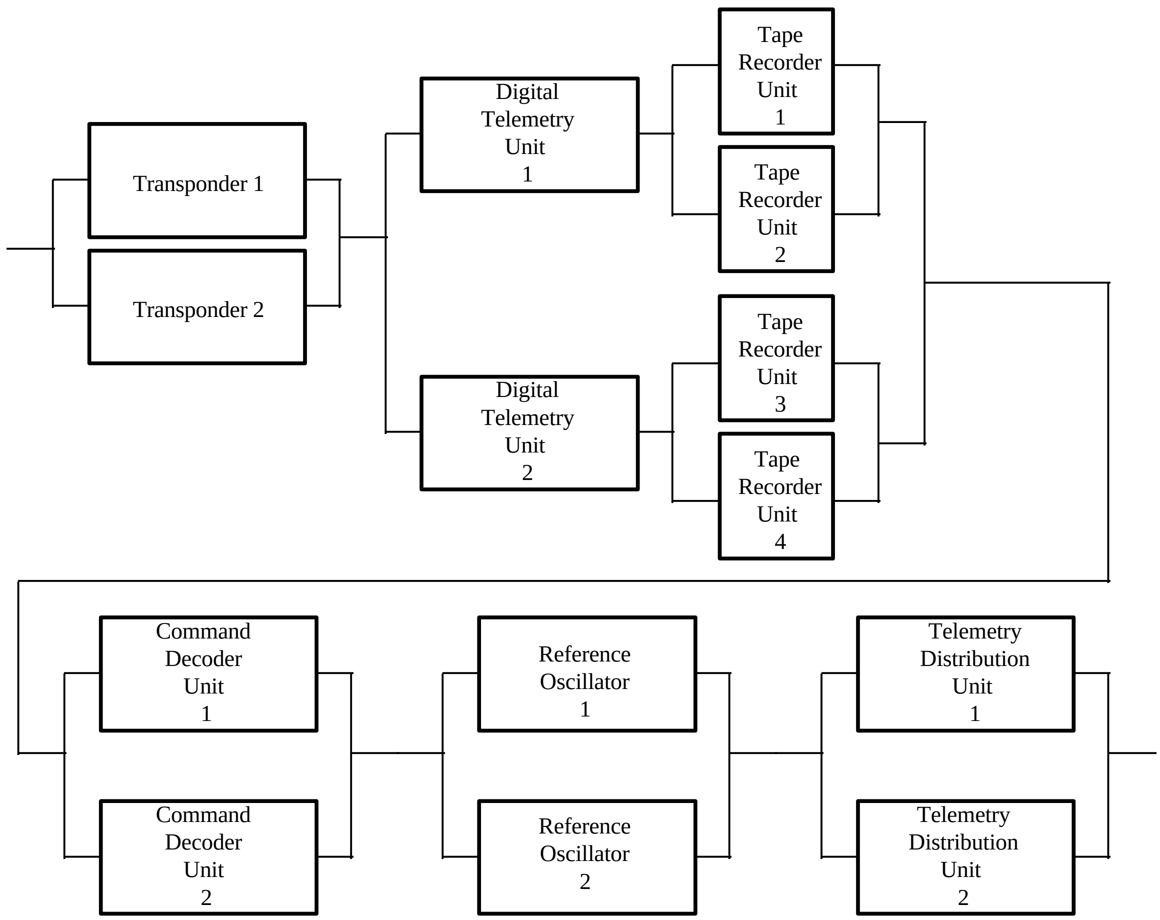

Figure 6. C&DH Reliability Block Diagram

The DTU's also house some timing chain circuitry, but this circuitry is regarded by operations as a function independent of DTU operations. Switching of the DTU telemetry data and DTU interfaces to other spacecraft components is handled by the Telemetry Distribution Units (TDU's). The primary function of the DTU's during the normal mission phase is to collect and format science data for temporary storage on a tape recorder unit (TRU). However, periodically it is required to load and dump the Command Storage Memory (CSM) and to check out the status of Electronically Steerable Spherical Array (ESSA). Accomplishing this task of formatting SAGE, ERBE, and CSM/ENG data with the two DTU's requires DTU format management. A series of command modules are used to cycle the DTU-2 format between the CSM/ENG and SAGE formats. DTU-1 is devoted to the ERBE format.

The Command Management program uses block memory command sequences to command DTU-2 to the SAGE format whenever a SAGE-II sunrise or sunset event is required. At the end of the SAGE-II event, the command sequence cycles DTU-2 back to the CSM/ENG format (1.6 kbps rate to maintain ground system configurations when the format switches). These command sequences will execute out of memory data specified SAGE-II event times, and generally require no intervention. Consequently, the DTU-2 will primarily be in the CSM/ENG format, except when SAGE-II events occur (twice per orbit for nominally 6 minutes, depending on the beta angle) while the DTU remains in the ERBE format. Using this method of DTU format management results in a simplified operating procedure. DTU commanding occurs out of CSM and does not require intervention. The CSM, which requires loading every 12 hours, cannot be dumped for comparison in real-time if the DTU is in the SAGE format. Therefore, CSM load and dump operations must be carefully prepared to avoid conflict with SAGE-II format.

The DTU is commanded to a particular format by the TDU command. The TDU-1 mode command establishes the format for DTU-1, and the TDU-2 mode command selects the DTU-2 format. In the event of a DTU failure, a backup scenario of DTU format switching has been defined. The basis of the backup approach is summarized in the following:

- The operational DTU will be switched to the Command Storage Module/ Engineering (CSM/ENG) format once every 12-hours to load and dump the CSM's. During the approximately 10 minute periods in the CSM/ENG format, science data (ERBE only) will be lost.
- the operational DTU will be switched to the "all-experiment" format during SAGE events (to collect ERBE and SAGE data simultaneously).
- During those periods when SAGE events are not scheduled, and CSM loading is not ongoing, the operational DTU will be switched to the ERBE format.

This backup scenario allows all mission objectives to be met with one DTU, including the collection of 95% of the science data on the TRU’s (assuming no major anomalies).

<u>TDU Functions</u>. The two TDU’s provided in the ERBS are redundant boxes which perform a variety of functions:

- Provide interfaces between the transponders and the Command Decoder Units (CDU's) (Transponder-1 to CDU-1, and Transponder-2 to CDU-2),
- Provide interface for and selection of DTU's to TRU's,
- Provide interfaces between DTU's and transponders,
- Provide interfaces between TRU's and transponders,
- Provide Flight Support Equipment interfaces from DTU's and TRU's to Shuttle avionics,
- Perform power supply conversions (bus voltage to regulated +5, +8 isolated and + 5 isolated),
- Provide TDU mode, temperature and Frequency Standard adjust telemetry,
- Perform latch and serial-to-parallel-convert serial digital command data from the CDU to the ERBE instruments,
- Test circuitry for testing high/low level discrete and serial digital commands, and
- Direct DTU format selections (TDU mode command).

The TDU's can be viewed as large switching centers, directing data paths inside the ERBS to meet the desired objectives. TDU-1 is used as the *prime* TDU. For those functions that can be approached from either TDU, TDU-1 is used. The Transponder-1/CDU-1/TDU-1 string of equipment is not cross-strapped with the Transponder-2/CDU=2/TDU-2 string of equipment. Therefore, to command a CDU, the associated transponder must receive the command via the selected antenna. Additionally, telemetry downlinking of data from a transponder is dependent upon the port selections established with the respective TDU. To command via CDU-1 or to load CSM-1, the Transponder-1 receiver must receive and validate the uplinked command; to command via CDU-2 or to load CSM-2, the Transponder-2 must receive and validate the uplinked command.

<u>TDU Switching to TRU Recording.</u> The TDU serves as the data path selector for TRU recording and playback functions. The record (data in) side switching options involve selecting a DTU and enabling an interface. On the record side, the desired record functions can be accomplished from either TDU-1 or TDU-2 (these circuits are completely redundant). Because TDU-1 will serve as the *primary* TDU, the TRU interfaces will be used to establish the tape recorder switching for data recording.

On the TRU dump (data out) side, the TDU switching required for a playback of TRU data involves selecting a transponder output port. The Transponder channels selectable for

TRU dumps are the TDRSS-Q channel and the GSTDN-1 playback port. These are the only two channels capable of receiving TRU playback data. TDU-1 is used to switch a TRU to a transponder port, it is switched to the Transponder-1 port only. Similarly, TDU-2 only switches TRU's to Transponder-2 ports. The TRU-to-Transponder data out switching is performed nominally eight times per day as TRU's 1, 2, 3, and 4 are cycled through four-orbit record periods and played backed to the ground accordingly. This switching is performed via a set of procedures which configure the port and the TRU for a dump operation.

The DTU telemetry data must be routed to two components in the ERBS:

1. TRU's for temporary storage of science data, and
2. Transponder for real-time channel downlink.

DTU-to-TRU selection was made early in the mission, and will be maintained throughout the life of the mission.

In summary, the DTU-to-TRU switching will involve:

- DTU-1 routed to TRU-1 and -2, selected in both TDU-1 and TDU-2.
- DTU-2 routed to TRU-3 and -4, selected in both TDU-1 and TDU-2.

The TDU-1 routing functions will be used by turning on the TDU-1 TRU interfaces. The TDU-2 routing functions will not be used (TRU interfaces OFF) unless a TDU-1 failure occurs. There are TDU mode command constraints applicable to both DTU-to-TRU selection and TRU interface on/off selection. The command constraint requires that all four TRU’s be commanded at the same time in the same TDU command.

<u>Tape Recorders.</u> The four tape recorders provided by ERBS are used extensively to capture science data and dump that data to the ground at regular intervals during each day. The tape recorder units require data playback to the ground after a four-orbit record cycle is completed. A TRU is available for playback every two orbits, for a total of 7 or 8 playbacks per day. The TRU dump is not allowed to exceed 4.5 hours after a record cycle ends.

The approach for operating the four ERBS tape recorders involves *ping-ponging* the record cycles for ERBE data between TRU-1 and TRU-2, and for SAGE-II data between TRU-3 and TRU-4. The basic concepts of this approach are:

- ERBE data (1.6 kbps) collection requires continuous recording of ERBE instrument and science data in either the ERBE or All-Experiment data formats. Nominally, this requirement will be satisfied by recording ERBE data on TRU-1 for four orbits (6.5 hours), then switching to TRU-2 recording for four orbits, while TRU-1 is dumped to the ground. The switch will include 10 minutes of overlapped data to allow for data acquisition at the beginning and end of the dumps (due to servo-lock times, acquisition times, frame sync lock in data processing centers, etc.).

- The 6.5 hours nominal ERBE record cycle uses about 33% of the available tape, leaving another two four-orbit record cycles if network support problems force a TRU dump.

RF Communication Subsystem. The RF communication subsystem is actually an element of the C&DH subsystem. But it is discussed here as a subsystem to facilitate grouping of RF components, highlighting functions and interfaces. The RF Communication subsystem is operated in several different configurations to accommodate a number of flight orientations. The RF Communications subsystem contains:

- Two standard NASA Transponders for RF communications with GSTDN and the TDRSS forward and return links.
- Two redundant power amplifier units for high power return link (telemetry data) transmissions to TDRSS.
- Two zenith antennas, consisting of one omni-directional single-element antenna and one Electronically Steerable Array (ESSA) antenna.
- One nadir omni-directional, dual element antenna.
- Two redundant ESSA beam control microprocessor units.
- Five RF switches for signal routing to and from the antennas, transponders, and power amplifiers.

In addition to these components, the RF subsystem contains C&DH subsystem interface circuits, cables, etc., to enable access to the telemetry data (real-time and TRU-stored data) and the Command Decoder Units (CDU's).

Transponder Operations. The ERBS design incorporates two NASA standard transponders, both of which are used to communicate via one of three antennas to TDRSS and to GSTDN sites. The transponders serve as the interface between the C&DH subsystem and the antennas, providing RF signal generation and reception.

Transponder Configuration. For the ERBS mission, the transponders are commanded to one of four configurations, for communications with either TDRSS or GSTDN, as required. The four configurations, 2 for TDRSS and 2 for GSTDN, are established on-orbit, as each situation requires.

- Transponder-1, selected to the zenith antennas, is set to the TDRSS NORM-2 configuration. This configuration is used for the majority of ERBS events.

- Transponder-2, selected to the nadir omni antenna, is set to the TDRSS NORM-2 configuration, for contingency and special events.

- When pitch 90 degree operations occur, both transponders are switched to the TDRSS NORM-2 configuration, and a handover from one omni antenna to the other will occur as the spacecraft track progresses towards, underneath, then beyond the TDRSS.

- For pitch 180 degree operations, both transponders are switched to the TDRSS NORM-2 configuration, and an antenna handover occurs as the pitch maneuver is performed. Then primary use of the nadir antennas occurs while stabilized in this position.

- When TDRSS one-way track is required (nominally once per day during pitch 0 degree operations), Transponder-1 is switched to the TDRSS NORM-1 configuration.

- When GSTDN events are scheduled during pitch 0 degree operations, Transponder-2 (nadir omni) is switched to the GSTDN configuration with the transmitter and playback channel turned on, as event times warrant.

- When GSTDN events are scheduled during pitch 90 degree and pitch 180 degree operations, both transponders are switched to the GSTDN configuration, and antenna handovers will occur as needed.

GSTDN-ONLY SUPPORT POSTURE

- Transponder-1, selected to the zenith omni antenna, is set to a GSTDN NORM-1 configuration .

- Transponder-2, selected to the nadir omni antenna, is set to a GSTDN NORM-2 configuration.

- The GSTDN Norm-1 configuration is maintained for all pitch configurations, except when communications to GSTDN via the zenith antenna is required. Then the GSTDN NORM-1 is used.

- The GSTDN-1 configuration is non-coherent (for one-way tracking). It is cycled to coherent during GSTDN events at the completion of one-way tracking, and prior to two-way tracking. It is then recycled to non-coherent just prior to the next GSTDN event.

One item driving the GSTDN transponder configurations is a restriction on simultaneous TRU playback, and two-way ranging operations. A typical GSTDN event starts out with a TRU dump, which prohibits ranging. For these "typical" events, a non-coherent initial configuration is employed, and one-way tracking is scheduled—for Temperature Controlled Crystal Oscillators (TXCO) frequency profile samples.

Transponder configurations are driven by the STDN event and tracking schedule. Transponder configurations are programmed into daily CSM loads. The CSM executes the reconfiguration commands approximately two minutes prior to an event, and cycles the transmitter at AOS [9] minus two minutes for GSTDN events, and at E+20 [10] seconds for TDRSS events. Transmitters are turned off by CSM at the end of each event.

Power Up and ON/OFF Sequences. The ERBS transponders were powered up in the Shuttle bay, and are nominally powered throughout the mission life. Power up is accomplished for the transponder receivers and transponder transmitters separately, with each receiving its power from a different spacecraft bus. The receivers, which house the transponder configuration mode control word, are powered directly from the essential bus (sometimes called the main bus) while the transmitters receive power from the non-essential bus (NEB).

Receiver and transponder mode control power up occurred for both transponders in the Shuttle bay at the initiation of the in-bay checkout. When the Shuttle crew enabled the ERBS main bus, both transponder receivers became active and available to accept command input (RF and hardline from Shuttle flight support equipment). The transponder configuration mode control word (a digital register in the receiver logic), upon main bus power up, establishes the transponder power up preset state. This state defined the initial transponder configuration to respond to command input reconfiguration, as needed. With the receivers and transponders configuration logic, powered by the essential bus; there is no requirement to perform any action to maintain power, even during spacecraft undervoltage or overcurrent conditions. The receivers remain powered facilitating command capabilities even during times of spacecraft anomalies.

The transponder transmitters, on the other hand, are powered by the Non-Essential Bus (NEB) and must be commanded on. The NEB power is routed to the transmitter module by way of one of a pair of transmitter inhibit relays and a commandable transmitter bus relay. The inhibit relays were designed in-line for Shuttle safety purposes. Once one of

[9] Predicted acquisition of signal.
[10] E = Scheduled TDRSS Event Start Time

Table 5. Transponder Power Up Sequence

Action	Purpose	Remarks
1. Main Bus on.	Powers both receivers and mode control logic.	Shuttle crew function, resulting in power up preset state.
2. PNUVOC Command.	Enables non-essential bus.	Performed in Shuttle Bay.
3a. NEI PREARM set.	Enables NEB power to the transmitter.	Shuttle crew function, primary method.
3b. Timer Relay Closed.	Enables NEB power to the transmitter bus.	Backup method to 3a NEI prearm, after Shuttle release.
4. XP1ON/XP2ON Commands.	Enables transmitter bus power.	Performed prior to Shuttle release, then remains on.
5. XP1/XP2 Commands with subfields.	Configures the transponder mode (receive and transmit), and turns on the transmitter.	Steps 1 through 4 occurred prior to Shuttle release and remain set throughout mission life. Only step 5 required for each transmission. Steps 2 through 4 are recycled if overcurrent or undervoltage occurs in the NEB.

the inhibit relays is closed, NEB power is passed to the transmitter bus. The power to the transmitter module can be enabled and disabled by ground command. Both transmitter bus relays were enabled prior to Shuttle deployment and will nominally be left enabled throughout the mission life.

Disabling of the transmitter bus is only performed in the event of a NEB undervoltage or overcurrent, or a failure in the transmitter module itself, which would render it useless. With power applied to the transmitter module via the transmitter bus, the transmitter may be turned on in one of three modes (STDN high modulation index, STDN low modulation index, and TDRSS) by the transponder mode control command. All transponder configuration activities are driven by the STDN schedule, and are controlled by the command management program.

Power Amplifier (PA) Operations. The ERBS contains two power amplifiers (a prime and a backup) that provide a nominal 28 Watts of increased power for the selected RF transmission. Operations use the power amplifiers as follows:

- All ESS antenna communications will require power amplification. This encompasses a nominal use of the PA for 7 to 8 25-minute TDRSS MA events and one 15 minute TDRSS MA event per day if in a TDRSS-only or GSTDN/TDRSS mixed support mode.

- The increased field-of-view by the Omni antennas when output power amplification occurs, plus the increased link margins warrant use of the power amplifier for Omni communications with TDRSS (within thermal constraints). Use of the power amplifier is not required in these situations (maneuvers) but adds to the overall communications capability, when most needed (during critical spacecraft attitude and orbit maneuvers).

- Power amplification is not used for GSTDN communications.

The Power Amplifier design provides the following:

Primary and Backup Power Sources

- Only one PA can be selected for power and use at a given time.

- The selected PA can receive power from a primary power source and a backup power source.

- The primary power source contains a thermostat control circuit that shuts off power when 50 degrees C is detected in the active PA. When the power is shut off, the power relay is opened, and the telemetry bi-level will changes states.

- The backup power source does not contain a thermostat control circuit and will not shut down backup power, regardless of the PA temperature.

- When the PA is powered, it will remain in the standby mode (active redundancy) until the selected transmitter is turned on. In the standby mode the PA draws less than one amp of NEB current. When on, it draws approximately 6 amps of NEB current.

- Only primary power is used. Backup power is used only if a failure in the primary power or the thermostat cutoff circuitry is detected, or if PA use is required despite the PA temperature (the latter of which is only performed with observatory engineer concurrence).

- The primary power was turned on in the early phase of the mission and will remain on throughout the life of the mission, barring any anomalies. Only the transmitter will be cycled on and off. This on/off-cycling saves 6500 cycles of the relay every two years of mission life, and also saves CSM space.

- If backup power is used, the temperature is closely monitored.
- Primary and backup power is not enabled at the same time.
- Leaving the primary power on at all times is not expected to cause any power or thermal problems.

PA Thermal Considerations. The ERBS power amplifiers generate a lot of heat and increase in temperature fairly rapidly. A copper plate was added to the RF plate to achieve a maximum temperature of about 40 degrees C at the worst case on-off-on cycle. Continuous operations up to 60 minutes achieved approximately 41 degree C maximum temperature.

Zenith Omni Antenna. The Zenith Omni antenna, located at the top of the spacecraft next to the Delta V thrusters are used in the following manner:

- Communicating with TDRSS and GSTDN during orbit transfer burns.
- Communicating with TDRSS and GSTDN during routine attitude and orbit maneuvers
  (yaw maneuvers, instrument calibrations, and orbit trim operations).

Nadir Omni Antenna. The Nadir Omni antenna, located on the +Z (sub-E) side of the spacecraft, is used in the following situations:

- Communications with TDRSS during attitude and orbit maneuvers,
- Communications with TDRSS during nominal operations, on a special basis only (when C&DH String-2 access is required, such as CSM-2 block memory contingency sequence loads, Frequency Standard-2 adjustments, CTD-2 clock set operations, etc.), and
- Communications with GSTDN.

As you can see, under this scenario, the Nadir Omni antenna is used frequently.

ESSA Antenna. The Electronically Steerable Array (ESSA) antenna is a unique antenna which provides the primary communications link with TDRSS for normal operations support.

ESSA functional Description. The functional description includes the functional components, controller/diver selection and operational use of the antenna.

ESSA Functional Components. The ESSA antenna is a system comprised of a microprocessor controller, an antenna driver, a switching power divider, and the ESSA dome, complete with 145 elements. The controller and driver components have backup units, with the ability to select one set or the other. The controller is a microprocessor unit that contains firmware (Programmable Read Only Memory, PROM) to control the driver and Switching Power Divider (SPD) in beam pointing operations. It operates in one of several modes and requires ephemeris data loads to properly steer the beam. The driver controls the SPD to activate and deactivate the elements, in response to controller commands. The SPD provides power switching to 12 of the 145 elements to select and generate an RF beam (26 degree beamwidth). The dome houses the 145 elements arranged in a semi-sphere, capable of beam pointing down to 90 degrees in the Theta angle, and 360 degree around the dome (Phi angle).

Controller/Driver Selection. Selection of a string for use is accomplished by high level discrete command. When a string is selected, the controller and driver are powered up and automatically set to the initialization mode. If not selected, the controller and driver remain unpowered. The Phi/Theta and 160-bit driver telemetry is provided in this downlink by the active controller string only. Separate commands exist for each controller/driver string. However, if the controller is not selected, it will not respond to commands (no power). After selection, a string requires a 5-second wait before commanding the selected controller.

String-1 and Backup. For the normal operations phase of the mission, String-1 will be selected and used throughout the mission life, barring no anomalies. If an anomaly is detected and suspected of being a controller or driver component, a switch to String-2 is performed (redundant).

ESSA Antenna use in Mission Operations. The ESSA antenna provides the primary link to TDRSS during normal Mission Operations. However, due to the nature of the ESSA design, it cannot be used in all applications. The following describes the use of the ESSA during flight operations:

- After deployment from the Shuttle, the ESSA became the primary link to TDRSS.
- ESSA pointing is strictly dependent upon a nominal Nadir earth-pointing attitude. As such, ESSA is not used during orbit or attitude maneuvers.
- During normal science data collection at mission altitude, the ESSA is exclusively used for the 7 to 8 TRU dump events and the SAGE monitor event each day (when in TDRSS-only or GSTDN/TDRSS mixed support posture).

- ESSA is not used with GSTDN.

- When used, ESSA beam pointing occurs within the entire 0-90 degree Theta range.

- When using the ESSA, the high power transmitter path is used, nominally with Transponder-1. RF switching to configure for ESSA use will primarily involve two switches: the Omni/ESSA select switch, and the antenna high power select switch. The ESSA will not be selected as the Zenith antenna. Instead, the Zenith Omni is selected for routine operations until TDRSS can be selected.

<u>RF Switch Operations.</u> The RF Communications Subsystem contains five switches which serve the role of data path selector for routing telemetry data and commands to and from the transponders and antennas.

Transmitter Switching. Selecting a transmitter output path requires three switch settings, and if the Zenith antennas are to be used, a fourth setting. The first switch encountered, when transmitting, is the transmitter high power select switch. This switch selects either Transmitter-1 or Transmitter-2 for high power output. By default, the transmitter not selected for high power output is routed to the low power transmitter path. The PA switch and the high power antenna switch will not operate when either the NEB is off (undervoltage or overcurrent), or the transmitter bus is disabled.

Along with the high power output path is a PA select switch. PA-1 is to be selected and used throughout the mission, barring any anomalies. When this switch is commanded, two relays respond to select PA input and output, as illustrated.

Both the high power and low power paths terminate at an antenna select switch. At this switch, the high power path is routed to either the Nadir or Zenith antennas, and by default, the low power is routed to the opposite. A high power antenna selection results in a low power path to the opposite antenna.

Once beyond the antenna select switch, the Zenith antennas have an additional switch. This switch selects either the ESSA antenna, or the Omni antenna.

These four switches fully select both the high power and low power data paths for both transmitters.

Several constraints exist when operating these switches, as follows:

- An RF transmit switch is never commanded while either transmitter is on. This activity called "hot switching" severely reduces the life of the switch. The PA must be on in order to transmit data via the high power path.

- On-orbit operations will not see RF transmit switching very often. The normal configuration is to remain constant except in maneuver situations. In maneuver situations, the switching is normally executed by the CSM.

Receiver Switching. Receiver Switching is simple when compared to transmitter switching. It only involves two switches. The first switch selects Receiver-1 or -2 to the Zenith antenna. As was the case with transmitter switching, selecting a receiver to the Zenith antenna results in a default setting of the Nadir antenna to the opposite receiver. The commands reflect the Zenith antenna/receiver selection. The second switch selects either the ESSA or the Omni antenna for use on the Zenith side.

Operationally, the receiver/antenna select switch is not used during flight operations—to avoid a single-point failure. If, for example, the switch failed in a way that neither of the two relay contacts were activated, spacecraft RF commanding would not be possible, and the mission could be lost. To avoid the potential for this happening, the switch is not used, and in addition, it is commanded daily by CSM commands to ensure a solid contact.

The normal flight settings for these RF switches are shown in table 6. These settings are not expected to change except in maneuver situations, and are used then after careful planning and consideration.

Table 6. Normal RF Switch Settings

Switch	Normal Settings
Transmitter High Power	Transmitter-1 High Power
PA Select	Power Amp-1
Antenna High Power	Zenith Antenna High Power
Zenith Antenna Select	ESSA Antenna
Zenith Antenna Receiver	Receiver-1

Electrical Power Subsystem (Power)

The ERBS power system is designed to provide power to support ERBS mission functions throughout its mission life. This subsystem is composed of the following major components:

- Two 50-amp Batteries
- Standard Power Regulator Unit (SPRU)
- Amp-Hour Meter Unit (AHMU)
- Power Monitor Unit (PMU)
- Power Control (PCU)
- Solar Arrays
- Essential Bus (ESS)
- Non-Essential Bus (NEB)
- Control System Bus (CSB)

Functional and reliability aspects of the power subsystem are diagrammed in figures 7 and 8.

<u>Batteries</u>. Power is generated by the two solar arrays and is stored in the batteries.

Current Sensor Performance Accuracy. Each battery is provided with two current sensors that measure charge and discharge currents of the battery (one high and one low current sensor). The low current sensors are high resolution sensors, but saturate at 2.92 amperes. Displayed values do not reflect charge of discharge identifiers (polarity—negative for discharge and positive for charge). The high current sensors are coarse-resolution sensors and are displayed with charge or discharge identifiers.

Battery Voltage. The battery voltage is one of the most important telemetry parameters of the Electrical Power Subsystem, in that it gives a direct indication of any anomalous conditions related to Power. Therefore, it is monitored closely during real-time operations so that the voltage monitor reflects an accurate value at any given time.

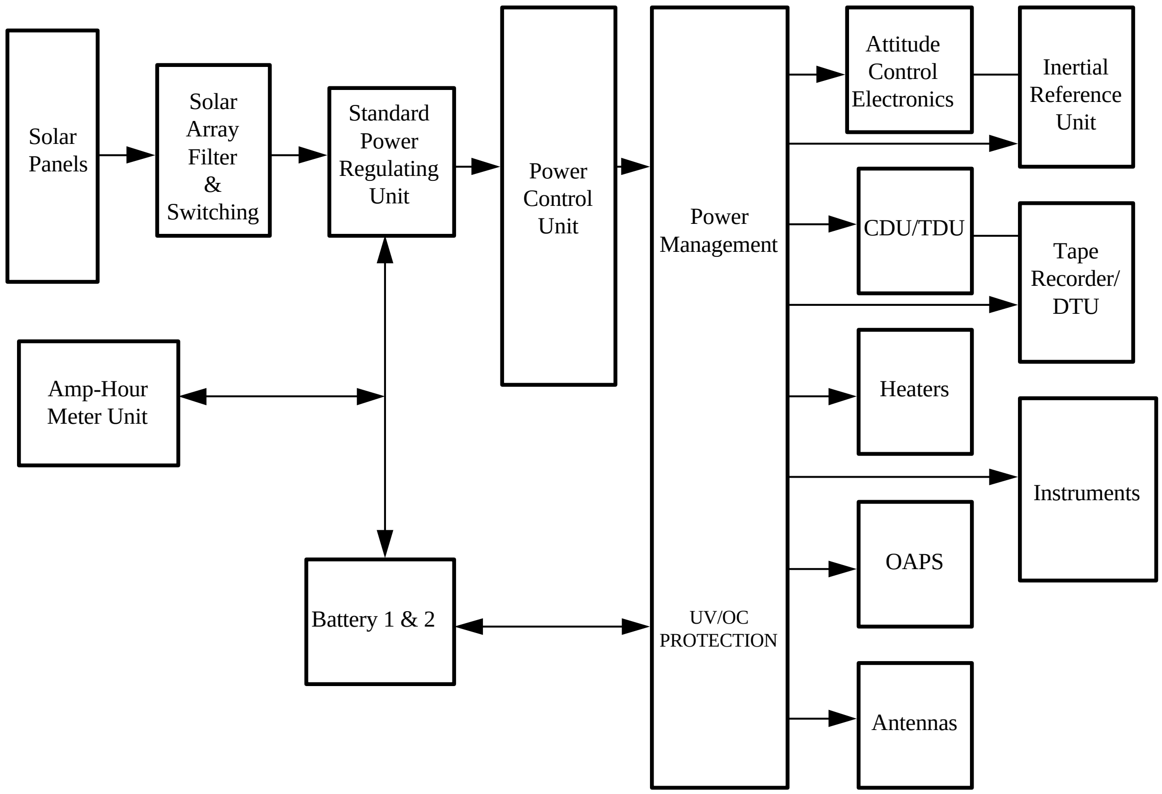

Figure 7. Power Subsystem Functional Diagram (Simplified)

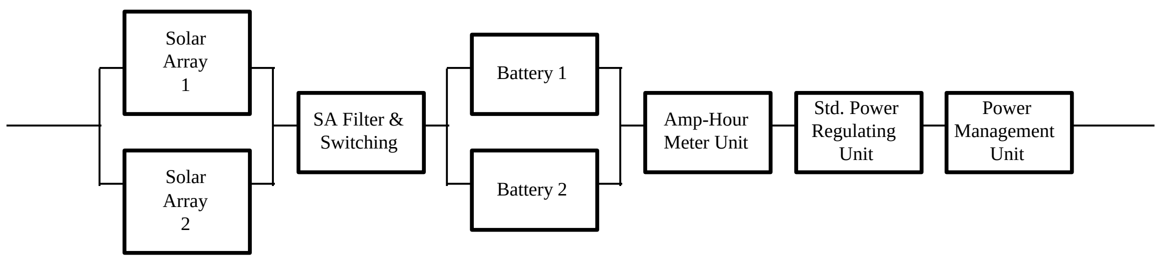

Figure 8. Power Subsystem Reliability Block Diagram

Standard Power Regulator Unit (SPRU). The Standard Power Regulator Unit (SPRU) controls the power regulation. It controls the flow of current from the solar arrays to the batteries and to the main bus. The Standard Power Regulator Unit (SPRU) regulates the amount of energy drawn from the solar arrays, depending on the spacecraft loads, power requirement and the amount of energy needed to effectively charge the batteries. To effectively regulate the power drawn from the solar arrays under any spacecraft conditions, the SPRU has auto-mode control functions using feedback from the batteries, the PMU, the AHMU, and internal feedback.

The SPRU has eight voltage/temperature settings (Levels 1 through 8) that are commandable. These VT settings provide control of battery charge acceptance utilizing battery voltage and the temperature of the hotter battery. The tolerance for these levels is with respect to Level 8 at 0 degrees. C.

During orbital operations, settings of the SPRU VT level vary depending on a combination of RF PA use (TDRS support) and the beta angle. SPRU VT level will be set to maintain battery charge to sustain normal spacecraft load. Through substantial orbit simulations and computer modeling, preflight VT level settings for different conditions have been established.

Amp-Hour Meter Unit (AHMU). The charge-to-discharge counter (C/D) in the AHMU requires battery temperature input. The mission was started using Battery-1 as the temperature source. However, if Battery-2 tends to run warmer, it will be selected.

The AHMU has four commandable charge to discharge ratio (CDR) levels, Level 1 being the lowest, and Level 4 the highest. The CDR is the amount of charge into the batteries as a function of the discharge, battery temperature, and commanded level. The four C/D levels are selectable by commands. Upon power up of the CSB, C/D Level 1 was preset.

Power Monitor Unit (PMU). The Power Monitor Unit has several fail-detect circuitry and automatic responses for failures or anomalous conditions in the Electrical Power Subsystem. For the automatic responses to take affect, several latching relays need to be closed. Power failures and anomalous conditions detected by the PMU response to the failure detected, require commands to close the latching relays to effect the PMU failure responses, and telemetry verification of the commands. For normal mission operations, all the latching relays to effect the PMU responses will be closed to protect the power subsystem from further damage, in the event of a failure or anomalous condition in one of the components.

Electrical Power Subsystem Bus Operations. The Electrical Power Subsystem has several buses that feed current to the different spacecraft boxes. The three commonly mentioned buses are the:

- Essential Bus,
- Non-Essential Bus, and
- Control System Bus.

These three buses are supplied with current from the solar arrays or batteries through the Main Bus. Two additional buses branch out from the NEB. These are the transmitter/primary Non-Explosive Initiator (NEI) and the propulsion/backup NEI bus.

Main Bus. The Main Bus, as the name implies, supplies power for all the spacecraft boxes through the other buses. This bus remains closed throughout the life of the mission, except during launch when it was disconnected (together with the solar arrays) through the Orbiter Standard Switch Panel (SSP). There are no ground commands to open or close the Main Bus, only the SSP had the switches to serve these functions.

Essential Bus. The Essential Bus provides power to several spacecraft boxes that require power, regardless of the spacecraft configuration. As such, this bus cannot be disabled, except through the Main Bus disconnect function of the Orbiter SSP. Table 7 shows the different components that are powered through the Essential Bus. The Essential Bus is not protected from undervoltage and overcurrent conditions.

Control System Bus. The Control System Bus (CSB) provides power to the Primary Attitude Control System (which is the MCS), hydrazine and instrument survival heaters, and the ESSA. It also provides power to some parts of the PMU and the PCU, and the 28V Regulator failure circuitry. A complete list of the components, powered through the CSB, is provided in table 7.

To activate the NEB overcurrent protection, the circuitry has to be enabled. Like the CSB, the NEB has a bypass relay that serves as a backup to the undervoltage and overcurrent relay. The bypass relay will only be used in the event of a failure in the undervoltage protection circuitry, inhibiting the closing of the undervoltage and overcurrent relay, or during anomalous conditions in the power subsystem.

Transmitter/Primary NEI Bus. The transmitter/primary NEI bus is one of two busses that branch out of the NEB to power several spacecraft components. This bus serves as a power relay to the transmitters, and as the primary Non-Explosive Initiator (NEI) bus for the spacecraft appendages.

Propulsion/ACE/Backup NEI Bus. The propulsion/ACE/backup NEI bus provides power to the spacecraft propulsion system, the Attitude Control Electronics (ACE), and serves as the backup NEI bus for the spacecraft appendages. The propulsion/ACE/backup NEI bus will remain on throughout the life of the mission.

Table 7. Spacecraft Bus Table

NON-ESSENTIAL BUS	CONTROL SYSTEM BUS	ESSENTIAL BUS
Digital Telemetry Units (DTU) (2)	ESSA Controllers (2)	Command Decoder Units (CDU) (2)
Tape Recorder Units (TRU) (4)	ESSA Antenna Driver	Telemetry Distribution Units (2)
Transmitters (2)	Magnetometer	Frequency Standards (2)
RF Power Amplifiers (2)	Torqrods (4)	Receivers (2)
Attitude Control Electronics	Magnetic Control Electronics	Power Control Unit
Inertial Reference Units (IRU) (2)	Momentum Wheel	Power Monitor Unit
Propellant Valves	Scanwheels (2)	Standard Power Regulator Unit (SPRU)
Thruster Valves	Sun Sensors	Wire Harness
Battery Heaters (2)	Propulsion Mounting Plate Heaters (7)	Batteries (2)
Catalyst Bed Heaters (8)	AMP Hour Meters (2)	
Deck Heaters (2)	Battery Shunt Assembly	
Propellant Valve Heaters (8)	Power Control Unit (PMU)	
Power Control Unit (PCU)	Power Monitor Unit (PCU)	
Power Monitor Unit (PMU)	28V Instrument Reg. (4)	
15V Regulators	Wire Harness	
ERBE Non-Scanner	ERBE Non-Scanner Survival Heaters	
ERBE-Scanner	ERBE-Scanner Survival	

NON-ESSENTIAL BUS	CONTROL SYSTEM BUS	ESSENTIAL BUS
	Heaters	
SAGE-II		
SAGE-II Heater		
Wire Harness		
28V Regulators (4)		
REMARKS: Bus is protected from undervoltage and overcurrent.	REMARKS: Bus is protected from undervoltage and overcurrent.	REMARKS: Bus cannot be disabled, except through the Orbiter Interface Box (OIB), RMS, and Ground Test Connector.

Orbit Adjust Propulsion Subsystem (Propulsion)

ERBS propulsion system consists of tanks, valves, and thrusters. Heaters are also included in the design for the thermal control and proper functioning of the propulsion system. Figures 9 and 10 provide functional and reliability diagrams of the Propulsion subsystem. Figure 11 provides a diagram of the thruster drivers, reflecting a redundant reliability relationship linked to the three axes.

The *primary* mission requirement for this subsystem was to provide the impulse required to reach the final orbit altitude of 610 kilometers from the starting orbit provided by the STS. This event occurred prior to the normal operations phase.

The Propulsion mission requirements during the normal operations phase are:

- Perform 180 degree yaw turn maneuvers for solar array illumination for a two year mission (25 sequences).
- Perform orbit trim burns as required for a two year mission (594 to 610 km).
- Provide total backup attitude control for the spacecraft Magnetic Control System (MCS) for one year.

<u>Yaw Turn Maneuvers.</u> The 180 degree yaw turn maneuvers are required monthly throughout the mission to maintain sun illumination on the solar arrays. The

maneuvers will be performed by the Propulsion thrusters operating in the RCS mode through the ACE. These maneuvers will be performed in real-time to the degree possible, as dictated by GSTDN/TDRSS coverage.

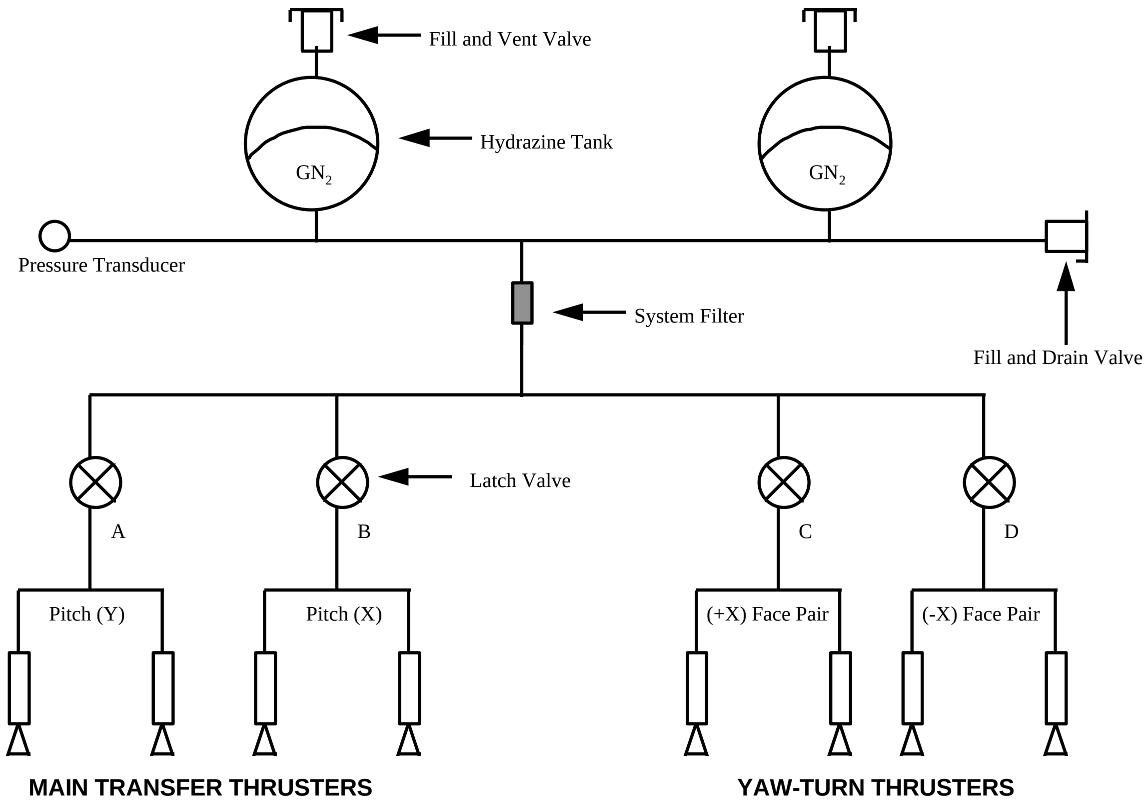

Figure 9. Propulsion Subsystem Functional Diagram (Simplified)

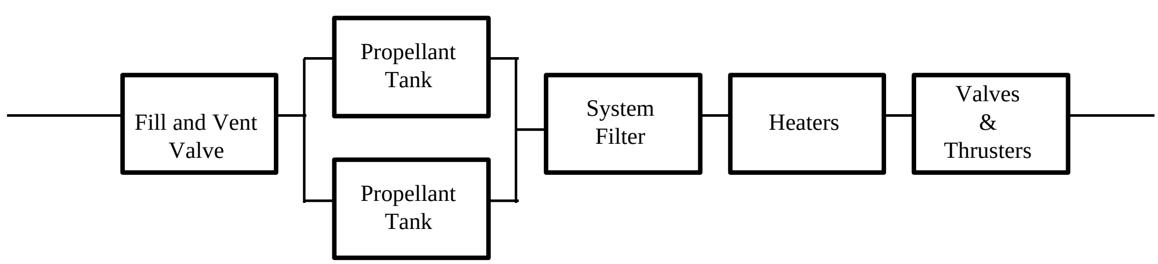

Figure 10. Propulsion Subsystem Reliability Block Diagram

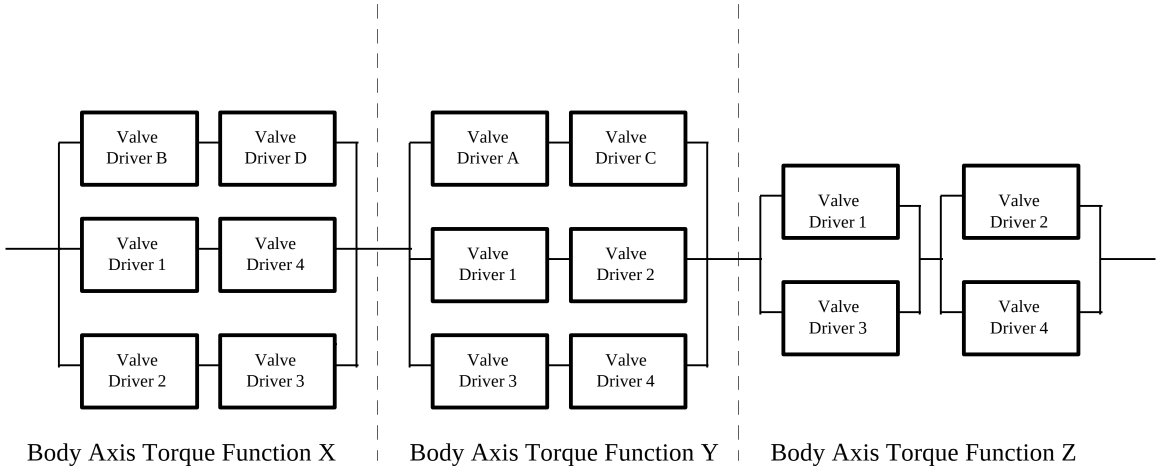

Figure 11. Thruster Valve Drivers Reliability Block Diagram

Orbit Trim Burns. Orbit trim burns will be required every two or three months throughout the mission to maintain the orbit altitude between 594 to 610 kilometers. These burns will be performed by the Propulsion in the RCS mode through the ACE.

The required 90 degree pitch maneuvers will be performed prior to and after the trim burn by the MCS.

Backup ACDS Functions. The Propulsion subsystem provides total backup to the ACDS MCS in the event of failure. Adequate propellant was originally available to handle all MCS functions for a minimum of one year. This included pitch maneuvers, and all pointing control.

In the event of RCS failures, the Propulsion subsystem can be activated manually rather than being driven by the ACE. The RCS has built-in-contingencies for thruster failures. However, in the event of serious or double RCS failures, manual thruster firing was planned.

Thermal Subsystem

The Thermal Subsystem contains thermal blankets, louvers, a radiator, and a variety of heaters which maintain thermal stability within the specified ranges. Most of the thermal subsystem components are passive devices. The heaters are critical to successful ERBS operation.

Thermally Critical Components. The ERBS contains many electrical, mechanical, and fluid components, all of which have operating limits and extremes. The Thermal subsystem is designed to maintain each component within the predicted maximum and minimum temperature limits.

Although all of the ERBS components have potential operating anomalies when the temperature limits are exceeded, certain components are more critical than others. Design emphasis was placed on these components to ensure adequate attention for temperature control. These components are Batteries, Tanks, Lines, Hydrazine, Inertial Reference Units, and Transponders.

Batteries. The ERBS batteries (two 50 amp-hour batteries) are large, relatively massive components which tend to change temperature slowly. Battery temperatures affect the overall battery performance, and can actually result in an explosive reaction should the temperature and charge reach high levels. Battery charge levels actually vary with the temperature, with higher temperatures restricting higher charge levels. This reduces the overall power available to the spacecraft. Therefore, maintaining the battery temperature limits (nominally 0 to 20 degrees C) is important to the mission as a whole.

Tanks, Lines, and Hydrazine. The ERBS Propulsion is a hydrazine blow-down system which relies upon system pressure to force hydrazine through the valves, resulting in combustion that then torques on the spacecraft body. The pressure in the system is related to temperature, and the efficiency of the thruster "burns" is variable due to temperature. In addition, liquid hydrazine freezes at 4 degree C. Contraction and expansion properties of the hydrazine have the potential to crack the hydrazine lines, thruster valves, etc. Propulsion is therefore a thermally critical subsystem of ERBS, residing mainly in the keel module, and must be controlled and monitored.

Inertial Reference Units (IRU's). The ERBS ACDS contains two IRU's, which serve an important role in attitude control maneuvers, and attitude determination. The IRU's are extremely sensitive devices that measure spacecraft motion (rate/time) about the X, Y, and Z axes. The temperature of the IRU's affects IRU rate signals (drifts and noise levels), reducing the overall accuracy of the system. IRU temperature control is important in determining IRU biases.

Transponders. The two ERBS transponders contain Temperature Controlled Crystal Oscillators (TCXO's) which are used many times daily in communication (RF) with the ground. The TCXO frequency varies with temperature, and can pose frequency shift problems during RF signal acquisitions. Because the ability to communicate is dependent on frequency within +/- 700 Hz, the transponder temperature becomes important to spacecraft operations

Appendix 6. Hours of Operation

Spacecraft Name	Launch_date	Term_date	Hours
AEM-A	4/26/78	9/30/80	21312
AEM-B	2/18/79	11/18/81	24096
AMPTE/CCE	8/16/84	7/14/89	43032
CGRO	4/5/91	9/30/96	48120
COBE	11/18/89	7/1/93	31704
DE-1	8/3/81	3/1/91	83928
DE-2	8/3/81	7/1/93	104400
ERBS	10/5/84	9/30/96	105072
EUVE	6/7/92	9/30/96	37824
GOES-2	6/16/77	1/1/79	13536
GOES-3	6/17/78	4/28/86	68928
GOES-4	9/9/80	3/10/87	56952
GOES-5	5/22/81	7/18/90	80256
GOES-6	5/22/81	7/1/93	106152
GOES-7	2/26/87	7/1/93	55608
GOES-8	4/13/94	9/30/96	21624
GOES-9	5/23/95	9/1/96	11208
HST	4/24/90	7/1/93	27936
ICE (ISEE-3)	8/12/78	7/1/93	130488
IMP-8	10/25/73	9/30/96	201024
ISEE 1&2	10/27/77	9/26/87	86904
IUE	1/26/78	7/1/93	135240
LANDSAT-3	3/5/78	9/5/83	48240
LANDSAT-4	7/16/82	7/1/93	96072
LANDSAT-5	3/1/84	7/1/93	81816
NIMBUS-7	10/24/78	7/1/93	128736
NOAA-10	9/17/86	7/1/93	59496
NOAA-11	9/24/88	7/1/93	41784
NOAA-12	5/14/91	9/30/96	47184
NOAA-14	12/30/94	9/30/96	15360
NOAA-6	6/27/79	3/3/87	67344
NOAA-7	6/23/81	6/5/86	43392
NOAA-8	3/28/83	1/1/86	24240
NOAA-9	12/12/84	11/8/88	34248
SAMPEX	7/3/92	9/30/96	37200
SMM	2/1/80	11/1/89	85464
TDRS-1	4/4/83	7/1/93	89784

Spacecraft Name	Launch_date	Term_date	Hours
TDRS-3	9/29/88	7/1/93	41664
TDRS-4	3/13/89	7/1/93	37704
TDRS-5	8/2/91	9/30/96	45264
TDRS-6	1/13/93	9/30/96	32544
TDRS-7	7/13/95	9/30/96	10680
TIROS-N	10/13/78	2/27/81	20832
UARS	9/15/91	9/30/96	44208
WIND	11/1/94	9/30/96	16776
XTE	12/30/95	9/30/96	6600