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BOEING

Final Report
Contract NAS2-8526

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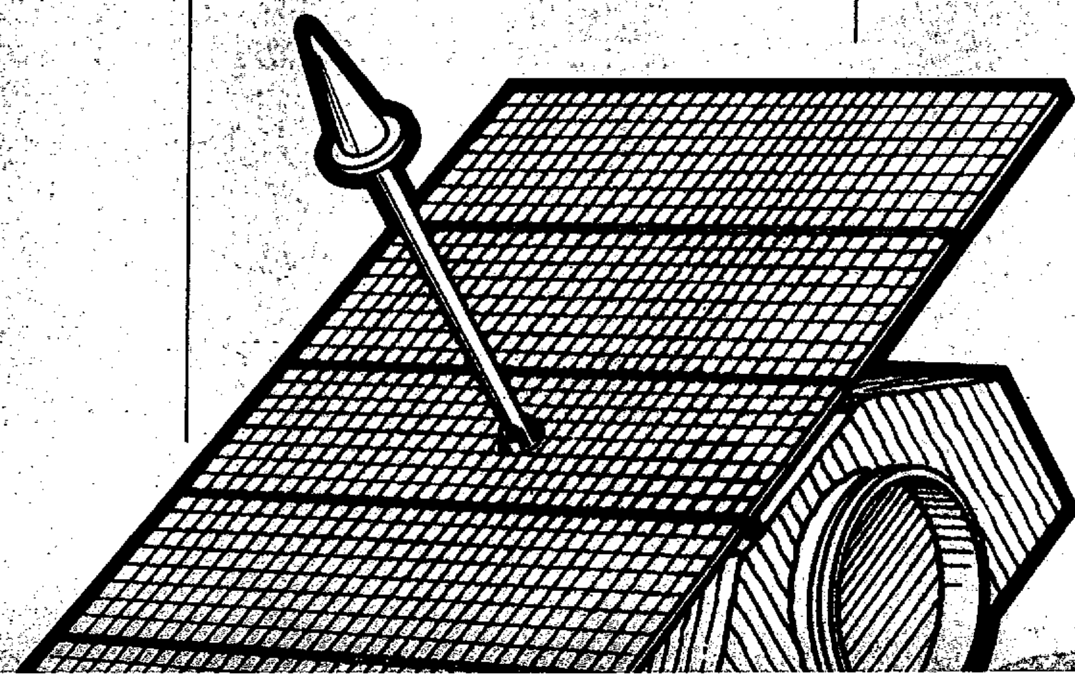
BSRM Design Document
D180-18450-2

FEASIBILITY STUDY OF THE BOEING SMALL RESEARCH MODULE

NASA-Ames Research Center
Advanced Space Projects Office

Boeing Aerospace Company
Space Systems Division



FOREWORD

This document is one of a series of four (4), describing the design, manufacture, and management of the Boeing Small Research Module (BSRM). The documents constitute the final report of Contract NAS2-8526, "Feasibility Study of a Small Research Module Concept," performed for the Advanced Space Projects Office of NASA-Ames Research Center.

The objective of the study was to define a low cost standardized spacecraft combining an existing spacecraft design with USAF Space Test Program (STP) management control and NASA aircraft program (ASSESS) instrument integration techniques. The results of the study, presented in this set of documents, includes a preliminary spacecraft design; plans for the management, development, manufacture, test and operation of the spacecraft; and rough order of magnitude (ROM) costs.

The four documents included in this final report are:

D180-18450-1	Executive Summary
D180-18450-2	BSRM Design Document
D180-18450-3	BSRM Program Definition
D180-18450-4	BSRM Program Costs

ACKNOWLEDGEMENTS

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1.0 INTRODUCTION

This document describes the design, capabilities, and subsystem options for the Boeing Small Research Module (BSRM). Specific scientific missions are defined based on NASA/AMES requirements and the BSRM capability to support these missions is discussed. Launch vehicle integration requirements and spacecraft operational features are also presented.

The BSRM design is based on the current S3 satellites being fabricated by Boeing for the USAF STP organization. The extensive use of S3 design features and subsystems insures a flight qualified design for the BSRM. Figure 1.0-1 shows the S3 satellites developed to fly piggy-back on a larger USAF host vehicle. Three spacecraft were fabricated in the S3 program using standardized structure and subsystems for all three vehicles. A total of 17 principal investigators supplied 35 experiments and 69 experimental packages which were successfully integrated demonstrating the feasibility of a modular satellite design concept.

Integration of scientific and experimental payloads from many government agencies and contractors has been accomplished by the BSRM team. Shown below is a list of experiment agencies that have flown in vehicles designed, built and tested by this organization.

- Rice University
- Cubic Corporation
- International Telephone & Telegraph
- Goodyear Aerospace
- TRW Systems
- Radio Corporation of America
- Air Force Cambridge Research Laboratory
- Office of Naval Research
- Advanced Research Projects Agency
- John Hopkins University
- MIT - Lincoln Labs
- MIT - Instrument Lab
- Lockheed Palo Alto Research Lab
- Aerospace Corporation - SPL
- University of California - Berkeley
- Naval Research Laboratory
- Army Ballistic Missile Defense Agency
- Space and Missile Systems Organization

This document is one of a series constituting the Final Report for study contract NAS2-8526, Feasibility Study of a Small Research Module Concept. The total series of documents are as follows:

D180-18450-1	Executive Summary
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Small Secondary Satellites (S3)

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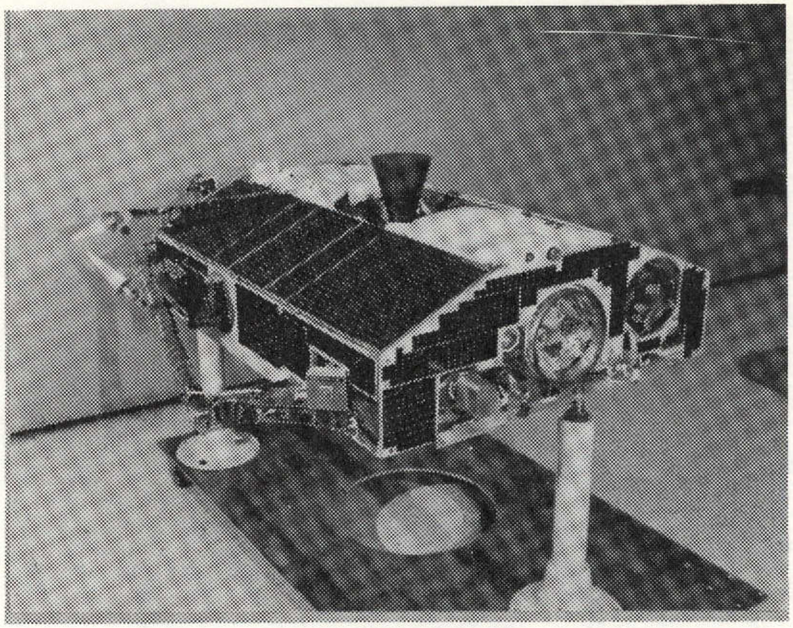

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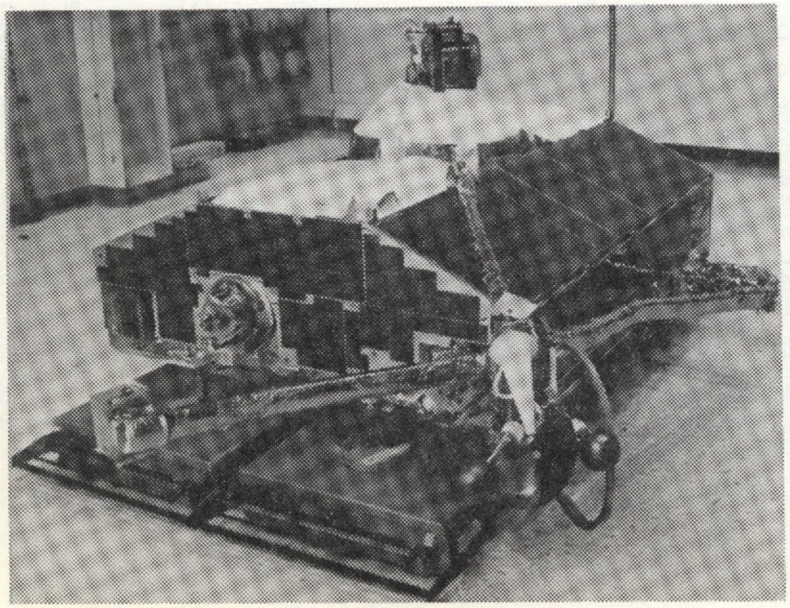

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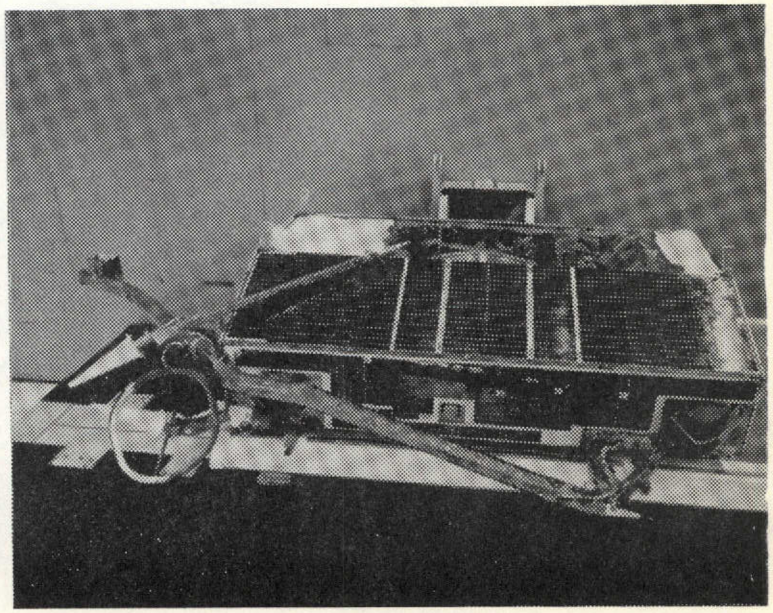

S3–3

Figure 1.0-1. F Small Secondary Satellites (S3)

2.0 CONCLUSIONS AND SUMMARY

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THE BSRM SATELLITE DESIGN PROVIDES A FLEXIBLE, LOW COST, AND RELIABLE SPACECRAFT READILY AVAILABLE FOR A WIDE VARIETY OF EARTH ORBITAL MISSIONS AND EXPERIMENTS. BOTH SPIN-STABILIZED AND 3-AXIS CONTROLLED VERSIONS ARE FEASIBLE USING A BASIC, MODULAR DESIGN SPACECRAFT.

The BSRM design developed in this study offers the following advantages:

- Flight proven subsystems and equipment, qualified for Scout, are used extensively to maximize reliability and reduce cost.
- Considerable mission flexibility is provided by a set of optional kits which accommodate a variety of experiment and mission requirements.
- Each BSRM experiment is located to achieve scientific objectives and to provide thermal control, mass balance, and proper spin inertia ratio within the gross weight limit.
- The software developed by Boeing for the S3 is readily adaptable to a wide variety of missions. The existing programs will be made compatible with NASA ground system computers.

2.1 STUDY CONCLUSIONS

This study contract has demonstrated the feasibility of conducting earth orbital science with a low-cost Small Research Module. A baseline spacecraft design with a small number of subsystem options can readily accomplish a broad spectrum of scientific objectives at minimum cost. Boeing has demonstrated this concept on the successful USAF S3 program and has adapted S3 proven principles to the proposed BSRM program. The extensive use of existing S3 subsystem designs, AGE/GHE, test procedures, orbital operations software, program plans and other documentation ensures an efficient, low-cost BSRM program.

The wide variety of specific missions analyzed in this study do not impose requirements beyond the capability of the baseline BSRM and its defined set of options. Consideration of NASA-Ames program objectives has shown that the low-cost techniques successfully used by Boeing on USAF spacecraft programs for many years can be applied directly to NASA missions. These low-cost techniques have resulted in a demonstrated flight reliability of 96% (22/23 successful launches). Program costs will be comparable to those previously demonstrated if the existing NASA-Ames approach to the BSRM program is maintained.

Consultation during this study with many Principal Investigators has shown their desire for working closely with the spacecraft integrator and BSRM standard interfaces. A key to the success of the Boeing S3 program was the development of close working relationships with experimenters leading to cooperative definition of interfaces and mission requirements. For an efficient BSRM program, early coordination with experimenters and constant "cross-talk" throughout the program is essential. The precise definition, and subsequent freeze, of spacecraft subsystem interfaces and requirements is necessary to permit each PI to complete his experiment design cost-effectively and on schedule. The use of the existing S3 satellite as the BSRM baseline enhances the success of this approach.

2.2 BSRM BASELINE DESIGN

The baseline BSRM spacecraft is a spin-stabilized vehicle, configured to be launched on Scout and using flight-proven S3 satellite subsystems and components extensively. Changes to the S3 design are incorporated where required to provide compatibility with the STDN ground system, Scout booster and specific NASA-Ames mission requirements.

The BSRM structure is formed by longitudinal beams and transverse bulkheads of conventional aluminum construction with extruded and formed chord members and stiffened shear webs. The short, deep beams and bulkheads of the box provide a stiff, weight-efficient structure that meets the critical design objectives of minimum weight, commonality between BSRM configurations, and booster stiffness requirements.

The orientation of the spinning satellite is maintained in inertial space by the angular momentum of the vehicle. Drift caused by environmental torques or orbit regression is corrected by a ground commanded magnetic torquing system. The smooth magnetic torques and a fluid loop damper combine to keep wobble amplitudes to very low values. The spin rate is maintained at precise values by a ground commanded magnetic spin coil. The entire concept has been flown successfully. Ground software reconstructs the spacecraft motion from sun sensor, earth sensor and magnetometer data and correlates the values with experiment data.

Electric power is supplied by solar cells mounted on deployable aluminum honeycomb substrate panels. The cells are 2 by 2 cm, 12-mil, 2-ohm-cm N on P, Boron-doped silicon, with 12-mil fused silica cover glass. Peak and occult power is supplied by a ten ampere-hour 21 cell, sealed nickel-cadmium battery. Power is conditioned by an ampere-hour meter and shunt regulator to control voltage to 28 ± 4 volts and to prevent excessive charge rates or battery temperature.

All communication links have a 6 dB margin above the signal level required for a 10^{-5} bit error rate for all nominal satellite orientations and orbits when communicating with STDN remote tracking stations. Payload and satellite narrow-band analog and digital data are processed at 16 kbps by a PCM processor. Playback data and sidetone turn-around range tones are multiplexed with the real time data, and transmitted on S-band, using an 8.5 watt transmitter. S-band receiver/demodulator, command decoder, and a timer/relay box provide real time and delayed-command capability for the satellite and payloads. Telemetry data are stored at 16 kbps and played back at 229 kbps.

Temperatures of all payloads and components are controlled by multilayer insulation, thermal coatings, and bimetal louvers. The large effective thermal mass of the system is utilized to absorb transient heat loads caused by intermittent operation of the electronic equipment. A passive heat sink is provided under the transmitter to minimize its effects on thermal balance during ground station contacts.

2.3 BSRM DESIGN OPTIONS

The baseline BSRM design can be easily adapted to a wide variety of earth orbital missions with optional kits. Specific options developed in this study include:

- A three axis attitude control system dynamically similar to the spinning satellite. Momentum bias is achieved by incorporating a scanwheel combining the functions of inertia wheel and horizon sensor. The third axis is controlled on-board by torquing the wheel. The wheel spin axis is controlled magnetically on ground command as in the spinning system. Spin rate control is no longer necessary but the second coil performs the analogous function of wheel desaturation.

- Increased power by the simple addition of solar panels to the deployable array. An available DC/DC converter can be readily added to the power subsystem to provide a regulated (± 2%) power bus for experiments if required.

- Higher or lower data rates to meet specific experiment requirements by minor modifications to the on-board processor and substitution of other flight-proven tape recorders.

- Autonomous on-board attitude control and determination by the addition of special purpose control electronics. This option considerably reduces the extent of the ground control effort and ground station access time required to support the operation of the BSRM.

2.4 BSRM INTEGRATION

Boeing has established close working relationships with various government agencies, experimenters and launch vehicle contractors on numerous successful spacecraft programs. This experience will be applied to BSRM to ensure meeting experimenter and payload requirements. Specific integration concepts to be emphasized include:

- Extensive use of Technical Interface Working Group meetings to provide continuous interchange of requirements and data.

- Development and control of Interface Control Documents for each payload and the launch vehicle.

- Thorough documentation and control of system and subsystem diagrams, schematics, specifications and drawings to maintain constant configuration control.

- Coordination of all operational, hardware and software interfaces with the launch facility and ground stations through Program and Orbital Requirements Documents (PRD and ORD).

A complete but efficient test program is proposed for each flight vehicle to ensure thorough integration and operation. The test plan emphasizes all-up system testing to verify subsystem operation and total spacecraft integration with and without experiments. Structural and thermal-vacuum tests for the first BSRM vehicle will be conducted to verify qualification of the configuration revision for the Scout booster. The test program is based on extensive experience gained during other similar spacecraft programs and uses existing AGE, GHE and test software including the Boeing-owned Mobile Test Lab.

BSRM integration with the Scout booster is simple. Boeing has completed similar integration tasks on numerous USAF satellite programs. The mechanical interface with Scout, including fairing dynamic envelopes, is completely defined and compatible with the BSRM. No changes are required to the Scout booster. There is no electrical interface with Scout ensuring low risk, minimum cost launch vehicle integration.

BSRM integration with the Delta and USAF Host Vehicle boosters was also considered in this study. Retention of the existing S3 general arrangement is recommended for these launch vehicles to better utilize fairing envelopes and simplify mechanical interfaces. The BSRM interface with the USAF Host Vehicle is flight proven. Field processing and orbital operations with these boosters is very similar to those developed for the baseline Scout launch vehicle.

2.5 BSRM MISSION APPLICATIONS

A variety of scientific missions were analyzed in this study to demonstrate the adaptability of the BSRM design to specific mission requirements. The baseline design meets the requirements for the NASA-AMES Auroral mission. With the 3-axis stabilization option, the NASA-AMES Aether Drift mission can be accomplished. Consideration of many other mission requirements defined by a variety of experimenters for their AO-7 proposals to NASA Headquarters shows they can be efficiently completed with the BSRM baseline design and its family of options developed in this study.

3.0 BASELINE SPACECRAFT DESCRIPTION

3.0 BASELINE SPACECRAFT DESCRIPTION

THE BOEING SMALL RESEARCH MODULE (BSRM) IS A LOW-COST, HIGHLY RELIABLE SPACECRAFT BASED ON THE FLIGHT PROVEN S3 SATELLITE DESIGN. COMPATIBILITY WITH THE STDN GROUND SYSTEM IS ENSURED THROUGH INCORPORATION OF QUALIFIED COMPONENTS AND THE NECESSARY FREQUENCY CHANGES IN THE S3 EQUIPMENT. THE POWER SYSTEM IS IDENTICAL TO THE S3 AND FEATURES HIGHLY EFFICIENT CONVERSION AND CONTROL.

The extensive use of the S3 satellite subsystems ensures a cost-effective BSRM program. Existing AGE, GHE, test facilities and software, procedures and other documentation are available to support BSRM. The Boeing-owned Mobile Test Lab will be used for automatic test and checkout both inplant and at the launch site minimizing BSRM program costs.

The following paragraphs describe the BSRM baseline in detail. Subsystem options for easily adapting the baseline module to other mission requirements are defined in Section 5 of this report. The options retain the emphasis on flight-proven hardware inherent in the baseline and thus provide great flexibility in accommodating a wide variety of scientific missions with the BSRM spacecraft.

3.1 CONFIGURATION

THE BASELINE BSRM IS CONFIGURED FOR LAUNCH ON THE SCOUT BOOSTER AND MEETS ALL SCOUT INTERFACE REQUIREMENTS. THE SPACECRAFT GENERAL ARRANGEMENT RELIES HEAVILY ON THE EXISTING S3 CONFIGURATION AND FEATURES A MINIMUM OF DEPLOYMENTS, UP TO 10 CUBIC FEET FOR EXPERIMENTS AND CONSIDERABLE FREEDOM IN LOCATING PAYLOADS TO MEET FIELD-OF-VIEW, ORIENTATION AND OTHER SCIENTIFIC REQUIREMENTS.

The baseline BSRM is shown in Figure 3.1-1. The configuration makes effective use of the volume available in the Scout 42" heat shield with space available for a variable payload mix. The closed-box structure is formed by longitudinal beams and transverse bulkheads of conventional aluminum construction with extruded and formed chord members and stiffened shear webs. The short, deep beams and bulkheads of the box provide a stiff, weight-efficient structure that meets the critical design objectives of minimum weight, commonality between BSRM configurations, and a first-mode resonant frequency greater than 100 Hz axially, and 20 Hz laterally.

The primary structural interface with the Scout fourth stage is the V-band clamp and adapter. The V-band is a standard design flown on previous Boeing spacecraft. The adapter will transmit the loads from the BSRM longitudinal shear beams to the Scout interface. No changes to the existing Scout launch vehicle are required to accommodate this.

The equipment, as on S3, remains located to optimize dynamic balance, thermal considerations, inertial requirements, and personnel access. The skin stringer construction of the BSRM provides flexibility for the installation of equipment and experiments. Packages of equipment and experiments may be located at any location within the BSRM and the stringers are sized and located as required to take the load.

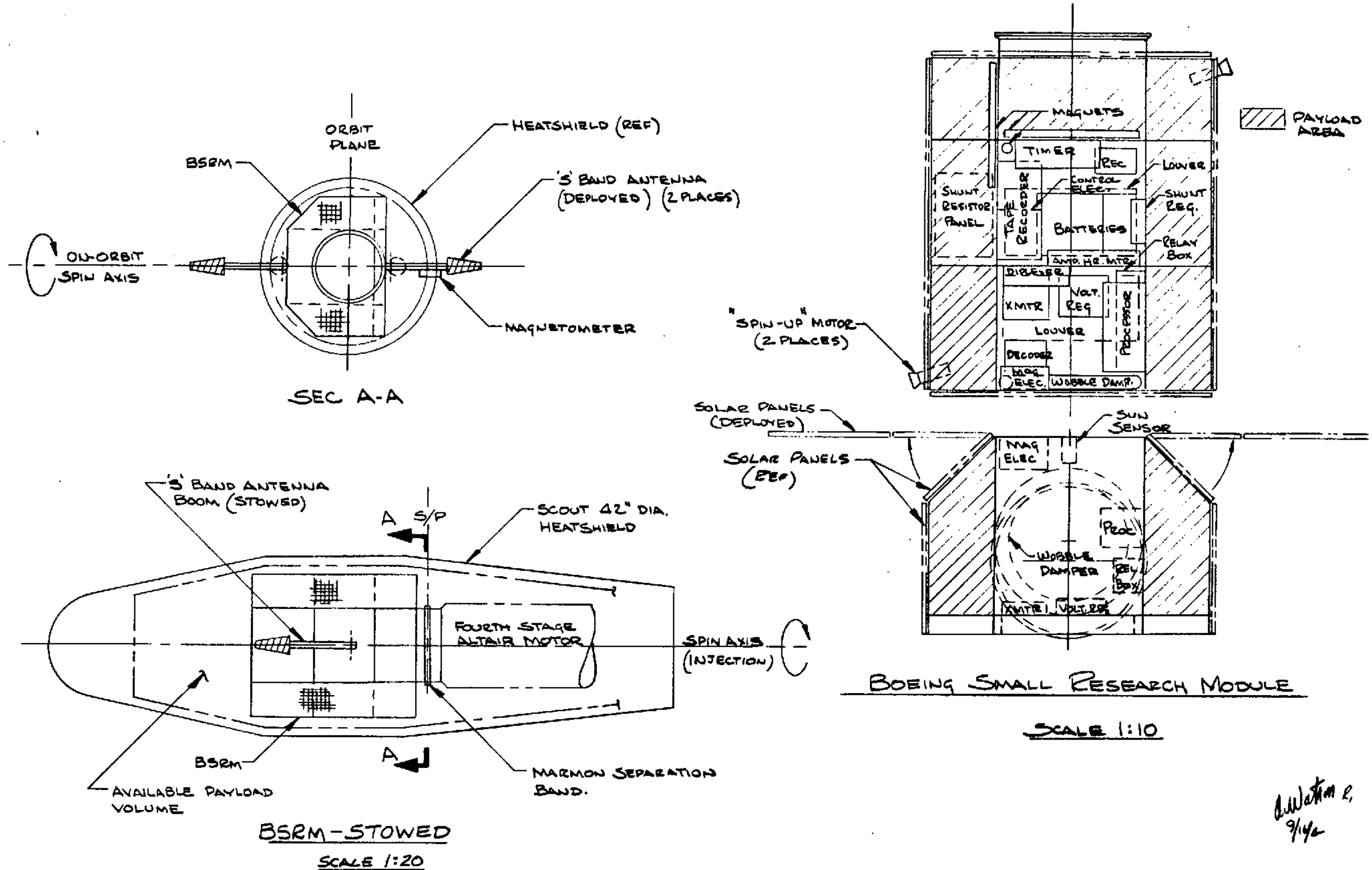


Figure 3.1-1. Baseline BSRM Configuration

The deployable S-band antenna booms were flown on S3. The design consists of a spring loaded hinge with the boom being released by a pin puller.

The solar arrays are similar to those flown on S3 except the BSRM will deploy the side and top panels. The design utilized in deploying the panels is the same used to deploy the STP P72-1 double-fold antenna, except small leaf springs are used in the hinge joints because of the small deployment angle. (See Figure 3.1-2). One flight proven pin puller releases each side of the deployable panels.

The structural arrangement of the vehicle allows grouping payloads with similar temperature requirements within the same compartment and controlling these compartments by balancing heat-in to heat-out. Sensitive components are protected with multilayer insulation and thermally coupled to the satellite interior where required. Multilayer insulation and small heater elements maintain required temperatures on externally mounted payloads.

Satellite payload and subsystem alignment requirements are satisfied by controlling equipment installation relative to satellite geometric axis and by controlling spin axis relative to geometric axis by dynamic balancing. The satellite geometric axis is established by tooling to hold booster interface structure within tolerances needed for mating. Readily achieved installation alignments leave ample margin for thermal and repeatability of deployment system variations. Thermal distortions are small with the spinning satellite; deployment repeatability is closely controlled to less than 0.5 degree for the booms. Payloads are aligned with the spin axis to $\pm 0.5^{\circ}$. Tighter tolerances can be achieved when required.

The BSRM weights are summarized in Table 3.1-1. A contingency is provided for normal weight growth. The weights shown for the subsystems represent almost all actuals since the equipment is available and flight proven. The large proportion of actuals at this stage of the configuration development provides high confidence in the satellite weight and performance.

The BSRM subsystems are discussed in detail in subsequent sections of this document. Flight proven S3 components are used extensively. Where BSRM mission requirements cannot be met with S3 equipment, other flight proven items are selected. Table 3.1-2 summarizes the equipment for the major baseline BSRM subsystems and indicates the flight history of each component. Section 3.9 discusses the environmental qualification status of these items for the Scout launch vehicle.

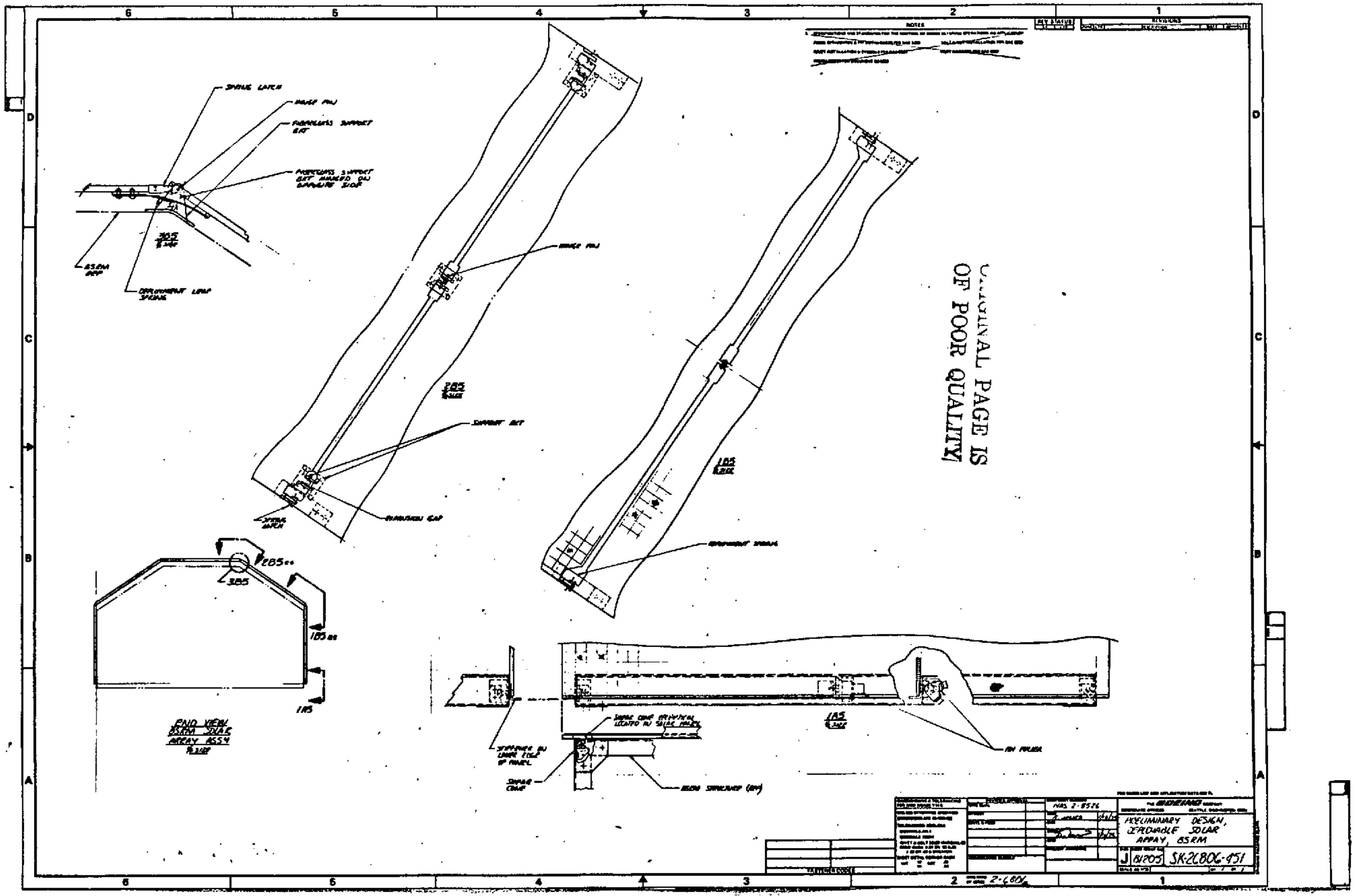


Figure 3.1-2. BSRM Solar Array Deployment Concept

Table 3.1-1. BSRM General Weight Statement

	LBS	KG
POWER SUBSYSTEM	41.30	18.77
*ATTITUDE & CONTROL DETERMINATION SUBSYSTEM	33.70	15.32
TT&C SUBSYSTEM	40.50	18.41
STRUCTURE	51.00	23.18
WIRING	19.00	8.64
THERMAL (LOUVERS, PAINT, BLANKETS, ETC.)	16.00	7.27
	201.50	91.59
GROWTH ALLOWANCE (15%) ON STRUCTURE, WIRING & THERMAL	12.90	5.87
TOTAL WEIGHT + GROWTH ALLOWANCE =	214.40	97.46

*ADD 18.3 LBS (8.31 KG) FOR THREE-AXIS SPACECRAFT

Table 3.1-2. Baseline BSRM Equipment List

	QTY.	SUPPLIER	FLIGHT HISTORY S3	FLIGHT HISTORY OTHER
POWER SUBSYSTEM				
VOLTAGE LIMITER	1	GULTON	X	
SHUNT RESISTOR PANEL ASSY	1	BOEING	X	
BATTERIES	3	EAGLE PICHER	X	
AMP. HOUR METER	1	GULTON	X	SKYLAB
SOLAR PANELS	5	BOEING/SPECTROLAB	X	
ATTITUDE CONTROL & DETERMINATION SUBSYSTEM				
ELECTROMAGNETS	3	BOEING	X	
TIMER/SEQUENCER	1	CELESCO	X	BURNER II
MAGNETOMETER SENSOR	1	SCHONSTEDT	X	
MAGNETOMETER ELECTRONICS	1	SCHONSTEDT	X	
WOBBLE DAMPER	1	BOEING	X	
SUN SENSOR (SINGLE AXIS)	1	ADCOLE	X	
EARTH SENSOR	1	BARNES	X	AEROS
TT&C SUBSYSTEM				
S-BAND ANTENNA	2	BOEING	X	70-1, P72-1
TAPE RECORDER	1	ODETICS	X	P72-1
TRANSMITTER	1	CONIC	X	P72-1, LCRU
PROCESSOR	1	TELEDYNE	X	
COMMAND DECODER	1	CONIC	X	
DIPLEXER	1	WAVECOM	X	USAF
RECEIVER	1	CINN. ELEC.		USAF
RELAY BOX	1	BOEING	X	

3.2 ELECTRICAL POWER

THE BSRM ELECTRICAL POWER SYSTEM EMPLOYS THE SAME SYSTEM AND HARDWARE TO PRODUCE, CONTROL AND STORE ELECTRICAL ENERGY AS THE FLIGHT-PROVEN S3 SATELLITE. IT DIFFERS ONLY IN THE PHYSICAL SIZE OF THE SOLAR PANELS AND THE SOLAR ARRAY DEPLOYMENT MECHANISM.

Spacecraft electrical power is provided by a true direct energy transfer (DET) system as shown in Figure 3.2-1. Power from the solar array is transferred directly to the spacecraft bus which is controlled to 28 $\pm$ 4 VDC. There are no series voltage regulating elements between the solar array and the spacecraft bus, between the solar array and battery, nor between the battery and the bus.

3.2.1 SOLAR ARRAY

The solar array consists of deployable panels, each covered with 12 mil thick, 2 cm by 2 cm solar cells with a base resistivity of 2 OHM-CM. The solar cells are protected by 12 mil thick quartz cover slides. The solar array output power varies with the panel temperature, the battery charge condition, system load demand, and orbital parameters. In a sun-pointed mode, the solar array generates 168W at the end of 6 months. The amount of power available to the payload depends on the operational requirements of the experiments. In the sun-pointed mode for a typical noon-midnight orbit the array is capable of supporting 45 watts of continuous experiment power.

3.2.2 BATTERY

During occulted periods and whenever system load exceed the solar array capabilities, the spacecraft loads are supported with electrical energy from a nickel-cadmium (Ni/Cd) battery. The battery is a 21 cell, ten ampere-hour sealed Ni/Cd type manufactured by Eagle-Picher. This battery is packaged in three identical assemblies of seven hermetically sealed cells per pack. Positive low impedance thermal paths are provided by individual finds on each cell employing thru bolts at the deck and thermal louver interface. Extensive testing during the S3 program has clearly verified the performance of this battery.

3.2.3 AMPERE-HOUR METER

Battery charge/discharge history is monitored by an ampere hour meter to automatically control battery charge levels and provide safeguards against excessively deep depths of discharge. The ampere hour meter provides a command to the voltage limiter to limit the system bus voltage level and thus the battery charge rate when the proper amount of recharge current is provided to the battery. In addition, if the battery state-of-charge inadvertently reached 30 percent, a signal is provided to the system relay box to inhibit non-essential loads until a battery state-of-charge level of 50 percent is reached. Charge/discharge current is measured by an internal current shunt to quantize time and current for control of the voltage limiter and to provide telemetry data for performance evaluation on the ground.

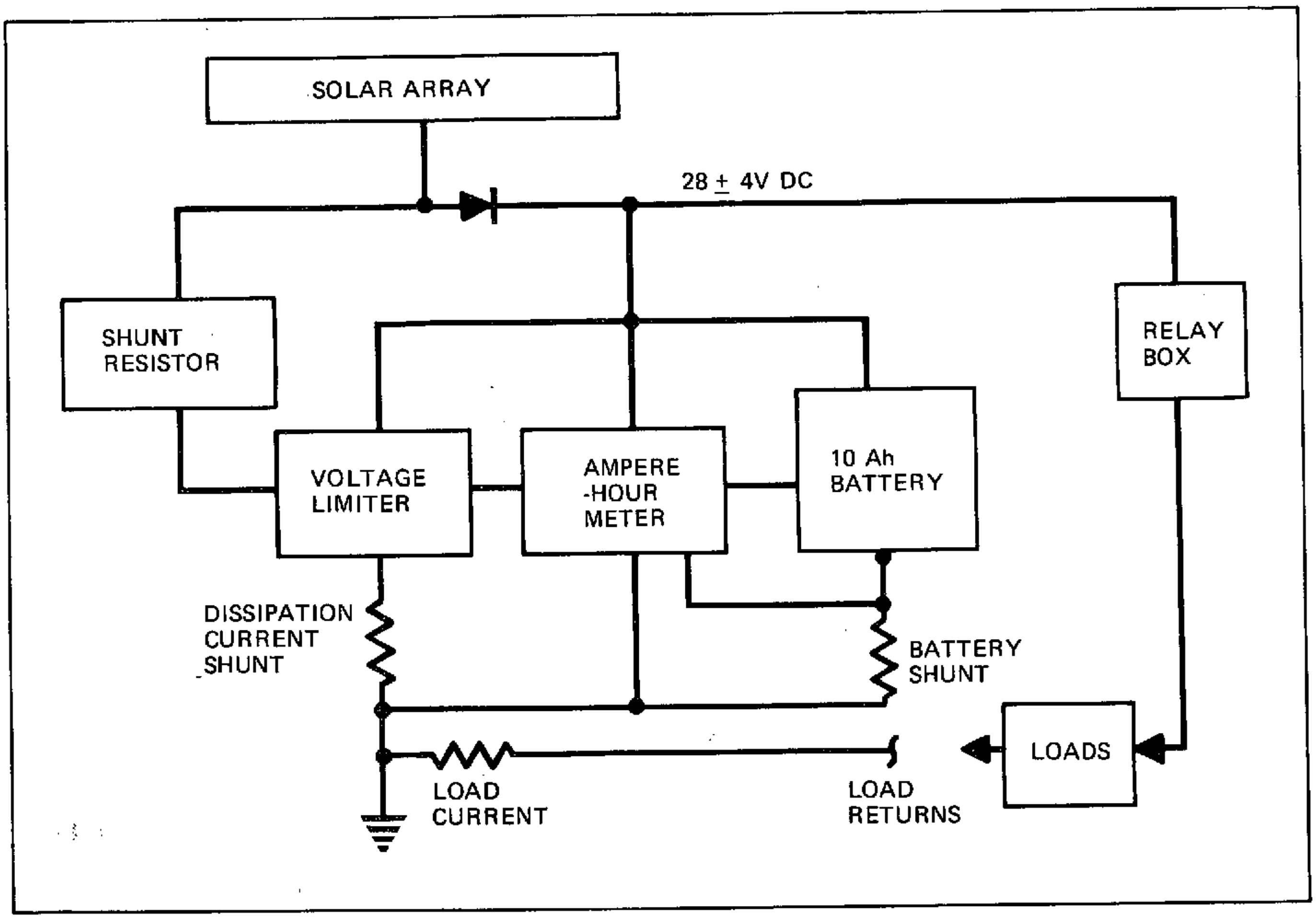


Figure 3.2-1. BSRM Electrical Power Subsystem

3.2.4 VOLTAGE LIMITER

Solar array voltage applied to the power system bus is controlled by shunting excess current through load resistors using the same equipment as in the basic S3 satellites. The voltage limiter provides a two-level control and switches the operation between the high voltage level (used for full battery charge) and the low voltage level (used for battery trickle charge). The low level voltage is selectable to seven levels in 0.5 volt steps to provide optimum operation at the battery temperature realized in space operation. Any one of these seven levels may be selected on the ground after observing telemetry data on battery temperature and is inserted by commanding the desired level through the TT&C command uplink. The voltage limiter contains a current shunt for measuring the amount of flow through the load resistors, providing telemetry readout on the ground for power system monitoring.

3.2.5 SYSTEM DESIGN

The selection of proper components and system design provides a simple, versatile and highly reliable electrical power source. The following features provide additional reliability and trouble-free operation:

- Single point ground to structure.
- All conducting surfaces are insulated with reference to structure ground point.
- Structure is non-current carrying.
- Wiring is designed to minimize EMI and magnetic fields.
- Non-magnetic materials are used throughout.

3.2.6 HOUSEKEEPING POWER REQUIREMENTS

The basic housekeeping functions are tabulated in Table 3.2-1 for three modes of operation: a minimum power standby mode; a record mode when data are put on magnetic tape; and a transmit mode where data are relayed to the ground. The transmit mode lasts typically about 7.5 minutes and is assumed to occur each orbit.

3.2.7 S3 MODIFICATIONS

The baseline power system of the BSRM has been modified from the flight-proven S3 satellite in two areas. The area of the solar panels have been altered to meet the shroud constraints of the Scout launch vehicle. The components and design remain the same.

Because of power requirements for various payloads it was determined that a deployable solar array should be used as a baseline, rather than the body-mounted system used on the S3 satellite. This modification is discussed in Section 3.1, Configuration.

Table 3.2-1. BSRM Housekeeping Power Requirements

Subsystem	Standby mode	Record mode	Transmit mode
Command	(1.86W)	(1.86W)	(1.86W)
• Command receiver/demodulator	0.92	0.92	0.92
• Command decoder	0.10	0.10	5.5
• Timer/sequencer	0.84	0.84	0.84
Telemetry	(5.0)	(16.1)	(79.4)
• S-Band transmitter	---	---	57.2
• PCM encoder/processor	5.0	10.5	12.6
• Tape recorder	---	5.6	9.6
Attitude control	(5.48)	(5.48)	(5.48)
• Magnetometer	0.98	0.98	0.98
• Electromagnets (3)	2.0	2.0	2.0
• Edge detector	2.0	2.0	2.0
• Sun sensor	0.5	0.5	0.5
Electric power	(2.15)	(2.15)	(2.15)
• Ampere-hour meter	2.7	1.7	1.7
• Voltage regulator	0.45	0.45	0.45
	14.49W	25.59W	88.89W

3.3 ATTITUDE CONTROL AND DETERMINATION

THE ATTITUDE CONTROL AND DETERMINATION (AC&D) SYSTEM USES TECHNIQUES EMPLOYED ON THE AIR FORCE STP 72-1 AND S3 SERIES OF SATELLITES. THE CONCEPT USES AN ANGULAR MOMENTUM BIAS DUE TO THE SPINNING SPACECRAFT TO MAINTAIN AN INERTIALLY FIXED ATTITUDE IN SPACE.

The attitude control and determination system provides the following capabilities:

- A flight-proven attitude control and detection system is used which is common to all satellite missions.
- Low duty cycle at the control facility permits periodic updates to be made infrequently. Even in low and eccentric orbits, ground commands need be updated no more often than once per day.
- Maneuver capability of 2^{0} per hour (at 5 rpm) available in low earth orbits.
- Solar and earth sensor data may be used in combination for single-point determination of payload attitude with 2σ accuracy of less than 1.0 degrees during favorable positions of earth and sun.
- Off line attitude determination to within 0.5 degrees (2σ) from recorded attitude data.
- Spin rate control capability of 0.5 rpm per hour in low earth orbits.

The angular momentum will drift under the action of environmental torques. Errors are determined by telemetering sensor data to the ground where the spacecraft motion is reconstructed by the Small Satellite Attitude Control (SSAC) software program. Corrections are applied, on ground command, by modulating the current to a single electromagnet aligned with the momentum vector. The control is thus closed loop with loop closed on the ground. This approach means slow response but provides considerable flexibility by permitting a choice to be made between various control and maneuver options. The slow response is not a liability since the stiffness imparted by the angular momentum ensures that the vehicle response will also be slow. An advantage of the method is the capability of discerning average and long term torque effects so that a torquing schedule can be devised both to correct current errors and compensate for expected future torques. Experience has shown that torque command updates may only be required as infrequently as once every two weeks. The frequency will depend on the torque environment and accuracy required, however, and will vary from mission to mission.

3.3.1 SYSTEM DESCRIPTION

The basic system components consist of three electromagnets with associated electronics for switching and setting current levels, earth, sun and magnetic field sensors and a passive fluid loop wobble damper. The system components and information flow are shown schematically in Figure 3.3-1. Magnetometer

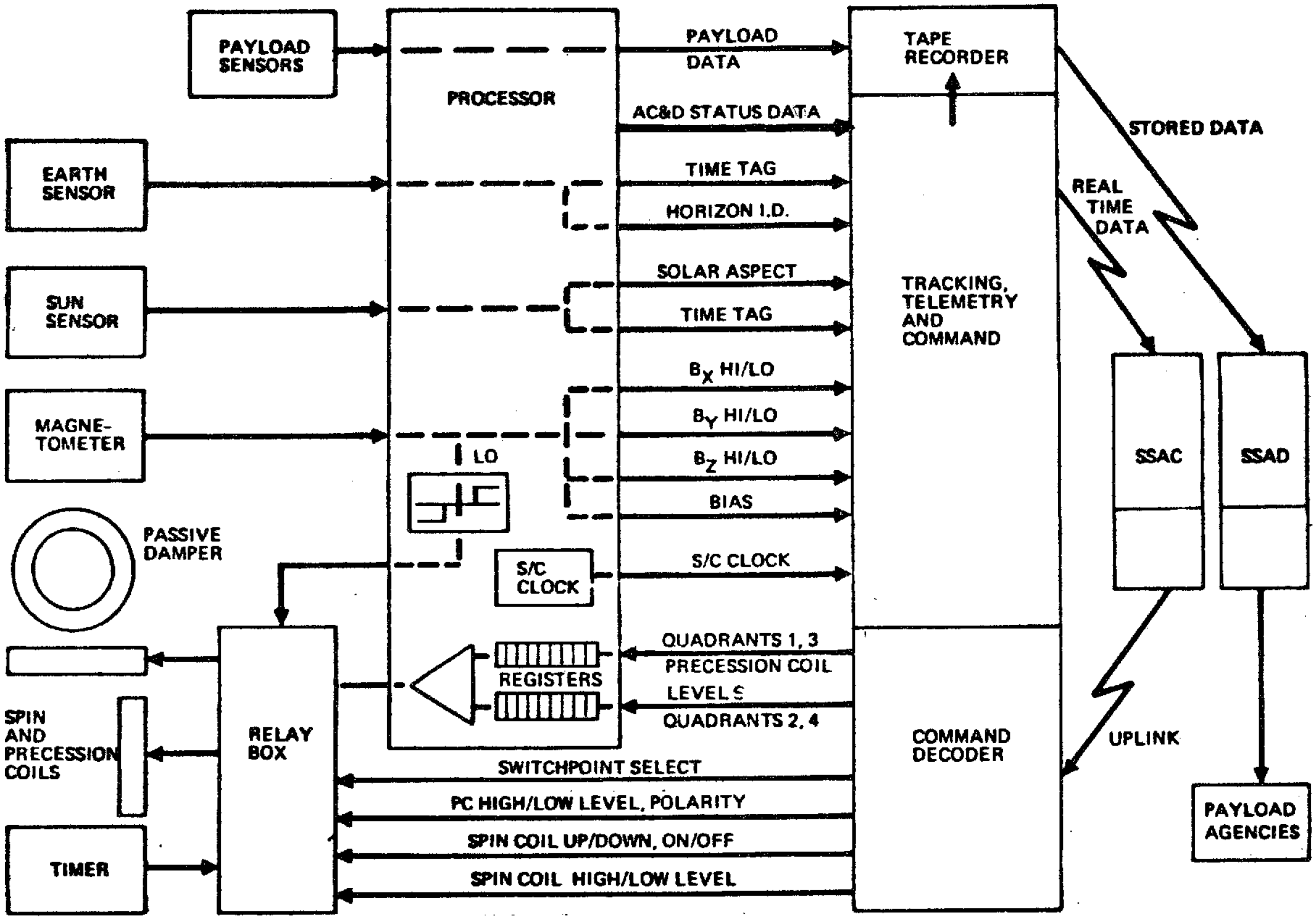


Figure 3.3.1. Attitude Control Subsystem Flow Diagram

and earth and sun sensor data are conditioned in the processor then passed to the Tracking, Telemetry and Command subsystem which transmits the information to ground in real time during station contacts. On the ground the SSAC software is employed to determine errors and generate corrective commands. Commands are sent to the spacecraft at a convenient pass and they are decoded and implemented on board.

Analysis shows that a net rotation in any direction can be achieved by the combination of a constant electromagnet coil current and a current which changes polarity at crossings of the equator and closest approach to the poles. The two current values are stored on board in two registers. Switching is accomplished by the combination of a timer or sequencer and a relay box, Figure 3.3-2.

The precession coil which controls the spin axis orientation is mounted in the spacecraft parallel to the momentum vector. A second coil, lying in the plane of the spin, is used to control spin rate. To provide net spin up and spin down torques the coil current is commutated using magnetometer signals to control the timing.

Attitude determination is accomplished in non real time using tape recorded attitude data which is periodically dumped. The ground software Small Satellite Attitude Determination (SSAD) program takes the data, reconstructs spacecraft motion and correlates the spacecraft attitude with experiment data. The information is then passed on to the payload agencies. Both the SSAC and SSAD programs do attitude determination. The purpose of SSAC is to provide the attitude for monitoring purposes and to generate corrective commands. The SSAD program operates on longer data spans and can thus reconstruct the motion and determine attitude with better accuracy. Normally SSAD outputs are not needed immediately and are run when convenient off line.

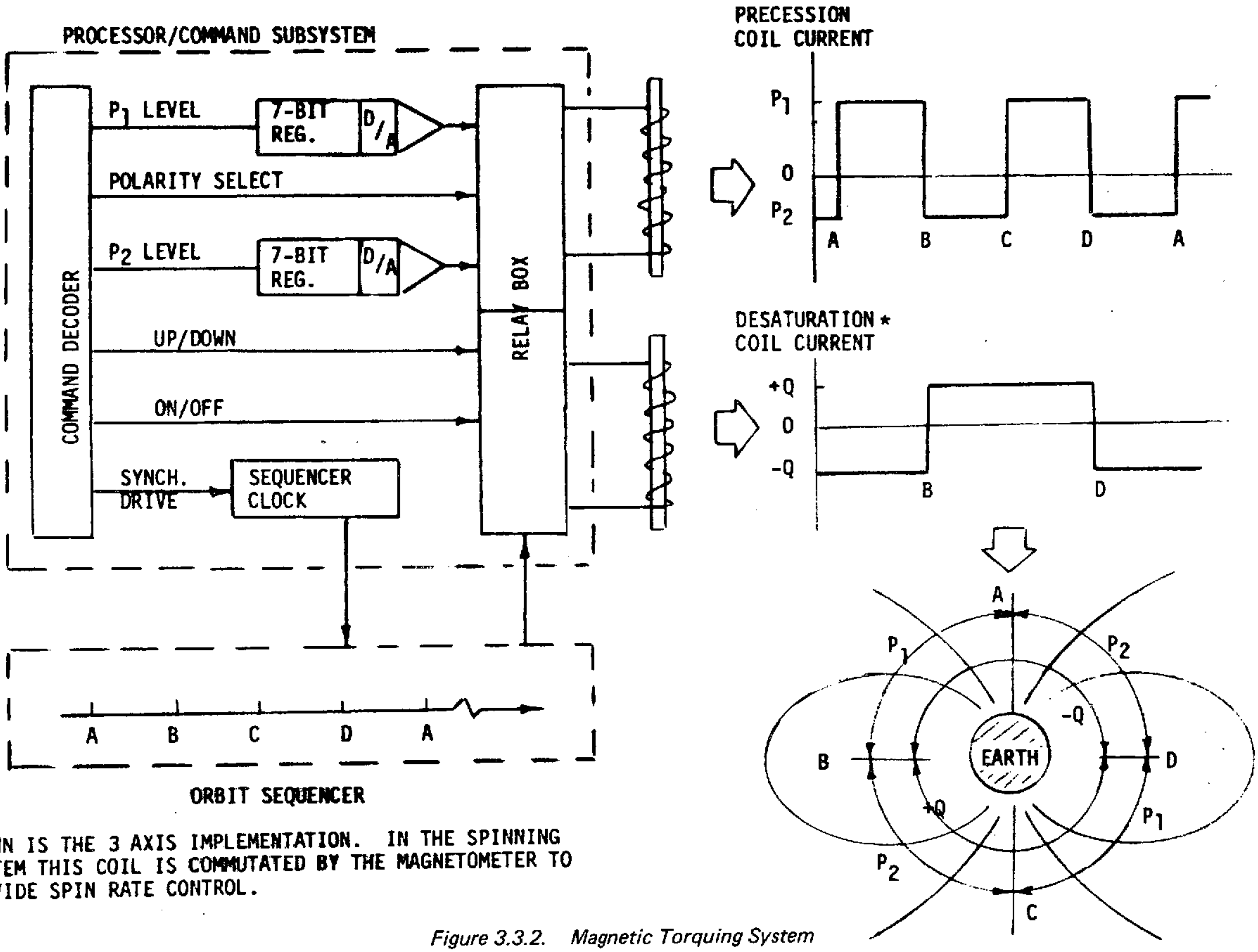
3.3.2 OPERATING MODES

Four modes of operation are possible with the spinning BSRM. These are the orbit normal, sun line, sun line normal, and inertial spin axis orientation modes.

Orbit Normal Mode - The vehicle spin axis is maintained perpendicular to the orbit plane. The primary attitude reference is the combination of earth sensor and sun sensor. Backup capability is provided by the three axis magnetometer which can be used in conjunction with either the earth or the sun sensor. Continuous earth sensor information is available and sun sensor data is unavailable only during occultation.

Sun Line Mode - A two-axis sun sensor with its axis along the spin axis is used to keep the vehicle spinning about the sun line. This provides a good reference about two axes. The earth sensor will see the earth twice per orbit to provide a periodic correlation of position around the spin cycle.

Sun Line Normal Mode - The spin axis is maintained normal to the sun line and in the orbit plane in this mode. The sun sensor is oriented perpendicular to the spin axis, sees the sun once per revolution and provides one reference. The second is obtained twice per orbit from earth sensor data which will also provide information on position in the spin cycle.



*SHOWN IS THE 3 AXIS IMPLEMENTATION. IN THE SPINNING SYSTEM THIS COIL IS COMMUTATED BY THE MAGNETOMETER TO PROVIDE SPIN RATE CONTROL.

Figure 3.3.2. Magnetic Torquing System

Inertial Pointing Mode - Theoretically the spin axis can be oriented in any direction in inertial space. Practically, there are a number of constraints including power and thermal control which will restrict the range of possible orientations. From the control and determination point of view a basic difficulty is the lack of adequate references. This complicates both acquiring and holding the desired attitude. In most applications the earth and sun sensors can be positioned to provide periodic data; thus attitude corrections can be made if sufficient time is available. This time will be of the order of days.

3.3.3 PERFORMANCE

Performance capabilities depend so heavily on the orbit parameters, spacecraft configuration and maneuver requirements that only broad generalizations can be made. The values in Table 3.3-1 represent near ideal conditions of a moderately low (~200 nm.) circular orbit at high inclination with no maneuver. In general, accuracies tend to be better in circular than elliptical orbits. An increase in altitude decreases control authority because the earth's field is weaker, but in general control and determination are not degraded.

Table 3.3-1. Estimated BSRM Performance Capabilities

System/mode	Attitude		Spin rate	
	Hold	Determination	Hold	Determination
Spinning System				
• Orbit normal mode	±1.0 deg	±0.25 deg	±0.1 rpm	±0.01 rpm
• Sun line mode	±1.0 deg	±0.5 deg	±0.1 rpm	±0.01 rpm
• Sun line normal mode	±2.0 deg	±0.5 deg	±0.1 rpm	±0.01 rpm
• Inertial pointing mode	±3.0 deg	±1.0 deg	±0.1 rpm	±0.05 rpm

3.3.4 S3 MODIFICATIONS

The baseline BSRM attitude control and determination subsystem differs from the flight-proven S3 system in the following areas. (All three modifications are considered to have minimal impact on the system.)

- A third electromagnet is required for the BSRM because the satellite leaves the launch vehicle spinning in the wrong orientation. It is necessary to have the third electromagnet to determine and assist in spacecraft reorientation.
- Acquisition logic is required to assure proper spacecraft orientation.
- Modification of the AC&D software is required to insure compatibility with STDN and NASA systems.

3.4 TELEMETRY TRACKING AND COMMAND

THE BASELINE TT&C SUBSYSTEM PROVIDES A 16 KBPS REAL TIME DATA SYSTEM WITH 210 MINUTES OF STORAGE PLUS COMMAND AND TURN-AROUND RANGING USING COMPONENTS FROM S3 SPACECRAFT.

3.4.1 BASELINE SYSTEM

The subsystem shown in Figure 3.4-1 meets all BSRM telemetry, tracking and command (TT&C) requirements using space proven components. Payload and satellite narrowband analog and digital data are processed at 16 kbps by a PCM processor. Recorded playback data and turn-around ranging are then multiplexed with the real time data at baseband, and transmitted on S-band using an 8.5 watt transmitter. An S-band receiver/demodulator, command decoder, relay box and timer provide real time and delayed command capability for the satellite and payloads. Telemetry data is stored at 16 kbps and played back 14:1 at 229 kbps using the tape recorder. Two conical log spiral antennas and a diplexer permit cuplex operation of the uplink and downlink.

Communication with NASA STDN ground stations is accomplished through the S-band up and down link. The 8.5 watt transmitter has a maximum slant range during high data rate playback of 7000 km with a 30 foot antenna. A bit error rate of 10^{-5} is maintained with 6 dB or greater signal to noise margin for both command and telemetry links. Real time 16 kbps data (formatted spacecraft subsystem status and payload digital and narrowband analog data) are phase-shift-keyed (PSK) on a 1.024 MHz subcarrier. The recorded data is transmitted at 229 kbps by PSK modulating a 1.7 MHz subcarrier during playback. The subcarriers are multiplexed with the turn-around ranging sidetones and transmitted in the 2200-2300 MHz band.

3.4.2 COMMAND RECEPTION

Two Boeing conical log spiral antennas each having hemispherical coverage are switched to assure ground tracking coverage. Mounted on the spacecraft spin axis and pointing in diametrically opposite directions, their crossover point is at ground station zenith. After each pass, the switch is set to accommodate the next station. Figure 3.4-2 depicts the antenna configuration. Simultaneous transmit and receive signals are filtered in the S-band diplexer. The receiver is continuously powered and assures command access at all times. Both command and ranging signals are STDN compatible. The original S3 receiver and demodulator are replaced by comparable STDN compatible units using an FM detector to recover the PCM/AM/PM signal. Commands are frequency shift keyed (FSK) at 1 kbps using tones in the 7-12 KHz range and amplitude modulated at 50% by a sinewave clock. The command demodulator produces digital data and clocking for the decoder. Ranging tones of 4 KHz, 20 KHz, 100 KHz or 500 KHz are sent to the baseband unit for combining with the data subcarriers and retransmitted to provide a noncoherent ranging turn-around.

Uplink calculations using a STDN 1 kw ground station and the 30 ft. parabolic antenna show greater than 6 dB signal margin out to 7000 km slant range (orbital altitude of 3400 km at 5^{o} elevation) which exceeds all expected mission requirements. Figure 3.4-3 summarizes the R.F. uplink.

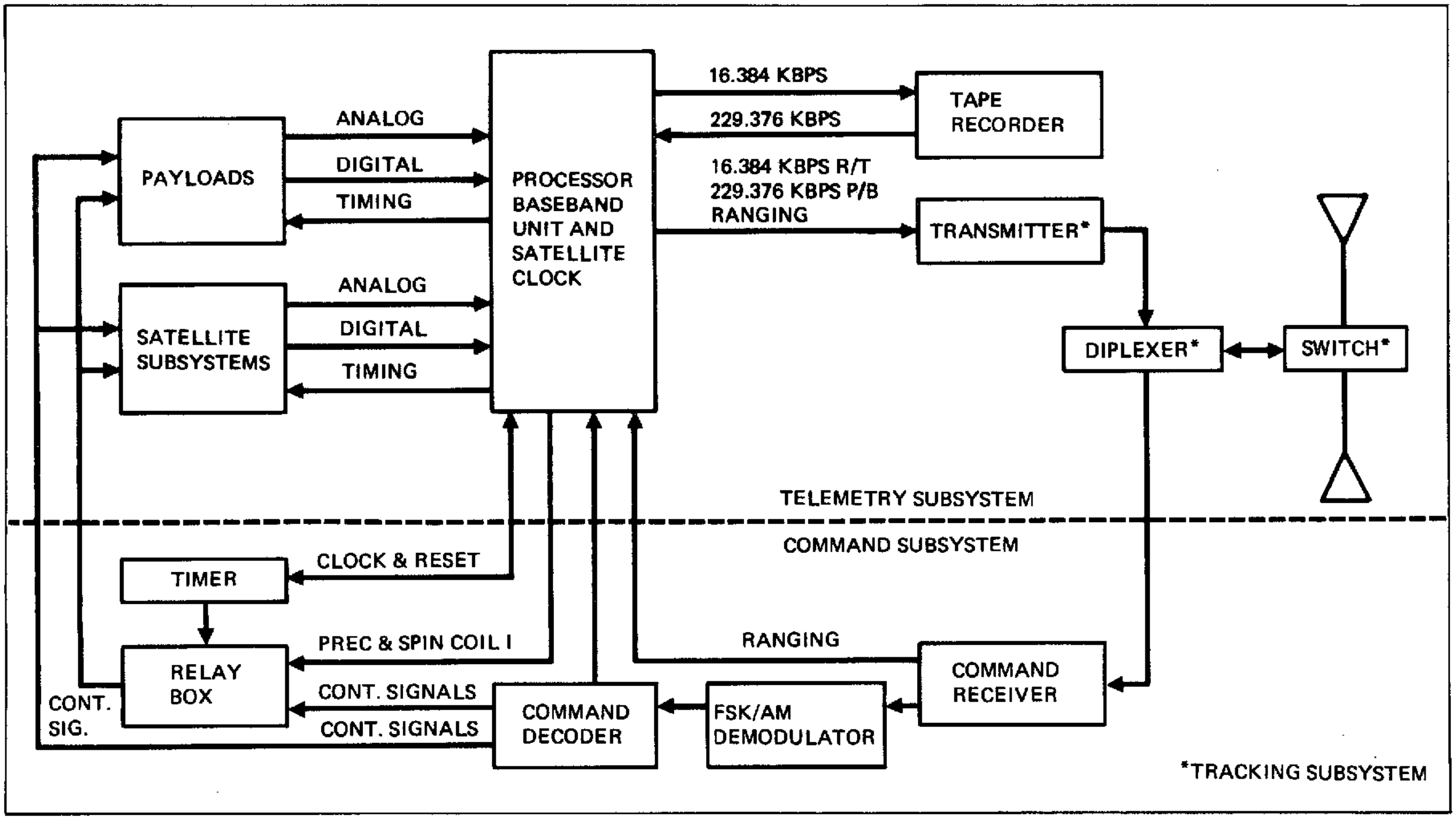


Figure 3.4-1. Baseline BSRM Telemetry, Tracking and Command Subsystem

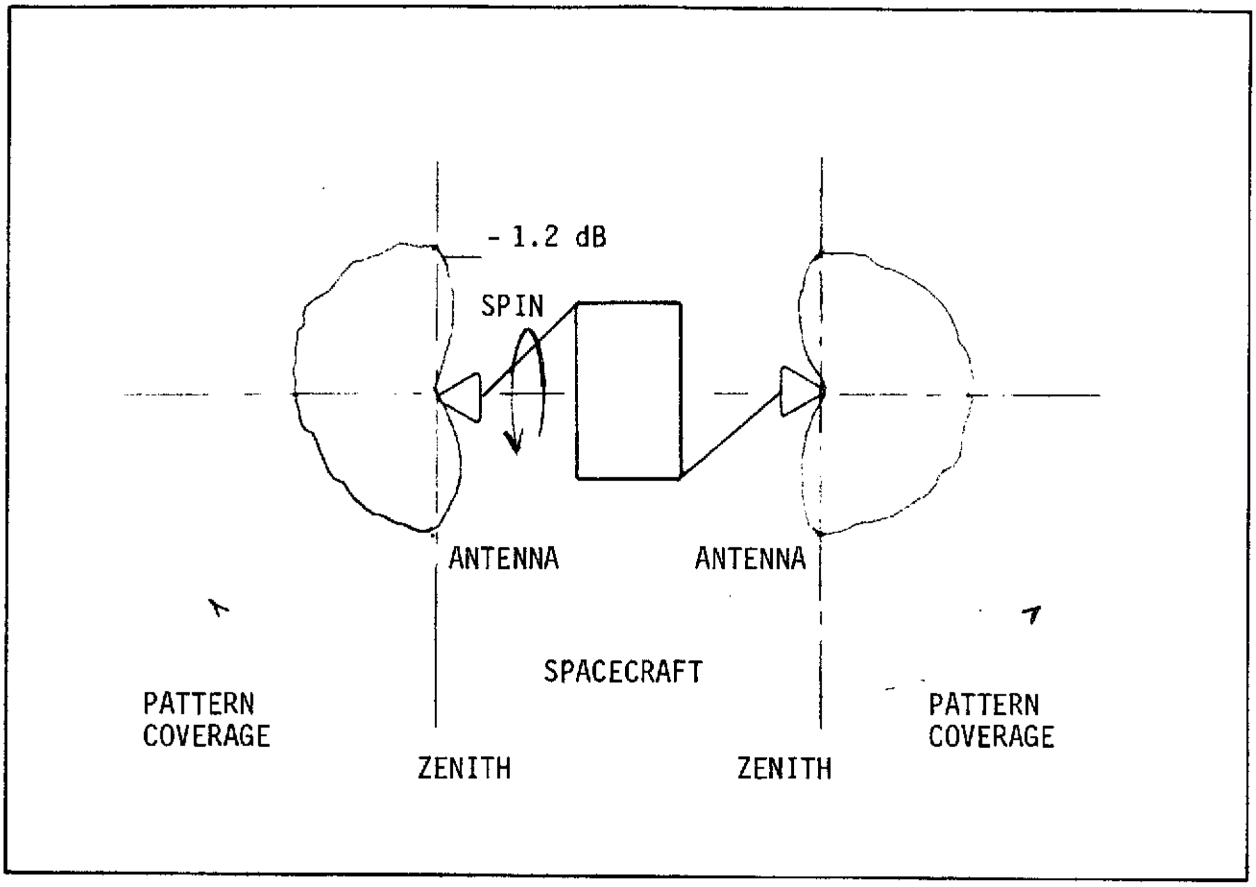


Figure 3.4-2: BSRM Baseline Antenna Configuration

	PARAMETER	WORST CASE VALUE	SOURCE
1.	XMTR POWER	+ 60 dBM	1 KW STDN 101.1
2.	XMTR. ANT. GAIN	+ 43 dB	USB 30' ANT. STDN 101.1
3.	SPACE LOSS (7000 KM S.R.)	- 175.7 dB	@ 2100 MHz
4.	POLARIZATION LOSS	0 dB	BOTH RHCP ANT.'S
5.	ATMOS. LOSS	.6 dB	@ 5^{o} ELEVATION
6.	S/C ANTENNA GAIN	- 1.2 dB	S3 SATELLITES
7.	S/C CIRCUIT LOSS	2.5 dB	S3 SATELLITES
8.	TOTAL RECEIVED POWER	+ 77.0 dB	SUM 1 - 7
9.	S/C RCVR. NOISE DENSITY	-167 dBm-Hz	CALCULATED N.F. 7 dB
		COMMAND PERFORMANCE	
10.	MODULATION LOSS	- 4.9 dB	M.I.=1.0 R.
11.	RECEIVED POWER	- 81.9	8 LESS 10
12.	RECEIVER SENSITIVITY	- 100 dBM	FOR 10^{-6} BER
13.	PERFORMANCE MARGIN	+ 18.1 dB	11 LESS 12
		RANGING PERFORMANCE	
14.	MODULATION LOSS	- 10.1 dB	M.I.=0.6 R (SINGLE TONE)
15.	RECEIVED POWER	- 87.1 dB	8 LESS 14
16.	NOISE B.W. (1 Hz)	0 dB	GSFC
17.	REQUIRED SNR	35 dB	GSFC
18.	THRESHOLD POWER	-132 dB	9 + 16 + 17
19.	PERFORMANCE MARGIN	+ 44.9 dB	15 LESS 18

Figure 3.4-3: BSRM Command R.F. Uplink Summary

3.4.3 COMMAND CONTROL

3.4.3.1 Decoding. The S3 command decoder is utilized with slight logic changes to accommodate the STDN preamble and vehicle address length. The overall word is maintained essentially the same (20 bits) and no requal is necessary.

The command word format (Figure 3.4-4) is depicted as a 20 bit command word. Each satellite has separate decoder address to identify each vehicle. A magnitude designator code of all zeros (000) specifies one of 128 possible discrete output control pulses (28 v.d.c @ 125 MS) from the command decoder. A current capability of 300 ma drives multiple control relays in the relay box. Magnitude words are designated by non zeros in the magnitude I.D. code. Data is contained in the 8 data bits (11 - 18) designated discrete command or magnitude word. Magnitude data is transferred serially to the PCM processor for processing. Each received command is transferred to the processor as a command word replica (bits 8-18) along with Accept and Reject bits. The PCM processor then transmits this to the ground (13 bits) to perform a command verification. If an accept is received, no further action need be taken. If a reject is received, the command is retransmitted until accepted. The command word replica may be investigated for transmission errors. The command decoder output capability is shown in Figure 3.4-5.

A spare bit in the word format (filler bit 19) could be used to expand the MAG. I.D. (4 bits vs. 3) if up to 16 total digital command magnitude words were necessary. Likewise, the 8 bit data word could be expanded to 9 bits if necessary without significant impact. This has not been considered necessary in any of the presented missions, but represents spare capacity.

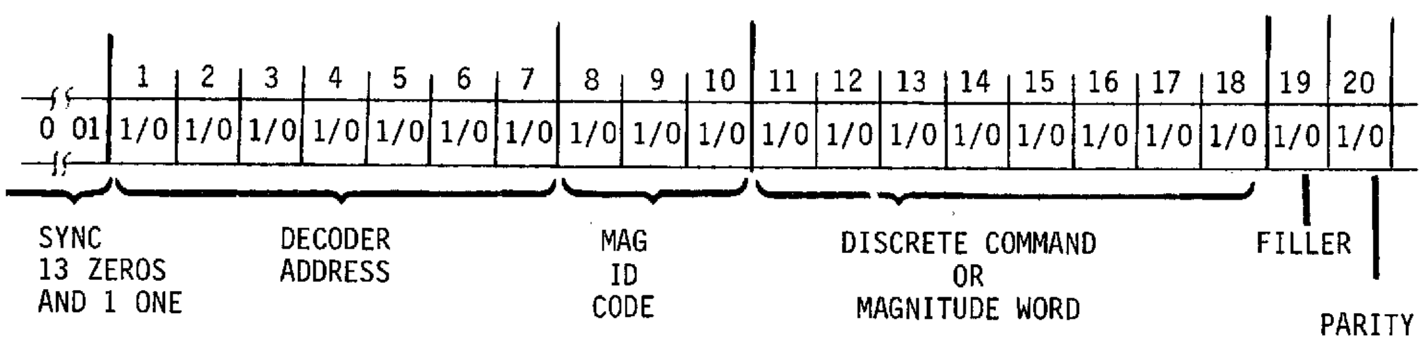


Figure 3.4-4: BSRM Command Word Format

	BSRM S/C	EXPERIMENT (OR SPARE)	TOTAL
DISCRETE COMMAND PULSES	60	68	128
DIGITAL COMMAND MAGNITUDE WORDS	4	3	7*

*See text on filler bit

Figure 3.4-5: BSRM Command Decoder Allocations

3.4.3.2 Control. Spacecraft and experiment control is depicted in Figure 3.4-6. Primarily, the majority of controlling functions are performed through the relay box and timer as shown in the figure. The relay box provides latching functions and switches high power loads. The timer provides delayed control outputs for mission sequencing of timed payload and spacecraft functions. The timer cycles, the speed of which is controlled by an uplinked magnitude time-word, are determined by mission data acquisition requirements. The timer is set to recycle once per orbit. Timed onboard commands are under ground control via setting up of the orbital cycling relay matrix. Up to six orbits can be set up in advance. Of the 23 commands 16 are allocated for spacecraft attitude control with 8 available for experiment ON-OFF functions. Coarse times are set during manufacture with fine adjustments done during flight by starting or stopping the timer at the appropriate orbit point.

3.4.4 DATA HANDLING AND STORAGE

3.4.4.1 Processing. Data generated by the experiments along with the spacecraft housekeeping information, are digitized in the S3 processor and formatted into a 16.384 kbps bit stream. To accommodate a more representative set of data channels, the original sub-sub-com multiplexer card is removed and replaced with a serial input gate card. This plus the inherent internal jumper wire format allows flexibility in matching incoming data to the S3 hardware. This will not alter the qualification status as the new card contains the same chips in addition to having the same form, fit, and function. Figure 3.4-7 describes the baseline input capacity of the PCM processor.

Analog and digital inputs are formatted into standard 8 bit words with 128 words per frame. Serial digital inputs of greater than 8 bits are accommodated by wire patching adjacent channel gating signals together to provide word lengths in increments of 8 bits. Each frame is sampled 16 times per second, giving 16,384 bits per second. Both analog and digital inputs are 0 to +5 volts. The telemetry format is shown in Figure 3.4-8.

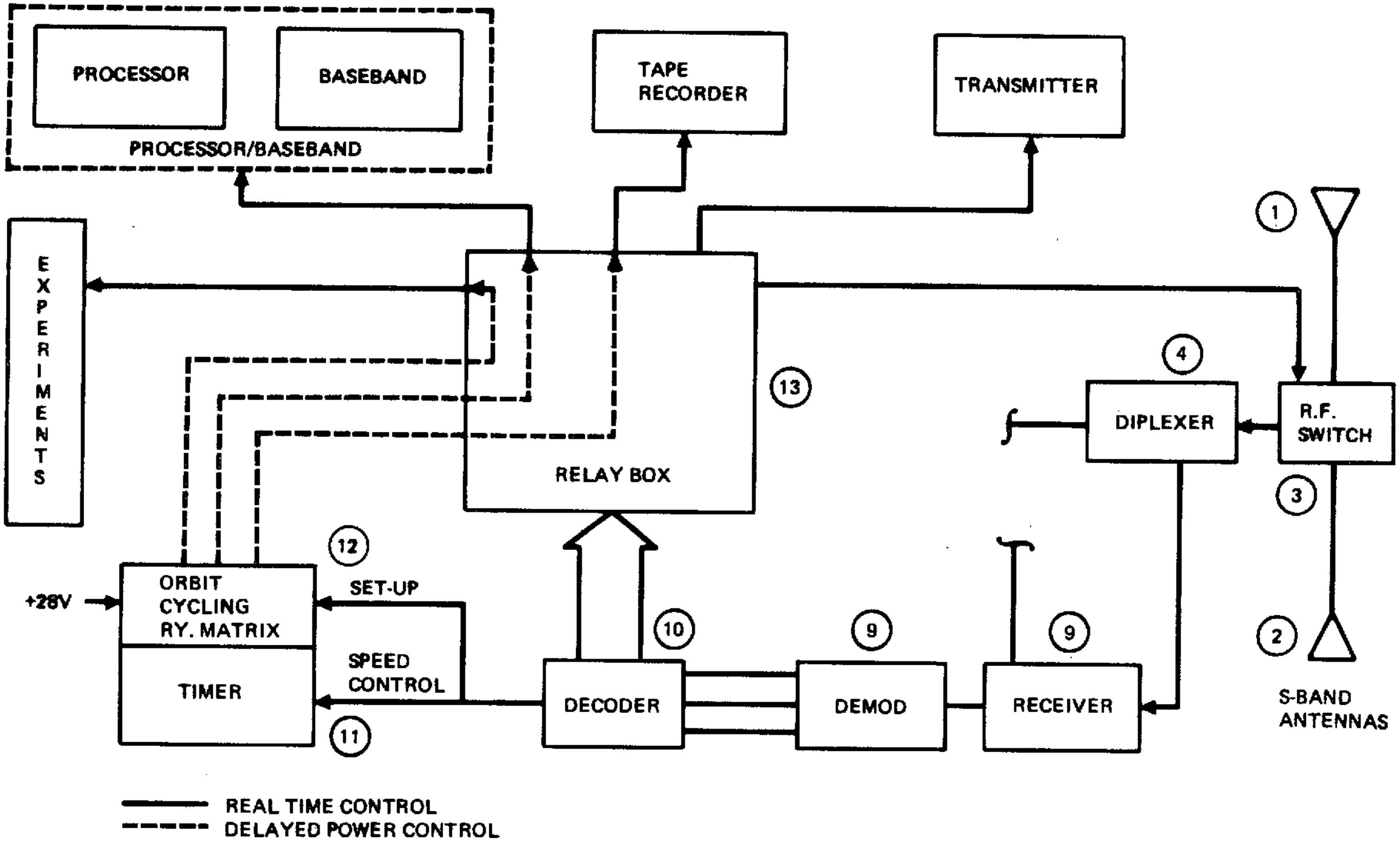
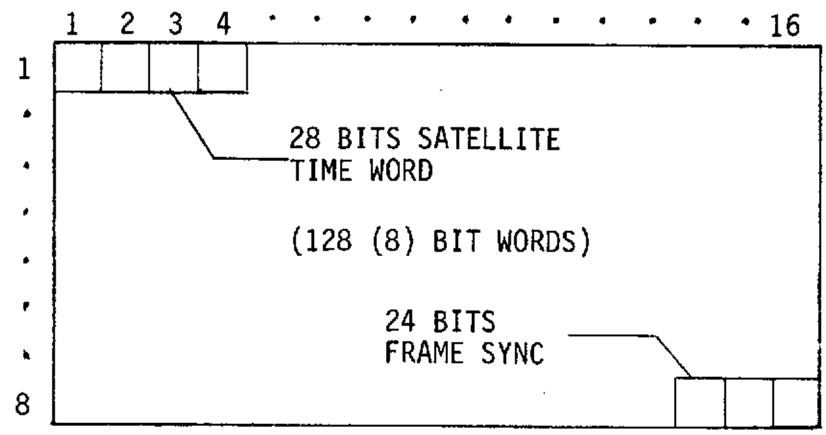


Figure 3.4-6. BSRM Command Control Configuration

NO. OF INPUTS EXPERIMENTS	BSRM S/C	TYPE OF INPUTS	SAMPLING RATES
6	--	ANALOG	16
--	2	ANALOG	16
--	7	ANALOG	4
--	37	ANALOG	1
76	--	ANALOG	1
82	46	TOTAL 128	
--	20	DISCRETES	1
24	--	DISCRETES	1
24	20	TOTAL 48	
2	--	SERIAL DIGITAL (8 BITS)	128
2	--	SERIAL DIGITAL (8 BITS)	64
1	--	SERIAL DIGITAL (16 BITS)	64
1	--	SERIAL DIGITAL (24 BITS)	64
7	--	SERIAL DIGITAL (8 BITS	32
1	--	SERIAL DIGITAL (64 BITS	32
1	--	SERIAL DIGITAL (40 BITS)	32
2	--	SERIAL DIGITAL (64 BITS)	16
--	1	SERIAL DIGITAL (16 BITS)	16
--	1	SERIAL DIGITAL (24 BITS)	16
--	1	SERIAL DIGITAL (28 BITS)	16
1	--	SERIAL DIGITAL (8 BITS)	16
--	1	SERIAL DIGITAL (16 BITS)	2
--	2	SERIAL DIGITAL (8 BITS)	2
1	--	SERIAL DIGITAL (8 BITS)	1
--	2	SERIAL DIGITAL (20 BITS)	1
--	1	SERIAL DIGITAL (8 BITS)	1
19	9	TOTAL 28	

Figure 3.4-7: PCM Processor Inputs

MAIN FRAME FORMAT



FORMAT CHARACTERISTICS

16 MAIN FRAMES/SECOND
128 WORDS/MAIN FRAME
8 BITS/WORD
16,384 BITS/SECOND

1 MASTER FRAME/SECOND
16 MAIN FRAMES/MASTER FRAME
2,048 WORDS/MASTER FRAME
16 WORDS/SUB COM

28 BITS/SATELLITE TIME WORD (STW)
24 BITS MAIN FRAME SYNC WORD
4 BITS BINARY SUBCOM ID (PART OF STW)

Figure 3.4-8: BSRM Telemetry Format

Spacecraft timing is generated by a 28 bit satellite time word giving a resolution of 61 microseconds. Every 194.2 days the clock recycles back to zero time. Stability is 1 port in 10^5 per day. The clock is used for synchronizing gating and timing experiment packages.

Other miscellaneous functions of the PCM processor are spin and precession coil current processing plus timer speed control. Also, located within the processor case is a Baseband unit and the combination is commonly called the Processor/Baseband unit.

3.4.4.2 Storage. Storage is provided by a two track S3 recorder having 2.10^8 bit storage capacity. Recording at 16 kbps results in 210 minutes of continuous data storage. For playback the speed is increased to 14 times, hence outputting 229 kbps PCM. One orbit of 95 minutes can be played back during a 6.8 minute pass with the capability of two full orbits storage. Bit error (BER) of the recorder is 10^{-6}.

3.4.5 TELEMETRY TRANSMISSION

Real time 16 kbps from the S3 PCM processor phase shift keys (PSK) a 1.024 MHz subcarrier within the baseband. During playback the 229 kbps phase shift keys a 1.7 MHz subcarrier. These two subcarriers are then combined linearly along with ranging tones, into a composite signal. Within the baseband commands control ON-OFF functions on the 1.7 MHz SCO and ranging signal. The real time 16 kbps SCO at 1.024 MHz normally runs continuously and has no control. Figure 3.4-9 describes the transmission configuration.

The composite has two modes; Mode I for real time and Mode II for playback. In Mode I a 1.7 MHz VCO is carried as spare and could be used to transmit another channel of PCM or wideband analog data as in the S3 satellites. Mode II, although a playback mode, also incorporates the real time 16 kbps transmission so that data is never lost due to recorder playback. Transmission of the composite signal is through the S3 8.5 watt S-band PM transmitter at 240/221 times the uplink command frequency. A modulation bandwidth of 100 Hz to 2 MHz passes all necessary signal components of the transmitted data.

Down link calculations using the 8.5 watt transmitter and the STDN 30 ft. parabolic tracking antenna shows greater than 6 dB signal margin out to 7000 km slant range (orbital altitude of 3400 km @ 5° elevation). This exceeds all expected mission requirements utilizing S3 mod index values and a ranging turn around ratio of .5 (uplink/downlink). Link optimization and refinements can be made in subsequent phases of this study. Figure 3.4-10 summarizes the R.F. downlink.

3.4.6 MODIFICATIONS TO THE S3 TT&C SYSTEM

The S3 TT&C subsystem requires no changes of significant impact to meet the BSRM baseline needs. The following describes briefly each component identified in Figures 3.4-6 and 3.4-9 by a numbered circle x , and the identified changes.

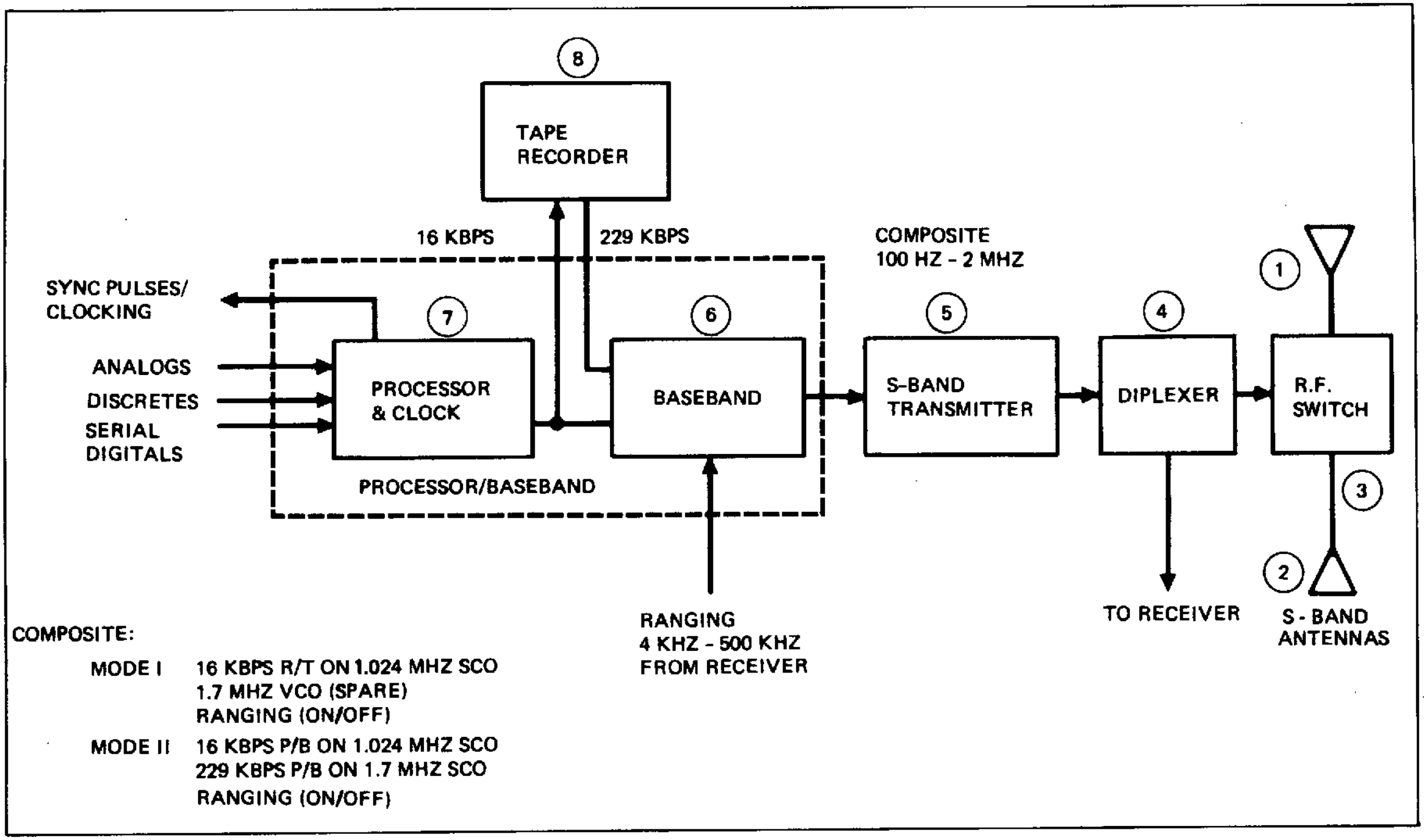


Figure 3.4-9. BSRM Telemetry Transmission Configuration

PARAMETER	WORST CASE VALUE	SOURCE
1. S/C XMTR PWR.	+ 39.3 dBM	8.5 W (MIN.) S3
2. S/C CIRCUIT LOSSES	- 2.5 dB	S3
3. XMIT. ANT. GAIN	- 1.2 dB	S3
4. SPACE LOSS (7000 KM S.R.)	- 176.4 dB	@ 2250 MHZ
5. POLARIZATION LOSS	0.0 dB	BOTH RHCP
6. ATMOSPHERIC LOSS	- .6 dB	@ 5^{O} ELEVATION
7. RECEIVING ANT. GAIN	+ 44.0 dB	USB 30' ANT. STDN. 101.1
8. TOTAL RCVD. POWER	- 97.4 dBM	SUM 1-8
9. RCVR. NOISE DENSITY	- 176.3 dB-Hz	T_S = 170^{O}K STDN 101.1
CARRIER PHASE LOCK ACQUISITION PERFORMANCE		
10. MODULATION LOSS	- 6.6 dB	CALCULATED
11. RECEIVED CARRIER PWR.	- 104.0 dBM	8 + 10
12. CARRIER NOISE B.W. (600 Hz)	27.8 dB	GSFC
13. REQUIRED SNR IN B.W.	7.0 dB	GSFC
14. THRESHOLD POWER	- 141.5 dBM	9 + 12 + 13
15. PERFORMANCE MARGIN	+ 37.5 dB	11 LESS 14
PCM DATA PERFORMANCE (1.024 MHZ REAL TIME		
16. MODULATION LOSS	- 11.0 dB	CALCULATED S3 (MI = .78)
17. RECEIVED S.C. PWR	- 108.4 dBM	8 + 16
18. S.C. NOISE B.W. (16 KHZ)	42.2 dB	CALCULATED
19. REQUIRED SNR IN B.W.	12.0 dB	GSFC @ 10^{-5} BER
20. THRESHOLD POWER	- 122.1 dBM	9 + 18 + 19
21. PERFORMANCE MARGIN	+ 13.7 dB	17 LESS 20
PCM DATA PERFORMANCE (1.7 MHZ PLAYBACK)		
22. MODULATION LOSS	- 3.9 dB	CALCULATED S3 (MI = 1.4)
23. RECEIVED S.C. PWR.	- 101.3 dBM	8 + 22
24. S.C. NOISE B.W. (229 KHZ)	53.5 dB	CALCULATED
25. REQUIRED SNR IN B.W.	12.0 dB	GSFC @ 10^{-5} BER
26. THRESHOLD POWER	- 110.8 dB	9 + 24 + 25
27. PERFORMANCE MARGIN	+ 9.5 dB	23 LESS 26
RANGING PERFORMANCE (SINGLE TONE)		
28. MODULATION LOSS	- 20.0 dB	CALCULATED (MI = .3)
29. RECEIVED POWER	- 117.4 dBM	8 + 28
30. NOISE B.W. (1 HZ)	0 dB	GSFC
31. REQUIRED SNR IN B.W.	35 dB	GSFC
32. THRESHOLD POWER	- 141.3 dBM	9 + 30 + 31
33. PERFORMANCE	+ 23.9 dB	29 LESS 32

Figure 3.4-10: BSRM Telemetry R.F. Downlink Summary

1 2 Antennas - No change.

3 R.F. Switch - No change.

4 Diplexer - The original unit will be replaced by one from the same supplier having proven flight history but tuneable to the STDN 240/221 S-band frequencies. A thermo-vac power test will verify BSRM operating levels.

5 8.5 W Transmitter - No change.

6 Baseband - Delete FM/FM P.C. card A3 and substitute dummy. Broaden 1.7 MHz B.P. Filter to pass 229 vs. 131 kbps.

7 PCM Processor - Delete sub-sub mux P.C. card and insert new digital input card. Rewire strapping jumpers to obtain required channels, sample rates, and experiment clocking signals.

8 Tape Recorder - Change playback speed ratio from 8:1 to 14:1 using flight proven circuits of other model recorders.

9 Receiver/Demodulator - This single unit will be replaced by two separate units from a different supplier having STDN compatible hardware available.

10 Command Decoder - Change logic P.C. card to one compatible with STDN format.

11 Timer - No change.

12 Orbit Cycling Matrix - No change.

13 Relay Box - Same components and construction but wired to accommodate BSRM peculiar experiments.

3.5 SOFTWARE

ATTITUDE DETERMINATION AND CONTROL IS PERFORMED USING SATELLITE SENSOR DATA PROCESSED BY SOFTWARE WITHIN THE GROUND SYSTEM. CONCURRENTLY DURING THE MISSION AN ATTITUDE DETERMINATION SOFTWARE FUNCTION IS ALSO CARRIED OUT IN SUPPORT OF PAYLOAD DATA REDUCTION REQUIREMENTS. THE STRUCTURE AND ORGANIZATION OF THESE SOFTWARE PACKAGES ARE IDENTICAL TO THE SMALL SATELLITE ATTITUDE CONTROL (SSAC) AND SMALL SATELLITE ATTITUDE DETERMINATION (SSAD) PROGRAMS DEVELOPED BY BOEING FOR PREVIOUS PROGRAMS.

Changes within subprograms will be necessary to support changes of the satellite dynamics, sensor system outputs, differing payload attitude determination requirements and to adapt to NASA facility interfaces. The overall interfaces of the total software system are outlined in Figure 3.5-1.

Use of the basic SSAD and SSAC software structure ensures:

- Low duty cycle at Operations Control Center (OCC).
- Same technical specialists as utilized on S3 to provide continuity in preparing program modifications.
- Minimal attitude determination software development required by NASA and participating payload agencies.

3.5.1 OPERATIONAL CONTROL SOFTWARE

The SSAC software, implemented at the Operations Control Center (OCC) determines vehicle attitude and computes all required satellite orientation and spin rate adjustment commands. BSRM response to commands is then predicted to support on-orbit operational planning.

Telemetry data of satellite attitude sensors, wheel speed (in the case of 3-axis systems) and control coil status required by SSAC is supplied to the OCC. These data are received at the network and relayed to the OCC. These data are then extracted, pre-edited to eliminate gross outliers and converted to engineering units prior to use by the attitude determination subprogram. Ephemeris data required by SSAC are available through the orbit determination interface. Knowledge of vehicle attitude and control system status is then utilized to select commands for vehicle attitude and/or speed adjustment in conformance with operational requirements. Printer outputs are used to display computed results and current values of data base parameters. Command transmission is performed through the operations network to the BSRM.

3.5.2 EXPERIMENT DATA REDUCTION AND SUPPORT SOFTWARE

The SSAD software is comprised of two elements. The major one, the Estimation Module (EM), is implemented at the OCC. The secondary one, the Output Module (OM), is used by the experimenter. The EM determines a time history of vehicle ephemeris and attitude from satellite sensor data and reference ephemeris vectors. These orbital time histories are compacted and transmitted to the experimenter in data card form for input to the OM. The OM enables the experimenter to obtain the orbit state vector and the line of sight of his sensor(s) time correlated to data observations.

Raw telemetry data tapes are shipped to the OCC from the Network Tracking Sites. These analog tapes are digitized and reformatted into Standard Telemetry Format (STF) tapes. These tapes contain all BSRM housekeeping, attitude, and payload data, a time correlation record from a time correlation library and the most recent ephemeris state vector. The first subprogram of the EM reads the STF, extracting the attitude sensor and control status data together with the time correlation and ephemeris reference vector records. The sensor data is time correlated and pre-edited for gross outliers. The data span time is computed and the reference ephemeris vector is integrated to obtain a number of valid ephemeris state vectors over the data span. The attitude determination subprogram then uses this data to obtain a vehicle attitude history over the data time span. This is followed by a curve fitting subprogram that compacts the time history into a time dependent mathematical function. Finally the ephemeris state vectors and the coefficients of the attitude function are output in card form and transmitted to the experimenters for use in their OM programs.

The experimenters receive an STF tape containing their payload data and a corresponding set of OM input cards. The experiment data reduction program can then scan their experiment data, time correlate an event of interest to them, and, by calling the OM with that time, obtain the satellite attitude and/or the line of sight of a particular sensor as well as the satellite ephemeris for the event time of interest.

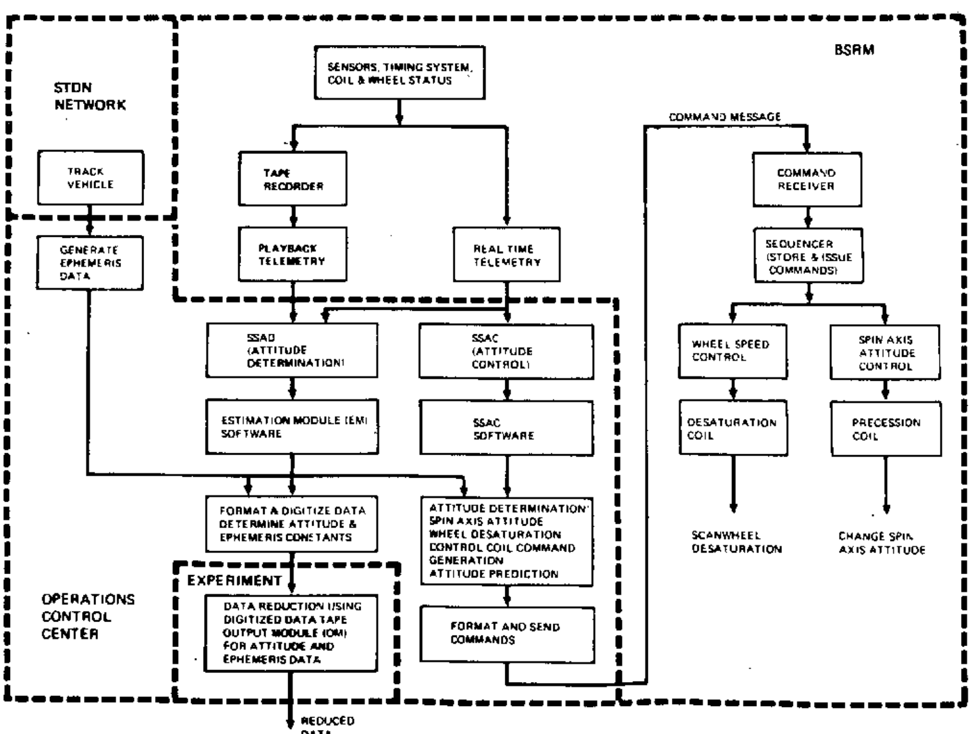


Figure 3.5-1. Software Interface Diagram

3.6 THERMAL CONTROL

THE THERMAL CONTROL CONCEPT USED ON BSRM IS IDENTICAL TO THE FLIGHT-PROVEN SYSTEM USED ON S3.

The thermal control subsystem design (Figure 3.6-1) is based on isolating the satellite from the environment except by modulated radiation exchange through temperature controlled louvers. In addition, a combination of open structure, equipments coated with highly emissive paint, and the equipment thermal mass are used to hold the satellite interior to ± 10°F through all operating modes. A single louver is dedicated to the battery to maintain its temperature to ± 5°F throughout the mission. Two additional louvers control the temperature of the remainder of the satellite.

The basic thermal design for the spinning satellite is directly applicable to a similarly shaped 3-axis stabilized satellite. The only real difference is the temperature profiles of the solar panels, and the temperature extremes are well within the present qualification ranges.

Typically, all internal components except the battery and transmitter are maintained in a range of 50 to 90°F. The desired temperature qualification range for satellite mounted experiments is 20 to 120°F to avoid special thermal control provisions. Limits outside this range may be accommodated through the use of local heaters or additional space radiators. Table 3.6-1 shows typical equipment, qualification ranges, and orbit predictions for the various flight-proven S3 satellites. All items proposed for the thermal control subsystem are flight qualified and are used on the current S3 designs.

A thermal math model is used to design, analyze and predict the performance of the proposed satellite and is identical in structure to the math models used for the present S3 satellites. This includes a special routine written to incorporate the thermal effects of the power subsystem such as battery heating during charge and discharge. The model is used to predict satellite temperatures from launch through full orbital operation.

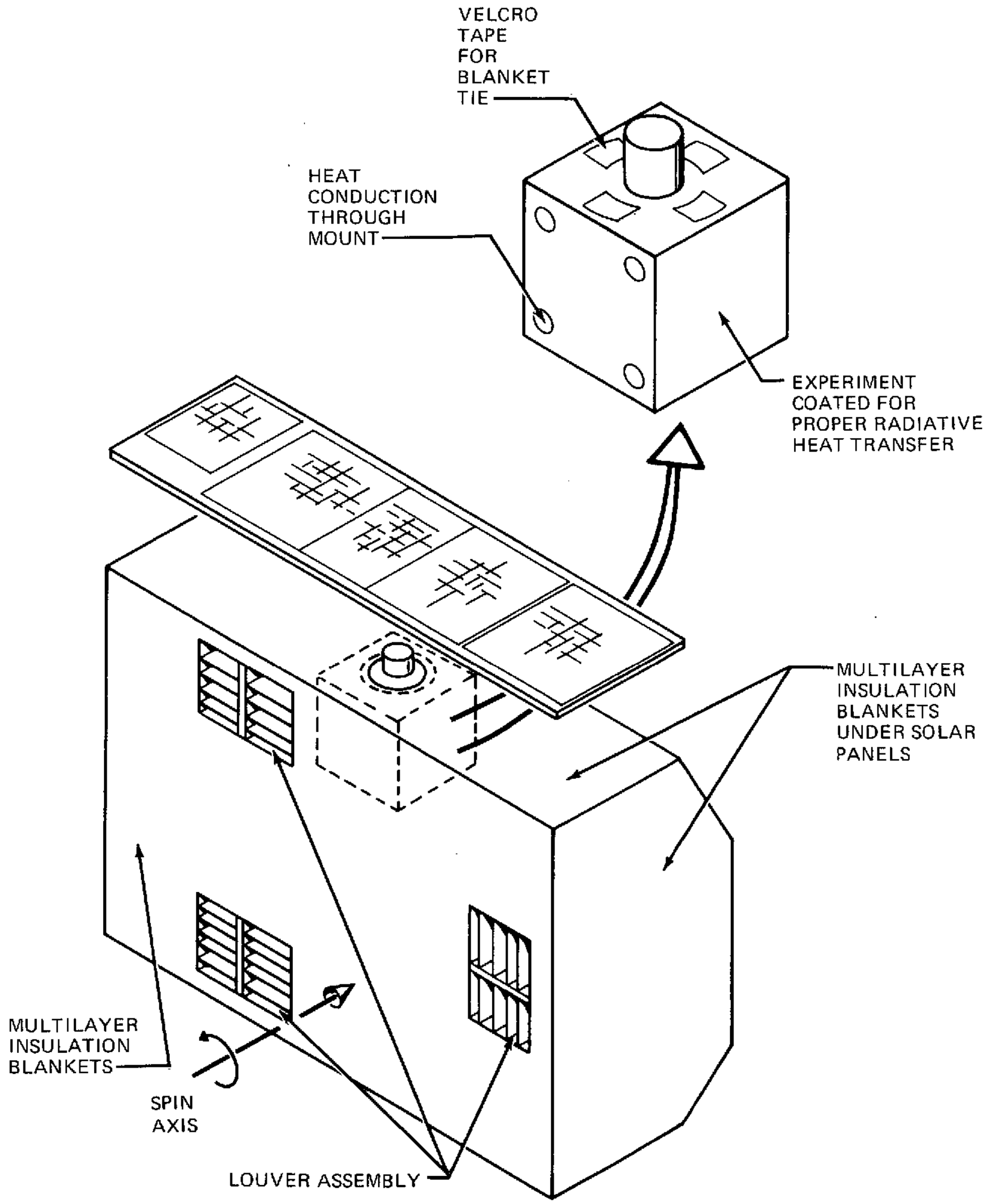


Figure 3.6-1. Thermal Control System Features

Table 3.6-1. Typical Component Thermal Characteristics

Satellite components	Qual. range (°F)	S3 Orbital predictions (°F)
Tape recorder	0 to 120	54 to 70
Command decoder	–31 to 149	55 to 73
Processor	–35 to 153	63 to 86
10W transmitter	0 to 160	52 to 107
Receiver/demodulator	–31 to 160	55 to 73
Timer	–67 to 212	53 to 71
Batteries (flight S3-1)	23 to 125	53 to 64
(flight S3-2)	23 to 125	61 to 71
Voltage limiter	–31 to 149	54 to 109
Magnetometer electronics	–4 to 140	52 to 72
Sun sensor	–4 to 140	48 to 77
Earth sensor	–4 to 140	51 to 76
Solar panels	–148 to 289	–106 to 237

3.7 AGE, GHE AND FACILITIES

EXISTING GROUND EQUIPMENT AND FACILITIES ARE AVAILABLE TO SUPPORT THE BSRM PROGRAM. NO ADDITIONAL FACILITIES WILL BE REQUIRED BY EITHER BOEING OR ITS SUBCONTRACTORS. THE FACILITIES PROPOSED FOR USE ON THIS PROGRAM HAVE BEEN USED ON SIMILAR TYPE WORK AND MEET ALL REQUIREMENTS FOR ENVIRONMENT, SECURITY AND TECHNICAL PERFORMANCE REQUIRED FOR THE BSRM.

Certain existing government facilities presently assigned to Boeing's subcontractors will be utilized on the program as provided for in the contracts covering these facilities.

The BSRM will be produced, handled and tested using existing AGE and GHE developed for the S3 program. The AGE/GHE is conventional equipment necessary for satellite handling and test. The Boeing-owned Mobile Test Lab, described in document D180-18431-2, will be used on the BSRM program.

3.7.1 ELECTRICAL AGE

Electrical AGE will be provided to support functional testing of the BSRM satellite with and without payloads installed. The equipment will be entirely portable, permitting support of testing at Boeing and any launch site.

Figure 3.7-1 shows a schematic of the BSRM electrical AGE and identifies the specific equipment items. All AGE shown is available from the S3 program with two minor differences for BSRM:

- o Additional breakout boxes will be required for the different connectors on added subsystem components.
- o The earth sensor stimulator will be a different unit for the 3-axis BSRM.

The similarity of S3 and BSRM subsystems will allow a large part of the S3 test procedures and computer software to be utilized without modification in the BSRM test program.

3.7.1.1 Control and Monitor Test Set (CMTS). The CMTS is a Boeing-owned set of equipment configured for testing S3 satellites. It is contained in three standard equipment racks and performs the following functions:

- o Provides electrical power to the satellite main bus and simulates solar array output.
- o Controls satellite functions (such as stepping the satellite timer), through the Mobile Test Lab computer.
- o Visually monitors critical satellite functions (such as battery voltage and transmitter temperature).
- o Manually controls satellite power, sequencing and monitoring functions during test operations.
- o Visually monitors test sequences.

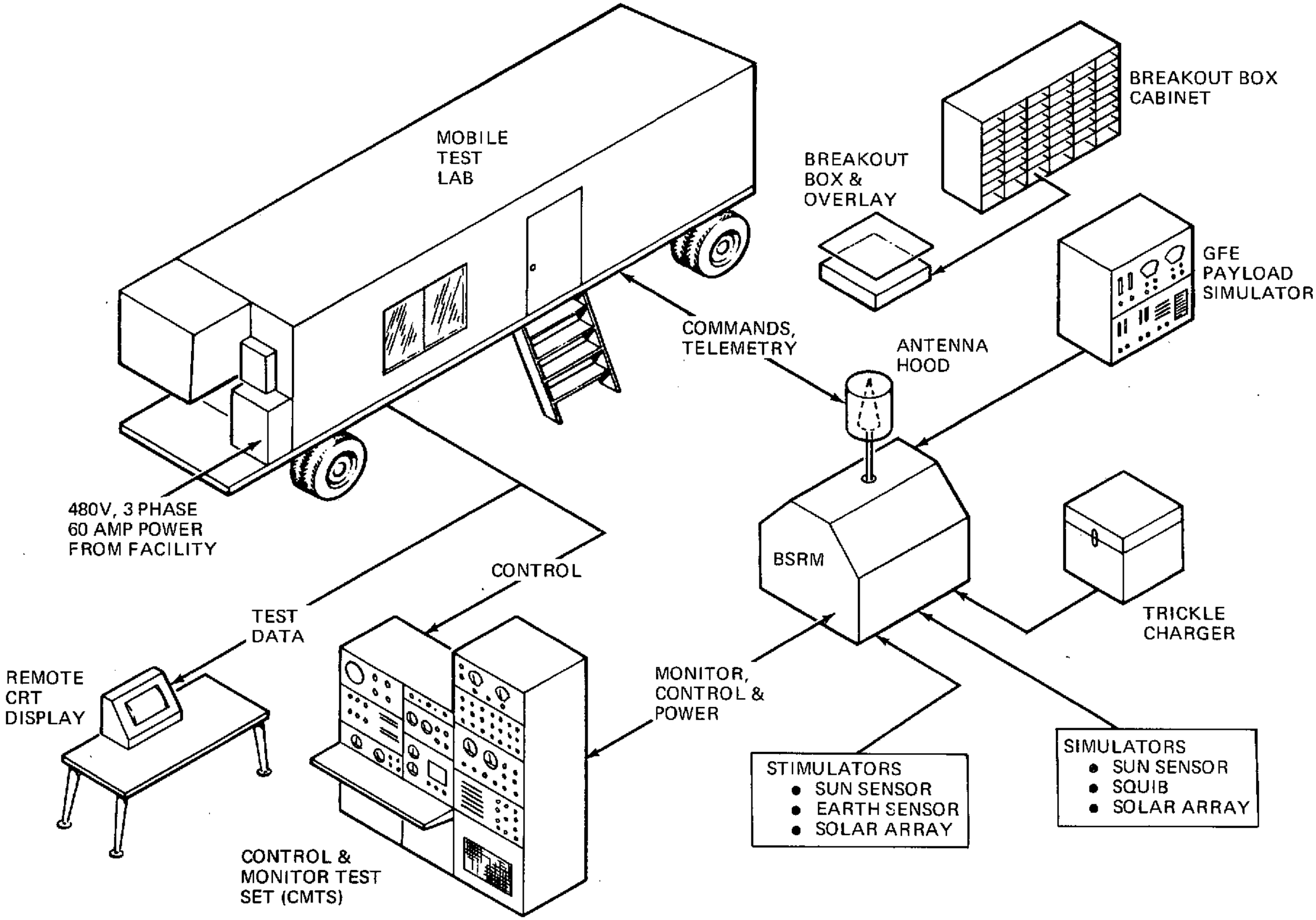


Figure 3.7-1. BSRM Electrical AGE

- o Troubleshooting with self-contained standard test equipment (oscilloscope, digital voltmeter, etc.).

The existing CMTS can be used essentially as is to support the BSRM program. A remote CRT which displays computer controlled test sequences is located at the CMTS to provide visibility for the test team.

3.7.1.2 Other Electrical AGE. The following additional items are available from the S3 program to support BSRM operations with little or no modification:

Battery Trickle Charger. This unit maintains the satellite flight batteries at a full state of charge until disconnected during the launch countdown. It is housed in an explosion-proof suitcase container with visual and audible alarms which are activated if the battery terminal voltage or charge current reaches preset limits.

Break-In Boxes. A break-in box will be provided for each type of satellite and satellite/experiment interface connector to support special tests and troubleshooting. Each box will have the capability of signal monitoring through plug-in lights and each connection can be opened or jumpered individually to provide current monitoring or individual circuit disabling.

Squib Simulator. This suitcase test equipment contains circuit breakers and indicator lights to simulate satellite squib loads and verify current supply capability of squib firing circuits.

Antenna Hoods. Antenna hoods are available from the S3 program for open loop command and telemetry communication between the BSRM and Mobile Test Lab.

Stimulators. Sun and earth sensor stimulators and a "sun-gun" solar array stimulator are available.

Simulators. A vendor-supplied sun sensor simulator is available. The CMTS contains a solar array simulator (power supply) to provide power to the BSRM satellite during testing.

3.7.1.3 Payload Electrical Simulators. An electrical simulator with operating procedures will be required from payload contractors for each experiment that interfaces with the satellite. The simulators will be used to:

- o Check out the BSRM systems at the experiment interface prior to installation of flight payloads.
- o Verify proper connections to payload without risking damage to flight experiments.
- o Aid in fault isolation in the event of indicated payload malfunction.

The payload simulators will be required to duplicate experiment orbital electrical loads, indicate receipt of command, and provide termination of telemetry input circuits to the BSRM satellite. Complete requirements for the payload simulators and operating procedures will be included in the Satellite/Experiment ICD's. The payload contractor will provide support for payload installation and alignment, servicing, special monitoring, stimulation and data evaluation. Also a payload representative will generally be present when AGE is being used and the payload operated.

3.7.1.4 Launch Site AGE. The following equipment items are required at each BSRM launch site to measure and verify signal strength and command format during compatibility and prelaunch testing:

- o S-band receiver with input capability matching satellite requirements and an output of individual 5, 0, 1 bits and a 500 Hz rectangular wave form.
- o Strip chart recorder with 20 Hz frequency response on two channels to record strength and time code.

The MTL requires three phase, 480 volt, 60 amp. facility power. The MTL is supplied with one 30 ft. cable for power connection.

3.7.2 GROUND HANDLING EQUIPMENT

Existing GHE from the S3 program will be used to support the BSRM requirements. Minor modifications are required to some items to adapt to the Scout launch vehicle processing. The use of existing GHE results in complete flexibility of operations in test and transportation. Either vertical or horizontal installation on test fixtures can be accommodated. The following paragraphs briefly summarize the GHE:

Container/Transporter. The existing S3 transportation dolly also serves as a shipping container. Shipping shock and vibration isolation is achieved with polyurethane foam cemented between two aluminum decks. An overbox is installed on the dolly during shipping. Provisions are incorporated in the existing design for contamination control barriers filled with dry nitrogen gas.

Handling Frame. A light-weight aluminum frame is used to lift the satellite for installation onto test fixtures or the transportation dolly. Sling attachment points permit rotation of the satellite to either horizontal or vertical positions to accommodate any test orientation.

Handling Sling. A single assembly is provided with interchangeable fittings to permit lifting of the satellite and handling frame in any orientation.

Solar Panel Containers. Each panel is individually packaged in non-static clean bags and purged with dry nitrogen gas to provide contamination control.

Cover - Solar Panel Protection. The solar panels will be installed on the satellite during the majority of the program. Damage to the panels during processing will be prevented by the installation of a clear hard protective cover mounted to the satellite structure. It is easily removed for satellite access, testing and flight.

Container - Battery Storage/Shipping. Each battery will have storage/shipping container to provide adequate protection for the battery during storage and shipping.

3.7.2.1 Payload GHE. Each payload agency will provide any unique GHE with operating procedures to meet any experiment requirements. The payload contractor will provide support for payload handling as required.

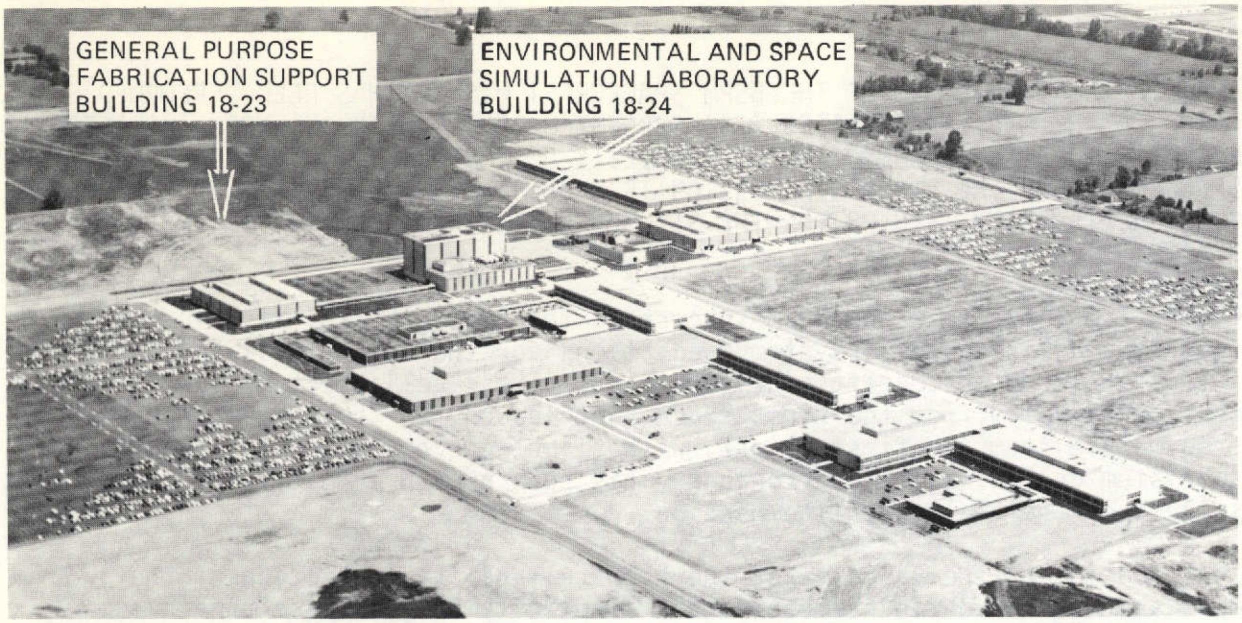
3.7.2.2 Transportation. The BSRM with experiments installed will be shipped to the launch sites by air or truck (no rail) in the existing satellite container/transporter. For truck transportation, a dedicated air-ride van will be used. No shock or vibration test equipment will be required as the existing transportation dolly has been tested on the S3 program to verify that transportation loads do not exceed flight loads.

3.7.3 FACILITIES

The Boeing Company proposes to accomplish the BSRM program at its Space Center in Kent, Washington. This facility complex, constructed since 1965, was developed primarily to support space exploration programs. The Space Center is located just south of Seattle and has ready access to the other Boeing facilities in the Seattle area and to highway, rail, water and air transportation systems.

All BSRM engineering, fabrication, assembly and testing will be accomplished at the Space Center. The program offices will be located on the second floor of the 18.23 Building, Figure 3.7-2, in close relation to the fabrication and assembly of the spacecraft which will be on the first floor of the same building. Final assembly, integration, functional testing, magnetic EMI, and spin balancing will be accomplished in the clean hi-bay area of the adjacent 18.24 Building. This is the same facility where similar functions were performed for the SESP/STP and Lunar Orbiter vehicles. Where local conditions require, Farr clean benches and portable down-flow clean booths are available. In addition, other facilities such as a horizontal laminar flow clean room, pressure test facilities, tube cleaning facilities, specialized laboratories, and computers in other buildings at the Space Center will be utilized as required to support the BSRM.

Thermal-vacuum testing with solar simulation will be performed in Boeing Space Chamber A in the 18.24 Building. Orbital sequences will be simulated and real time and recorded data transmitted to the Mobile Test Lab. All facilities for conducting this test are available for BSRM.



BOEING SPACE CENTER – KENT, WASHINGTON

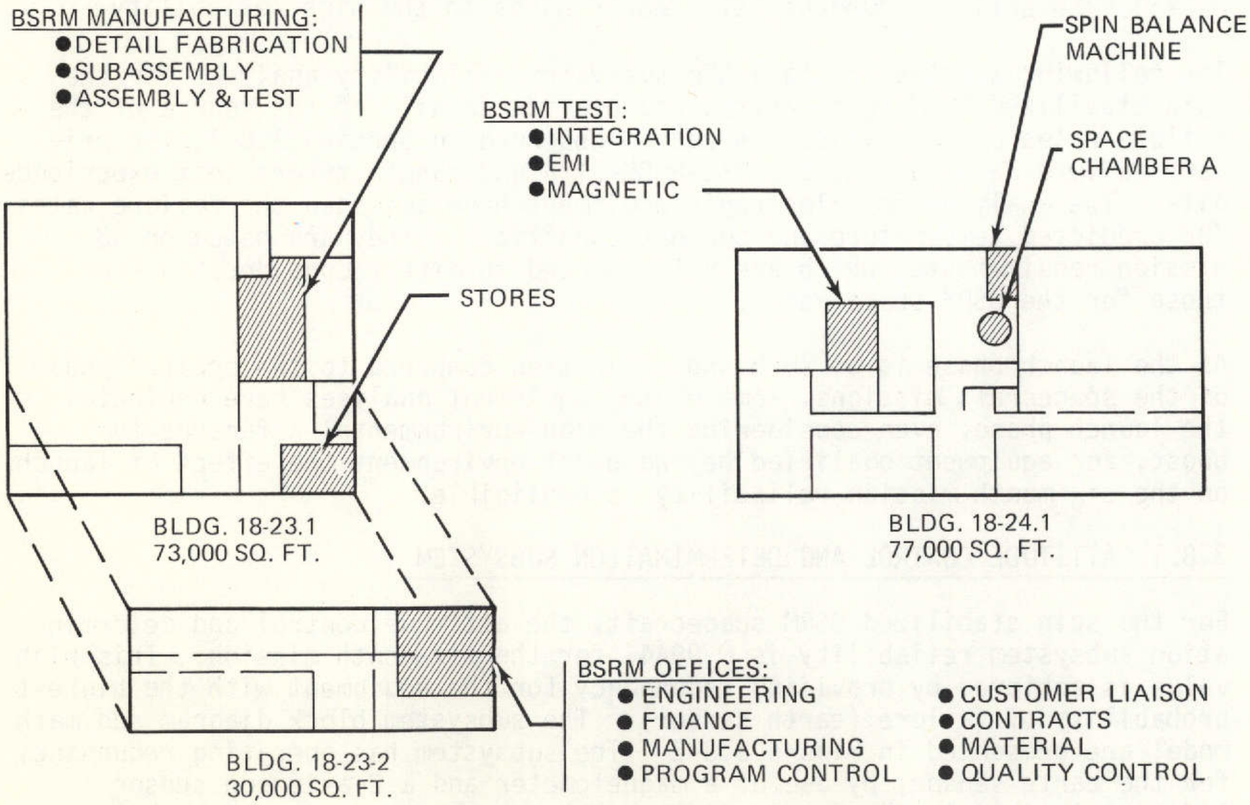


Figure 3.7-2. BSRM Co-located Facilities

3.8 RELIABILITY ANALYSIS

A RELIABILITY ANALYSIS OF THE BSRM SPACECRAFT MISSION HAS BEEN PERFORMED, BASED ON THE SMALL SATELLITES (S3) SUBSYSTEM RELIABILITY ANALYSES, MODIFIED FOR MINOR DESIGN CHANGES. THE RELIABILITY OF THE SPIN STABILIZED BSRM SPACECRAFT FOR THE SIX-MONTH MISSION IS 0.88; THE 3-AXIS STABILIZED SPACECRAFT RELIABILITY IS 0.88 FOR THE SIX-MONTH MISSION AND 0.78 FOR THE ONE YEAR MISSION.

Figure 3.8-1 presents the BSRM spacecraft top level reliability block diagram and lists the subsystem predictions for the spin stabilized and 3-axis stabilized spacecraft configurations. These reliability values are considered valid from the first spacecraft launch, as the system is considered mature from the first flight, with flight proven equipment, and as the subsystem design, with minor changes, has been proven effective by the current success of the S3-1 satellite mission.

Contributing to the validity of the high reliability prediction of the two BSRM spacecraft configurations is the simplicity of the space proven subsystem designs, 100% burn in electronic equipment, test proven, ample life margins for life limited equipment (i.e., battery and tape recorder), and the high quality of electronic parts (MIL-STD-883 Class B for microcircuits and JAN TX or MIL ER, level R or better, or equivalent, for other parts). Selective redundancy also contributes to the high reliability.

The following sections contain the subsystem reliability analyses for the spin stabilized BSRM spacecrafts, and provide details of the source of the failure rates and analysis. As can be observed in Section 3.8.7, the primary failure rate sources are MIL-HDBK-217A and manufacturers test experience data. The analyses for electronic equipment have adjusted the failure rates for predicted temperature and current conditions. They are based on S3 mission requirements, which are not expected to differ considerably from those for the BSRM spacecrafts.

As the launch phase is of such short duration compared to the orbital phase of the spacecraft missions, some of the equipment analyses have neglected the launch phase; even considering the high environmental K factors for boost, for equipment qualified beyond boost environment the effect of launch on the six-month mission reliability is negligible.

3.8.1 ATTITUDE CONTROL AND DETERMINATION SUBSYSTEM

For the spin stabilized BSRM spacecraft, the attitude control and determination subsystem reliability is 0.9944, for the six-month mission. This high value is achieved by providing redundancy for the equipment with the highest probability of failure (earth sensor). The subsystem block diagram and math model are presented in Figure 3.8-2. The subsystem has operating redundancy for the earth sensor, by use of a magnetometer and a 2-axis sun sensor. Additional partial redundancy exists, although not shown, as the subsystem can operate after failure of both the earth sensor and the sun sensor. Although this is a valuable design feature, for reliability purposes this additional capability has not been considered in the math model, or evaluated,

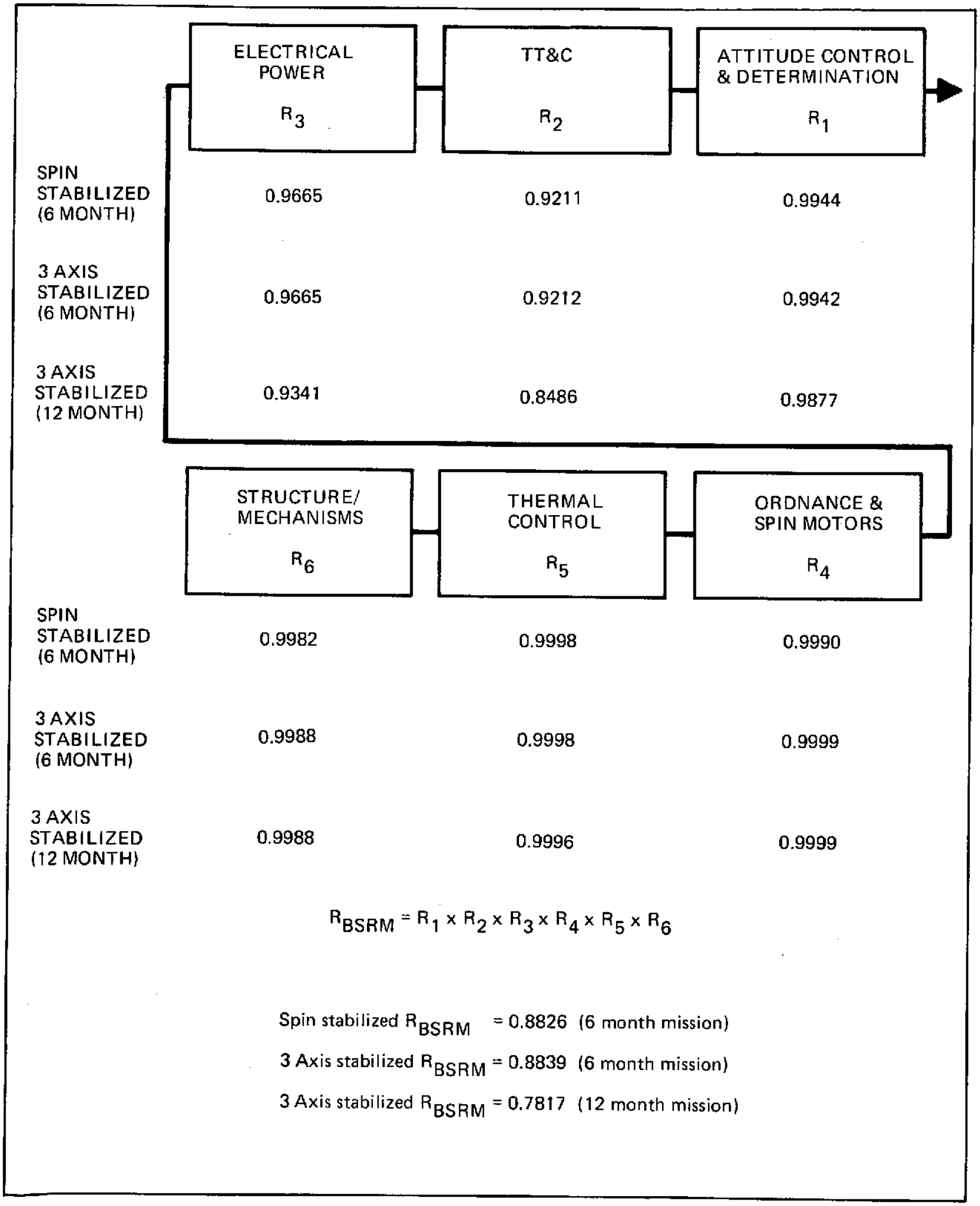
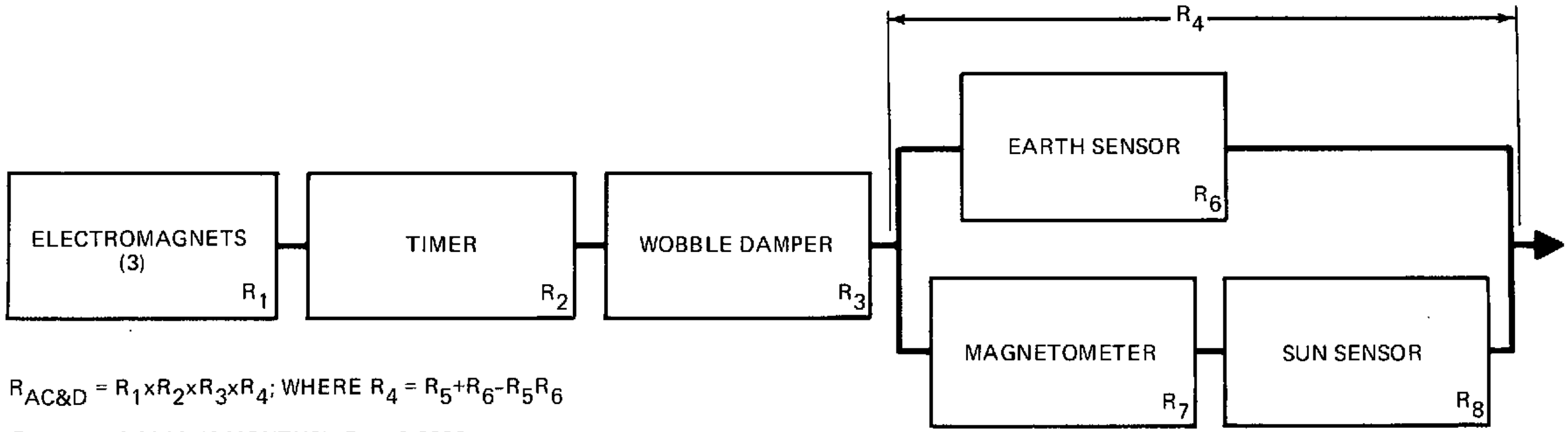


Figure 3.8-1. BSRM Spacecraft Reliability Block Diagram



$R_{AC\&D} = R_1 \times R_2 \times R_3 \times R_4$; WHERE $R_4 = R_5 + R_6 - R_5R_6$

$R_{AC\&D} = 0.9944$ (6 MONTHS) $R_4 = 0.9998$

$R_5 = R_7 \times R_8 = 0.9883$

Item	$\lambda \times 10^6$	D	t	N	R	Data source
1. Electromagnets	.05	1	4320	3	0.9993	1
2. Timer	-	1	4320	1	0.9954	1
3. Wobble damper	0	1	4320	1	0.9999+	Structural allowance
6. Earth sensor	3.54	1	4320	1	0.9848	1
7. Magnetometers	1.69	1	4320	1	0.9927	1
8. Sun sensor	1.02	1	4320	1	0.9956	1

1 Analysis data sources are referenced in section 3.8.7

Figure 3.8-2. Attitude Control & Determination Subsystem (Spin Stabilized) Reliability Block Diagram as the satellite operation would be in a degraded mode. Figure 3.8-2 also presents the subsystem equipment reliability; for the entire subsystem, the duty cycle D is one (100%), therefore, the equipment reliability $R = e^{-N\lambda t}$, where N is the number of items, λ is the failure rate, and t = 4320 hours (six months). The reliability of the wobble damper is a conservative structural allowance, as its operation is due to internal friction, and with proper design, its probability of failure is remote. An overall failure rate for the timer has not been provided, as the Celesco analysis has been performed developing detailed individual circuit failure rates for the various timer subfunctions, with individual duty cycles. The solid rocket spin motors have been included in the ordnance subsystem.

For the three axis stabilized spacecraft, the subsystem reliability is 0.9942 for the six-month mission, and 0.9877 for one year. The subsystem is similar to that of the spin stabilized spacecraft, with the exception that the earth sensor has been replaced by a scanwheel, and the redundant sun sensor is a more complex, two axis unit. Figure 3.8-3 presents the three axis stabilized subsystem reliability block diagram and mathematical model, as well as the equipment reliability for a six-month mission. The subsystem also provides additional capability inasmuch as it can operate, in a degraded mode, with the magnetometer, after failure of both the scanwheel and the sun sensor. It is noted that the scanwheel has a considerable amount of internal redundancy.

3.8.2 TT&C SUBSYSTEM

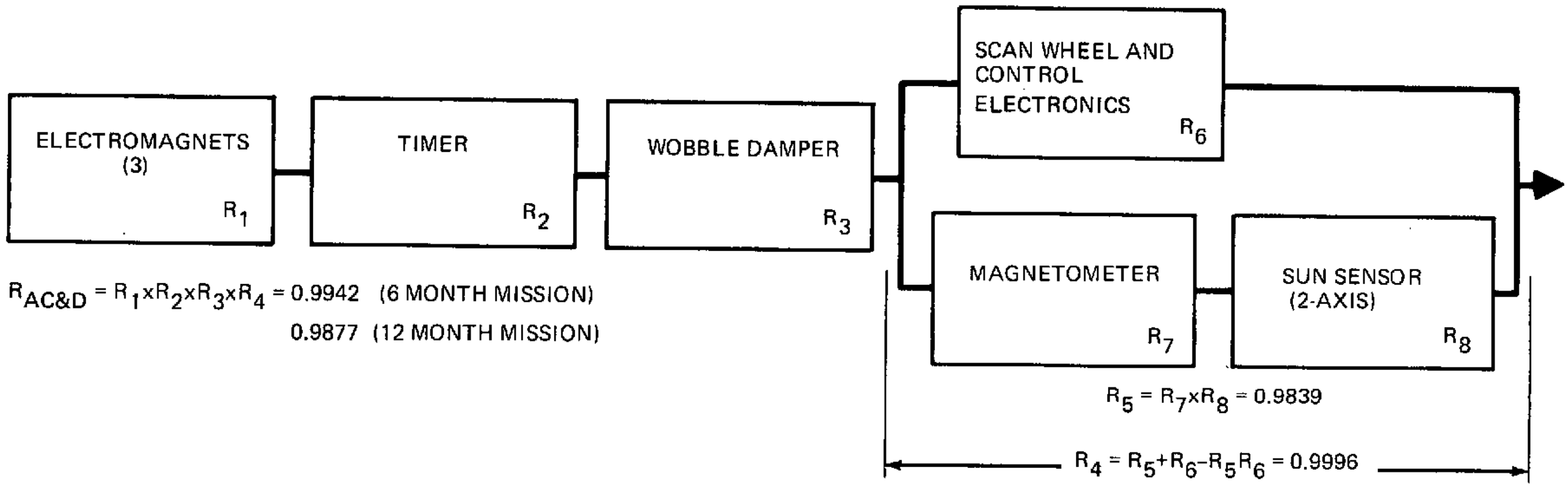
The TT&C subsystem is nearly identical for both the spin stabilized and three axis stabilized BSRM spacecrafts. The difference is in the antennas: The spin stabilized configuration requires two antennas on deployable booms, whereas the 3-axis stabilized unit requires only one fixed antenna. The subsystem reliability is 0.9211 for the spin stabilized spacecraft, and for the 3-axis stabilized spacecraft the reliability is 0.9212 for the six-month mission and 0.8486 for one year. Figure 3.8-4 presents the subsystem reliability mathematical model and tabulates the reliability of the equipment. A block diagram has not been provided as all the equipment is in a series relationship.

Equipment duty cycles (D) have been estimated to be similar to those for the S3-1 satellite mission. Equipment reliability R has been evaluated considering dormant failure rates as a factor K of operating failure rates . Thus, the equipment reliability $R = e^{\lambda tD + K\lambda t(1-D)}$, where t is the total mission time (4320 hours for the six-month mission).

Alternate configurations with different tape recorders have not been evaluated, but are considered to affect the third significant number of the total spacecraft reliabiltiy.

3.8.3 POWER SUBSYSTEM

The power subsystem for both the spin stabilized and 3-axis stabilized BSRM spacecraft configurations are identical. The reliability for the subsystem is 0.9665 for both configurations, for the six-month mission and 0.9341 for the 12 month mission of the 3-axis stabilized spacecraft.



Item	$\lambda x10^6$	D	t	N	R	Data source
1. Electromagnets	.05	1	4320	3	0.9993	1
2. Timer	-	1	4320	1	0.9954	
3. Wobble damper	0	1	4320	1	0.9999+	
6. Scan wheel & electronics	5.79	1	4320	1	0.9753	
7. Magnetometer	1.69	1	4320	1	0.9927	
8. Sun sensor (2-axis)	2.04	1	4320	1	0.9912	1

1 Analysis data sources are referenced in section 3.8.7

Figure 3.8-3. Attitude Control & Determination Subsystem (3-Axis Stabilized) Reliability Block Diagram

Item	$\lambda x10^6$	K*	t	D	N	R	Data source
1. S-Band antenna	.04	1	4320	.5	2 (1)**	0.9998 (0.9999)	MIL-HDBK-217A
2. Diplexer	.28	1	4320	1	1	0.9988	2
3. RF switch	4.0/CY	N/A	500CY		1	0.9980	
4. Receiver/demodulator	3.18	.77	4320	.07	1	0.9892	
5. Transmitter	5.63	.1	4320	.1	1	0.9954	
6. Processor	13.50	.4	4320	1	1	0.9701	
7. Command decoder	12.15	.24	4320	.05	1	0.9855	
8. Tape recorder	8.70	.1	4320	.25	1	0.9878	
9. Relay box	4.97	-	4320	.20	1	0.9940	2

$$R_{TT\&C} = \prod_{1}^{9} R_n$$ = 0.9211 (For spin stabilized spacecraft, 6 month mission)
0.9212 (3-Axis stabilized spacecraft, 6 month mission)
0.8486 (3-Axis stabilized spacecraft, 12 month mission)

* Ratio of operating failure rate to dormant failure rate.

** For 3-axis stabilized spacecraft configuration only.

1 20% full oper 80% standby

2 Analysis data sources are referenced in section 3.8.7

Figure 3.8-4. TT&C Subsystem Reliability

The baseline electrical power subsystem, nearly identical to that demonstrated successful in the S3-1 satellite, employs an array of three solar panels with 2x2 cm solar cells, and a nickel cadmium battery to provide the necessary electrical energy. Bus voltage and battery charge current control is accomplished by a shunt regulator that monitors bus voltage and shunts excess current through a shunt resistor panel when the upper limit of 32 volts is reached. An ampere-hour meter measures battery state of charge and provides control to assure proper charge. The math model for the electrical power subsystem is:

$$R_{power} = R_{array} \times R_{battery} \times R_{s.\ regulator} \times R_{s.\ resistors} \times R_{distribution} \times R_{switches}$$

where:

$R_i = e^{-\lambda_i t}$ for the shunt regulator, amp-hour meter, distribution and switches

λ_i = Effective failure rate of item i during the mission

t = Mission time

R_{array} = Predicted array reliability. The array has considerable cell redundancy

$R_{battery}$ = Predicted battery reliability. Cell temperature, number of charge/discharge cycles and depth of discharge has been based on S3 power analysis.

The shunt resistor panel assembly consists of two sets of parallel resistors and requires two resistor failures for the subsystem failure. The model is:

$R_{s.\ resistors} = (1-Q_R^2)^2$, where $Q_R = 1-e^{-\lambda_R t}$, and λ_R being the resistor failure rate

Figure 3.8.5 presents the reliability of the power subsystem equipment. As the subsystem's equipment is all required for mission success (in series), a block diagram has not been provided.

3.8.4 ORDNANCE (AND SPIN MOTORS) SUBSYSTEM

For the spin stabilized BSRM spacecraft, this subsystem consists of two solid propellant spin motors and the ordnance required to actuate five pin pullers and two clamp release separation nuts. The math model is as follows:

$$R_0 = R^2_{spin\ motor} \times \left[1-(1-R_{pc})^2\right]^6$$

where R_{pc} is the reliability of a pressure cartridge. There are a total of five redundant pairs of pressure cartridges for pin puller actuations (two boom deployment pin pullers, two solar array pin pullers, and a yo-yo release) and two additional cartridges for actuation of two (angle cartridge) redundant separation nuts to release the satellite separation V-band.

$R_{spin\ motor}$ = 0.9995 based upon an S3 spin motor analysis by the manufacturer (Atlantic Research). It is noted that the spin motors have redundant

Item		$\lambda x10^6$	t(hrs)	D	N	R	Data source
1.	Array	N/A	4320	1	1	0.9999	[2]
2.	Battery	N/A	4320	N/A	1	0.9967	[2]
3.	Shunt regulator	1.26	4320	1	1	0.9945	[2]
4.	Shunt resistors	.02	4320	1	1	0.9999	[3]
5.	Amp-hr meter	4.45	4320	1	1	0.9809	[2]
6.	Distribution	1.30	4320	1	1	0.9944	[3]
7.	Switches	.10	4320	1	[1]	0.9999	[3]

$$R_{POWER} = \prod_{1}^{7} R_n = 0.9965 \text{ (Spin stabilized \& 3-axis stabilized spacecrafts 6 month mission)}$$

$$= 0.9341 \text{ (3-Axis stabilized spacecraft, 12 month mission)}$$

[1] 4 Separation switches, quad redundant

[2] Analysis data sources are referenced in section 3.8.7

[3] Analysis by Boeing based on S3 configuration with failure rates from MIL-HDBK-217A data

Figure 3.8-5. Power Subsystem Reliability initiators.

R_{pc} = 0.9998 at 95% confidence, based on manufacturer (Hi-Shear) analysis of pressure cartridge test firing data (15152 firings with no failures).

$R_0 = 0.9995^2 \times \left[1-(1-0.9998)^2\right]^6 = 0.9990$ (spin stabilized spacecraft)

The subsystem for the three axis stabilized spacecraft contains no spin motors or ordnance for boom deployment. Its reliability is therefore:

$$R_0 = \left[1-(1-R_{pc})^2\right]^4 = 0.9999 \text{ (3-axis stabilized spacecraft)}$$

The six-month and one year mission reliabilities are identical as the ordnance has only one actuation at the start of the mission.

3.8.5 THERMAL CONTROL SUBSYSTEM

The thermal control subsystem consists of thermal coatings, insulation blankets, transmitter heat sink and thermal louvers to provide thermal control for the payload and satellite components. All the components except the louvers are passive, the reliability model relates primarily with the louvers, while for the other elements of the thermal control system, the reliability estimate is an allowance based on adequate design margins and previous tests and experience.

A thermal control system failure is defined as a condition which will cause the equipment temperature to be outside of its qualification requirements. The most critical condition contributing to system failure is for the louver blades to fail in the closed position. Based on S3 thermal analysis, for this preliminary evaluation it is considered that adequate temperature control can be maintained in the BSRM equipment bay with two louver blades failed in the closed position. Each of three louver assemblies have four independently actuated louver blades, so the louver system is considered to consist of 12 louver blades. The math model is:

$$R_{\text{thermal control}} = (R_{LB})^{12} + 12\, R_{LB}^{11}(1-R_{1b}) + 66\, R_{LB}^{10}(1-R_{LB})^2 \quad R_{PS}$$

where the louver blade reliability $R_{LB} = e^{-\lambda n}$

$\lambda = 2 \times 10^{-6}$ per cycle * failure rate

n = 1800 cycles/mission

$R_{LB} = e^{-(2\times10^{-6})1800} = 0.9964$

Reliability of passive elements = R_{PS} = 0.9999 (allowance)

$R_{\text{thermal control}}$ = 0.9999 x 0.9999 = 0.9998 for the spin stabilized spacecraft, with the six-month mission.

* Based on test data - 51 louvers cycled 10.000 each with no failures, some testing (up to 360.000 cycles) in vacuum conditions. Reference: Paper presented at ASME AICHE Heat Transfer Conference, Aug. 1963. STP P72-1 Reliability Analysis Report D2-116195-1.

The thermal subsystem reliability for the 3-axis stabilized spacecraft is identical, and for the one year mission, it has been conservatively estimated at $0.9998^2 = 0.9996$.

3.8.6 STRUCTURES/MECHANISMS

For the spin stabilized BSRM spacecraft, the structures/mechanism subsystem, for reliability analysis purposes, consists of the following items: satellite structure, deployable antenna and magnetometer booms (2), deployable solar array structure (2), V-band separation mechanism, and yo-yo despin. The subsystem math model is as follows:

$$R_{S/M} = R_{structure} \times R^2_{boom} \times R^2_{SA\ structure} \times R_{separation} \times R_{yoyo}$$

There is no submodel required for the $R_{structure}$, since all items involved are also structure and are not adaptable to a conventional reliability analysis. The reliability of the satellite structure is assured through the establishment of positive safety margins in the structural analysis and by the successful completion of the test program. The prediction (0.9999) is an allowance for the lower limit of structural reliability, based on historical data of similar structures.

The boom deployment math model is as follows:

$$R_{boom} = (R_{spring} \times R_{bearing} \times R_{pin\ puller} \times R_{boom\ struct}) \text{ (for 2 booms)}$$

where the reliability of the spring, bearing and boom structure all approach one, for one actuation. As a lower limit allowance, a combined value of 0.9999 is used in the analysis. The pin puller reliability is 0.9998 based on manufacturer's analysis (Hi-Shear) and evaluation of test data of similar equipment.

$$R_{boom} = 0.9999 \times 0.9998 = 0.9997 \text{ for each boom}$$

The reliability of the deployable solar array structure is estimated to be the same as for the boom; the release system is identical and the deployment system requires one actuation for mission success.

The satellite separation consists of a V-band with two release clamps, either of which will release the satellite; the clamp is opened by a separation nut with a single ordnance cartridge. Four compression springs separate the satellite from the booster upon V-band release.

$$R_{separation} = R_{V\text{-}band} \times 1- (1-R_{sep.\ nut})^2 \times R^4_{spring} = 0.9998$$

$R_{V\text{-}band} = 0.9999$ (structural allowance)

$R_{sep.\ nut} = 0.9998$ based on supplier (Hi-Shear) test data: 3605 sep. nut actuations with no failures (50% confidence)

$R_{spring} = 0.99999+$, based on FARADA data

The yo yo despin can be considered structure with a pin puller release.

$$R_{yoyo} = R_{yoyo\ struct} \times R_{pin\ puller} = 0.9999 \times 0.9998 = 0.9997$$

Based on the above, the reliability prediction of the structures/mechanisms subsystem is:

$$R_{S/M} = 0.9999 \times 0.9997^2 \times 0.9997^2 \times 0.9998 \times 0.9997 = 0.9982$$ for the spin stabilized spacecraft.

The three-axis stabilized spacecraft does not have deployable booms, and consequently $R_{S/M}$ is slightly higher:

$$R_{S/M} = 0.9999 \times 0.9997^2 \times 0.9998 \times 0.9997 = 0.9988$$ for the 6-month mission

The reliability for the one year mission is considered identical, as the equipment malfunctions are not time dependent.

3.8.7 ANALYSIS REFERENCES

Attitude Control and Determination Subsystem

ELECTROMAGNETS - BOEING analysis for S3, "Reliability Maintainability allocation, Assessment and Analysis Report", document D233-10015-1 dated 12/5/72. Failure rate is based on FARADA inductor data.

TIMER - CELESCO analysis dated 12/1972 for the 4051 high reliability timer, based primarily on MIL-HDBK-217 part failure rates. Failure rate for ferrite cores was obtained from Martin's T-70-48891-007 "Handbook of Piece Part Failure Rates (Long Life Space Vehicle Investigation)", dated 6/22/1970.

MAGNETOMETER - SCHONSTEDT analysis SAM-63C-1, dated 12/12/72. Part Failure rates are based on RCA's PPL 1971202, factored for power, temperature and voltage operating levels.

SUN SENSOR - ADCOLE analysis, report # QD10053, dated 2/23/73, based on part failure rates from RCA's PPL 1971202, derated for max. operating stress levels of temperature, voltage and power. Failure rate for two axis sun sensor, in the 3-axis stabilized BSRM configuration was estimated to be twice that of the single axis sun sensor, by Boeing.

EARTH SENSOR - BARNES analysis for S3, report # BEC REL 2585-02, dated 2/8/73, based on part failure rates from Hughes Aircraft Co. Document PEH 06-305, "Approved Failure Rates and Parts Derating Policy for Subcontractors", dated 8/31/71, and MIL-HDBK-217A (Thermistors Only). Failure rates were evaluated by Boeing and found acceptable.

SCAN WHEEL - ITHACO analysis, report #90400, dated 4/26/74. Part failure rates are based on MIL-HDBK-217A and for I.C.'s:National Semiconductor Report, dated May 1970.

TT&C Subsystem

TAPE RECORDER - ODETICS analysis, report #9112003, dated 4/10/73, based on MIL-HDBK-217A failure rates. Analysis was modified by Boeing for S3 operational duty cycles; BSRM application is considered similar to that for S3 program.

TRANSMITTER - CONIC analysis by similarity to Teledyne TF-2400 transmitter. Analysis report #2007021, dated 2/14/73, based on part failure rates from RADC Handbook TR-67-108, derated for operating stress levels.

PROCESSOR - TELEDYNE analysis report #2006870, dated March 1973, based on part failure rates from Lockheed's reliability data. Evaluation by Boeing indicates they are comparable to other acceptable sources.

COMMAND DECODER - CONIC analysis report #11959, dated 3/15/73, based on failure rate data from MIL-HDBK-217A. JAN-TX and MIL-ER parts failure rates used are .1 of those in the handbook, to correct for extra screening and burn-in.

DIPLEXER - WAVECOM analysis #93-163-RA dated 1/8/73, based on MIL-HDBK-217A and Wavecom experience failure rate data.

RF SWITCH - TRANSCO analysis #Q91C-2114, dated 10/14/69, based on life test data.

RECEIVER - CINN. ELECTRONICS; the reliability has been based on the analysis of a Motorola unit (Report #01-P05555D, dated 5/19/72) used on the S3 program. It is expected that the two units have equivalent reliability. The analysis, modified by Boeing for the S3 application, is based primarily on MIL-HDBK-217A part failure rates, although some have been changed, based on Motorola test experience, per report (special memo) #188 "Part Failure Rates", dated 10/31/69.

RELAY BOX - BOEING analysis for S3 program, based on failure rate data from MIL-HDBK-217A. Relay cycles, for S3 satellites, are considered an upper limit for BSRM missions.

Power Subsystem

SHUNT REGULATOR - GULTON analysis, report #2752, dated 12/13/72. Part failure rates are based on MIL-HDBK-217A and MIL-HDBK-1470. Rates for semiconductors have been reduced to 0.1 of those in handbook due to use of JAN-TX parts.

BATTERIES - EAGLE PICHER analysis for SAR-8088 battery for S3 program, dated 1973. Cell failure rates are based on Eagle Picher Ni-Cd cell test data, considering number of cycles, depth of discharge and temperature for S3 program. Similar operational conditions are expected for the BSRM 6-month mission. The battery can support the 12-month mission of the 3-axis stabilized spacecraft with adequate life (cycle) margin.

AMP-HOUR METER - GULTON analysis, report #2759 dated 1/19/74. Part failure rates are based on MIL-HDBK-217A. Failure rates of semiconductors changed to 0.1 of values in handbook, due to use of JAN-TX parts (Ref. RADC-TR-67-108).

SOLAR PANELS - SPECTROLAB analysis, report #021219, dated 1/16/73. Part failure rates are based on Sepctrolab experience. High reliability is due to extensive internal redundancy.

Ordnance and Associated Equipment

SPIN MOTOR - ATLANTIC RESEARCH CORP. analysis dated 3/73 for the S3 program, based on ARC spin motor firing test data.

PC CARTRIDGE - HI SHEAR analysis based on firing test data over a 12 year period (1961 to 1973) 15152 units tested with no failures.

SEPARATION NUT - HI SHEAR analysis based on test data over a 12 year period (1961 to 1973) 3605 units tested with no failures.

PIN PULLER - Based on Hi Shear test data for similar equipment over a 12 year period (1961 to 1973). 192 pin pullers tested by Hi Shear with no failures are not sufficient to demonstrate high reliability with a reasonable level of confidence, i.e., 50% or higher.

3.9 ENVIRONMENTS

THE QUALIFICATION OF THE EXISTING FLIGHT-PROVEN EQUIPMENT ITEMS WAS REVIEWED WITH RESPECT TO THE SCOUT LAUNCH ENVIRONMENT. ACOUSTIC, SHOCK AND RANDOM VIBRATION REQUIREMENTS ARE MET. THE HIGHER SCOUT ACCELERATION EXCEEDS COMPONENT QUAL LEVELS BUT IS CONSIDERED A NON-CRITICAL ENVIRONMENT.

Maximum boost accelerations and acoustic generated random vibration are the critical launch peculiar environments affecting BSRM structural design, equipment qualification, and spacecraft development. Current practice requires all equipment components to be qualified by testing to these environments along all three axes. The Scout boost trajectory is characterized by relatively high boost accelerations resulting from motor ignition transients and by relatively low acoustic levels because of its low first stage thrust. Most off-the-shelf equipment components are qualified to vibration levels well in excess of Scout requirements. Because acceleration is considered a low risk environment, it is not recommended that additional qualification be accomplished.

A maximum boost acceleration of 16.7 g's is predicted for first stage ignition. The corresponding qualification test level for components, with a 1.5 factor of safety, is 25 g's. This acceleration compares with the following qualification levels specified for past programs:

Burner II	19.3 g's
STP P72-1	12.0 g's
Burner IIA	16.0 g's
Small Satellite (S3)	16.0 g's

The most severe loads to which components are designed are generally induced by random vibration. The fact that some existing components are not qualified to the Scout g-level is therefore not considered a problem. Specific components could easily be subjected to acceleration testing if required, but this is not recommended at this time.

Equipment random vibration qualification test requirements derived for Scout are defined in Figure 3.9-1 and compared with levels specified for S3, Burner IIA, and STP P72-1. Qualification test levels were derived by adjusting response levels measured during S3 and P72-1 ground tests by the ratio of acoustic levels. The maximum expected spacecraft acoustic environment is compared to the S3 and STP P72-1 ground test levels in Figure 3.9-2. On the basis of this comparison, all equipment qualified for these previous missions are qualified by similarity for vibration.

BSRM separation shock environments are similar to those encountered on previous Boeing developed upper stages and spacecraft and present a low technical risk. Qualification for shock by testing at the component level was not performed on past Boeing programs and is not recommended for the BSRM spacecraft.

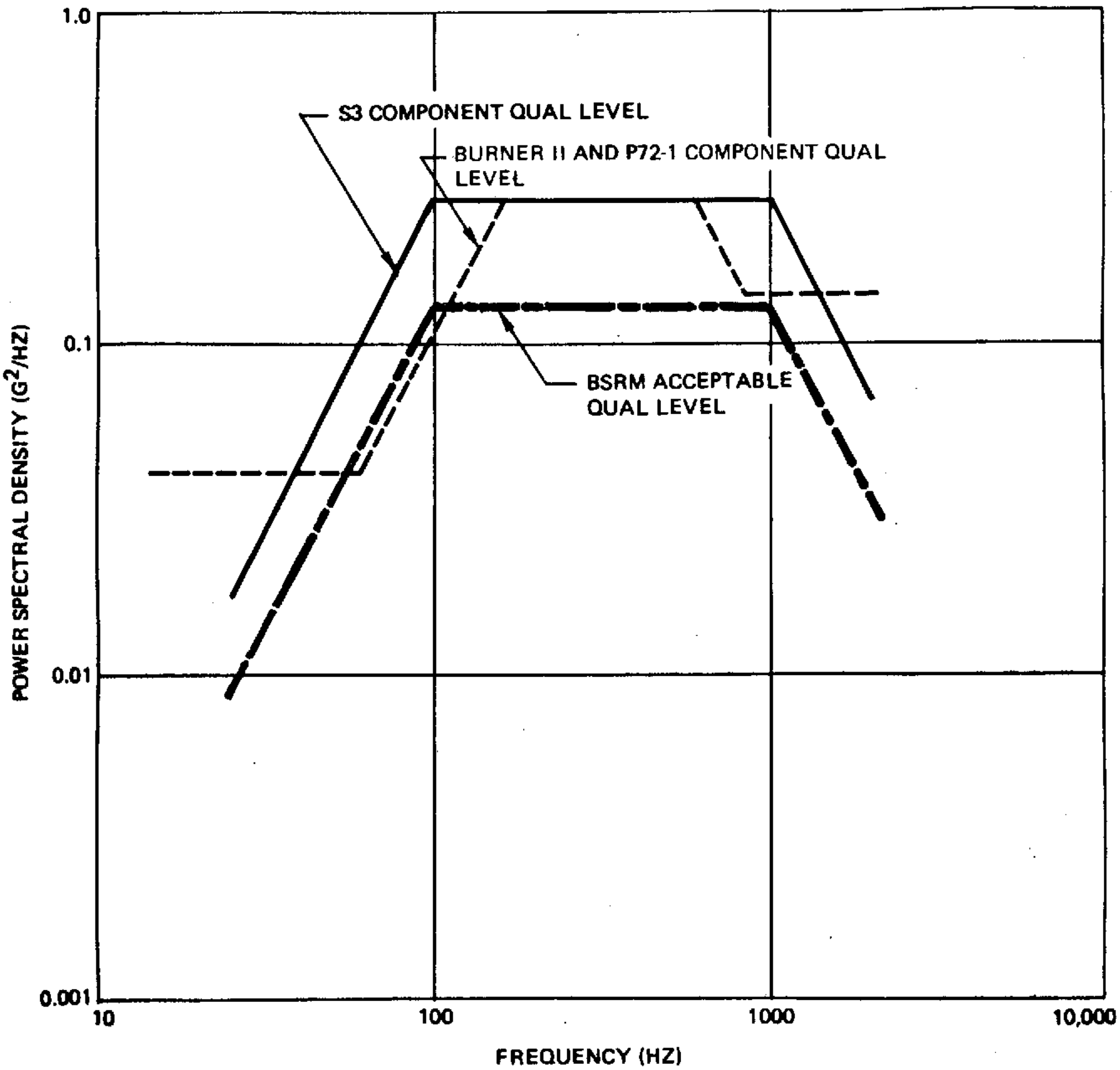


Figure 3.9.1. Random Vibration Qualification Test Levels

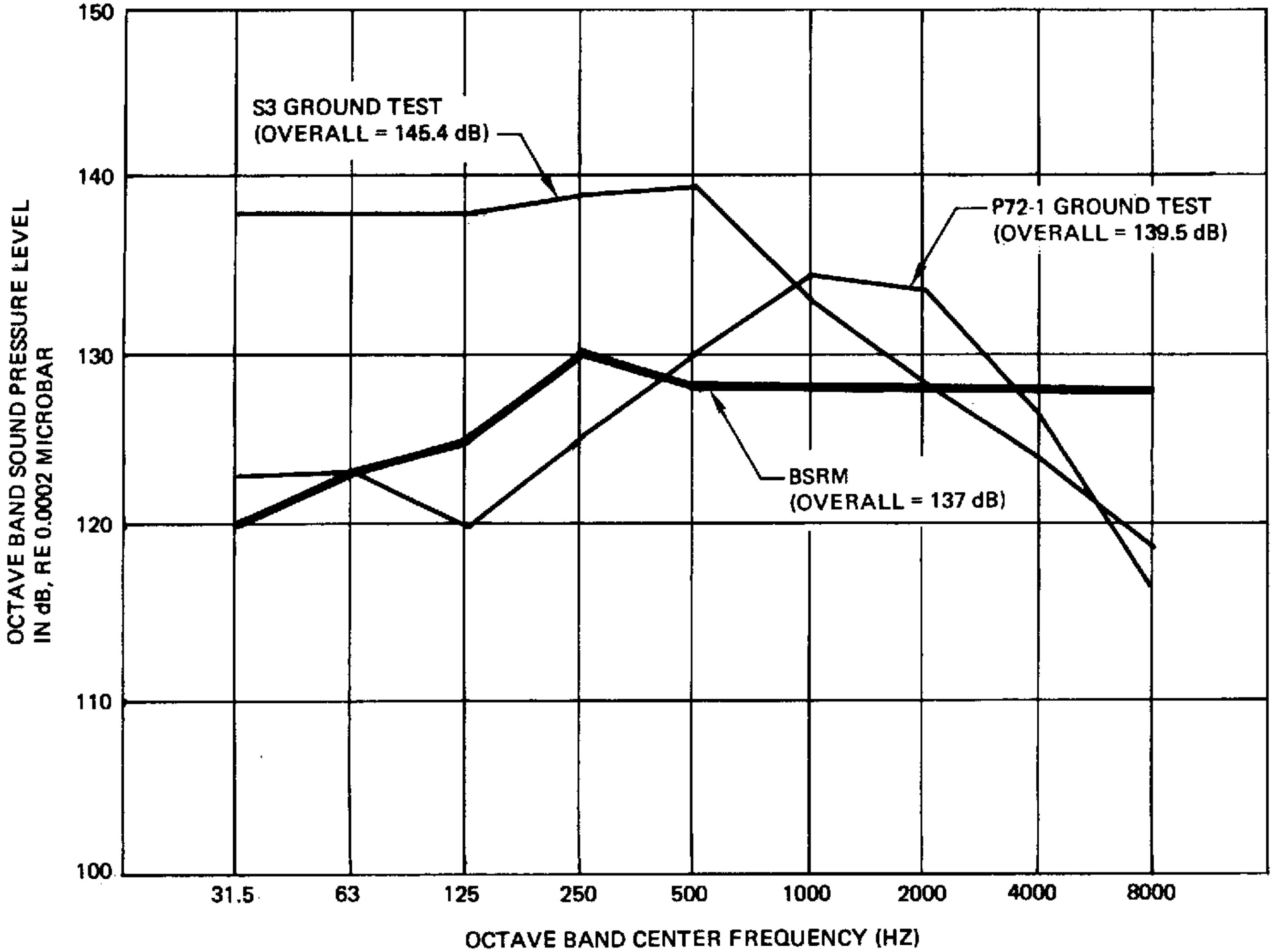


Figure 3.9-2. Acoustic Environment Comparison

4.0 SPACECRAFT INTEGRATION

4.0 SPACECRAFT INTEGRATION

BOEING HAS ESTABLISHED CLOSE WORKING RELATIONSHIPS WITH VARIOUS LAUNCH VEHICLE CONTRACTORS AND GOVERNMENT AGENCIES THROUGH EXTENSIVE EXPERIENCE ON NASA SPACECRAFT PROGRAMS, USAF SPACE TEST PROGRAM (STP) MISSIONS AND OTHER CLASSIFIED PROGRAMS. THESE OPEN CHANNELS FACILITATE THE COORDINATION AND IMPLEMENTATION OF THE PAYLOAD, OPERATION, AND MISSION RELATED REQUIREMENTS OF EXPERIMENTERS AND AGENCIES AT ALL PROGRAM LEVELS.

Boeing has integrated spacecraft with a number of payload contractors and agencies including the Aerospace Corporation Space Physics Laboratory, Cambridge Research Laboratory, Lockheed Palo Alto Research Laboratory, Naval Research Laboratory and various universities. This experience has provided Boeing with a detailed familiarity and appreciation of how to solve a wide range of problems in meeting experimenter and payload mission requirements.

4.1 SYSTEM INTEGRATION

4.1.1 INTERFACE DOCUMENTATION

Immediately after program go-ahead, Boeing will meet with experimenters to refine and update interface requirements and data. This will be the first Tecynical Interface Working Group (TIWG) meeting. An early output of the TIWG meetings will be formal Interface Control Documents (ICDs) for each BSRM approved by all affected contractors. Each ICD will consist of a basic document body containing mission and satellite data common to each experiment including orbit parameters, design environment, telemetry formatting, power quality, grounding requirements, etc. An annex to the basic ICD will be prepared for each experiment defining its detailed interface requirements down to connector pin assignments.

Proposed changes to the ICDs will be submitted by the originator to NASA and all affected contractors and agencies for evaluation of cost, schedule, and program impact and approved or rejected at the next TIWG meeting. Formal Interface Revision Notices (IRNs) will document all approved interface changes.

4.1.2 INTEGRATION COORDINATION AND DOCUMENTATION

The requirements contained in the basic ICD and ICD annexes will be incorporated into the satellite documentation at all levels. Integration documentation will include: block diagrams and schematics, component specifications, installation drawings, test plans and procedures, and design analysis documents. Since the BSRM is operational hardware, every effort will be made to minimize changes to the existing hardware, documentation and software.

Boeing will coordinate with each payload contractor and agency the detailed test requirements and procedures for integration and checkout of the experiment with the satellite system. Boeing will then develop integrated test procedures including go/no-go criteria. Boeing will also provide facilities and utilities to support inplant payload personnel and test equipment. The Boeing automated Mobile Test Lab will be used for efficient integration testing of the spacecraft and experiments.

4.1.3 PROGRAM COORDINATION AND DOCUMENTATION

Boeing will coordinate all operational, hardware and software interfaces of the BSRM including its payloads with the VAFB range and Range Safety offices, the STDN network, the launch vehicle, and, if necessary, the Atomic Energy Commission if radioactive materials are to be used with flight or test hardware.

Interfaces with and support required from the VAFB range are documented in the Program Requirements Document (PRD). The PRD is prepared by the Test Wing using inputs supplied by the booster contractor and Boeing. Detailed coordination and scheduling of range operations is accomplished at Launch Test Working Group (LTWG) meetings. These meetings are chaired by the Test Wing and attended by range agencies and the booster and integrating contractors. Attendance at LTWG meetings by experimenter personnel is required starting about six weeks before launch.

Interfaces with and support required from the NASA Space Tracking and Data Network is documented in the Orbital Requirements Document (ORD). The ORD will be prepared by Boeing using inputs supplied by the experimenters. Detailed coordination and scheduling of STDN operations is accomplished at Orbital Test Working Group (OTWG) meetings. Attendance of these meetings by experimenters is required.

Interfaces with the booster and launch site are normally defined in a payload/booster ICD prepared by the booster contractor. Detailed coordination of operations and schedules is accomplished at TIWG meetings.

4.2 SYSTEM TEST

The System Test Program for the BSRM was developed from extensive experience gained during similar test programs for other systems. The test program as shown in Figure 4.2-1 ensures a thoroughly tested and reliable functional satellite. The test program is discussed in detail in section 4.0 of D180-18450-3 and features the following:

- The satellite systems are functionally complete and checked out, including simulated payload interfaces, prior to mating flight experiments.

- Complete subsystem tests assure proper operation under simulated flight conditions.

- Complete integration tests with all flight payloads installed verify end-to-end operation for maximum assurance of satisfactory operation in orbit.

- Use of the Boeing-owned Mobile Test Lab with telemetry ground station receiving and processing equipment accelerates the test schedule.

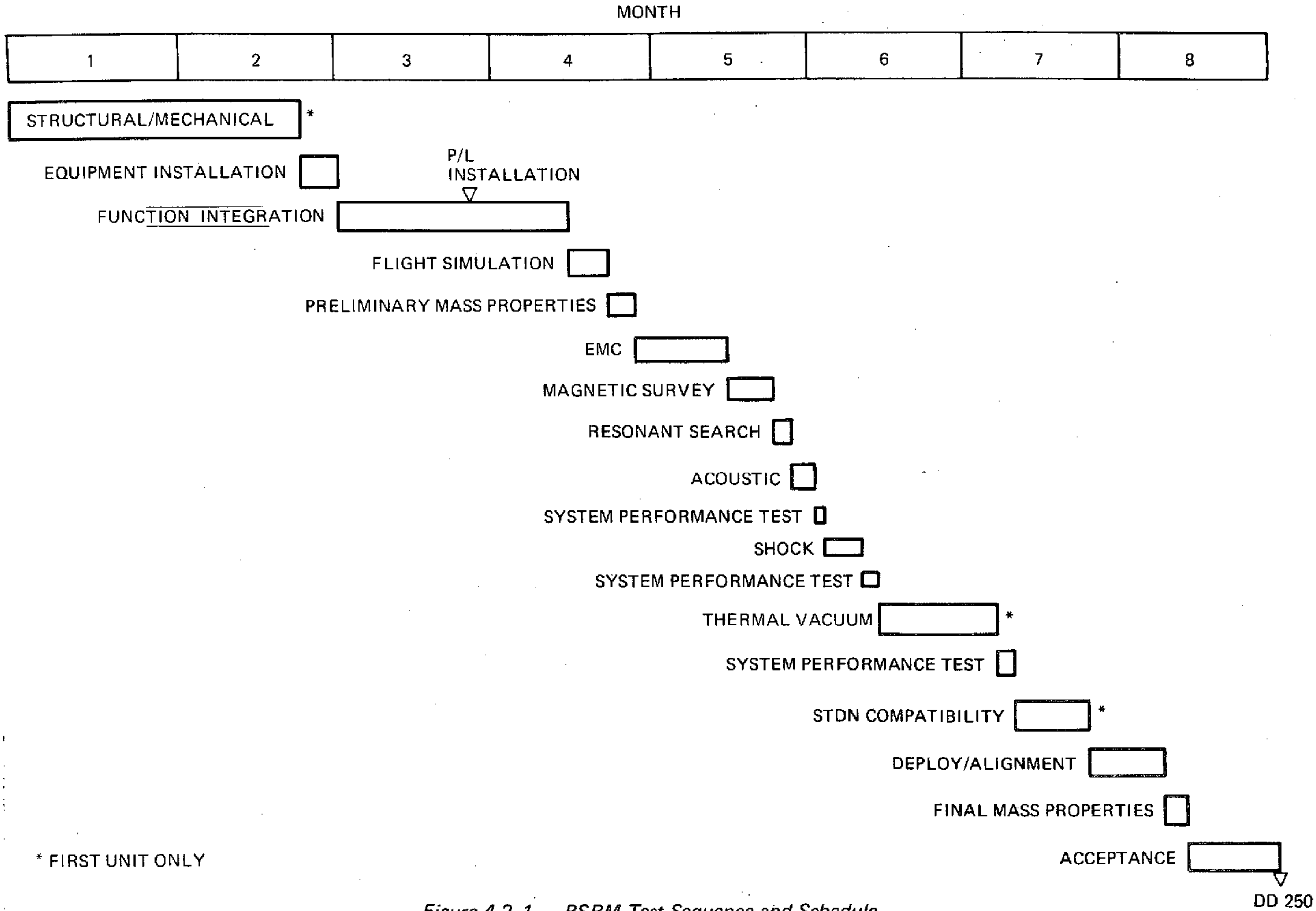


Figure 4.2-1. BSRM Test Sequence and Schedule

- Test procedures, techniques and software developed on existing spacecraft programs are available to support BSRM.

The Mobile Test Lab is a Boeing-owned command and telemetry ground station used to verify satellite performance during system testing. The lab contains the equipment required to:

- Receive and evaluate satellite telemetry data.
- Record test data.
- Transmit satellite commands.
- Identify out-of-tolerance measurements.
- Format and display selected test data.
- Service operator requests for data processing, command generation and transmission, display formatting and software operation.
- Retrieve, display and print selected data.

The equipment is contained in an 8 feet high by 8 feet wide by 34 feet long module mounted on a four-wheel trailer. The wheels are removable for air or road mobility. Figure 4.2-2 presents a block diagram. The capabilities and features of this Boeing capital equipment item are fully described in document D180-18431-2.

4.2.1 FUNCTIONAL INTEGRATION WITHOUT PAYLOADS

Starting with a complete satellite structure and wiring assembly, the satellite systems are installed and integrated as follows:

Power System - The ability of the power system to operate properly under solar power and battery power, including load sharing, battery charging, bus regulation, shunt regulation and heat rejection is verified.

Command System - All command functions are verified, including real-time and stored commands. A final open loop command test verifies the integrity of the satellite antenna and coaxial cable.

Data System - Analog and discrete telemetry channels are simulated individually and monitored through a hardline connection from the data transmitter to the Mobile Test Lab for verification of correct channel assignment and data value. Simulated payload outputs are provided to the data system PCM processor at the payload/satellite interface connector.

Telemetry Calibration - Calibrated voltages are inserted into the satellite TT&C system at the interface connectors from satellite sensors or payloads where calibrated analog telemetry outputs are required.

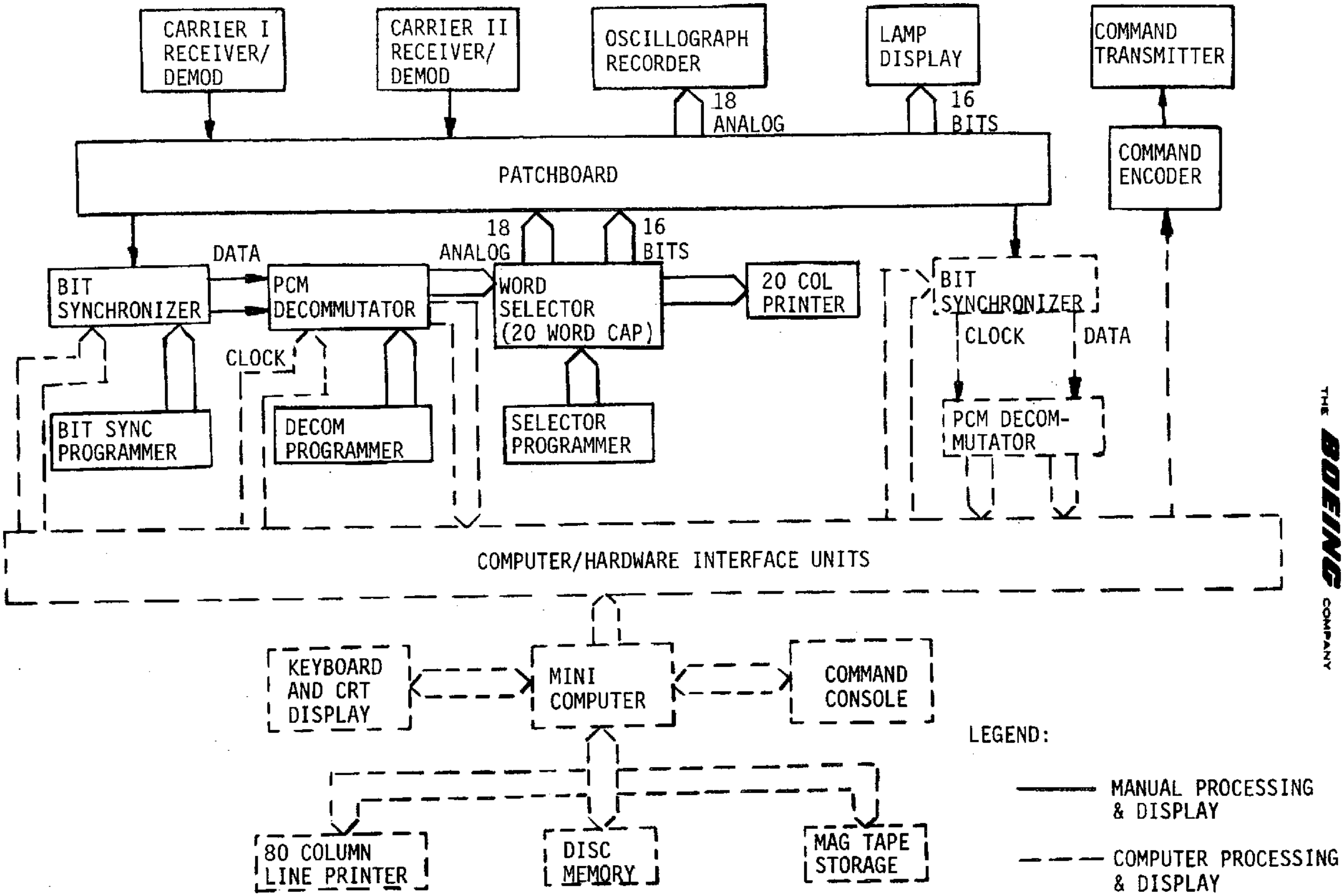


Figure 4.2-2. Boeing Mobile Test Lab

Attitude Control System - Real-time and stored commands are used to command the various ACS modes, and the resultant control action monitored as magnetic field intensity variation at the magnetic torquing coils. Spin maintenance and control is verified by stimulating the magnetometers with a simulated rotating magnetic field and commanding a spin rate change from the command system. A horizon sensor test set supplies time varying infrared stimulation to the horizon sensor to simulate earth crossings at the satellite spin rate. Correct outputs from the horizon sensors are verified through the satellite telemetry system.

System Performance Test (SPT) - This test verifies readiness of the satellite for installation and functional integration of the payloads.

4.2.2 FUNCTIONAL INTEGRATION WITH PAYLOADS

After all satellite systems have been completely checked out with simulated payloads, and the telemetry system calibrated, the flight payloads are installed and connected. All payload command functions are initiated, and payload status and data channels (both real time and recorded) are transmitted to the Mobile Test Lab through the telemetry system. The payload data is processed for analysis by payload contractor personnel.

Flight Simulation. A satellite and payload flight sequence is performed starting at satellite separation from the booster and ending when all selected payload functions have been performed including timed events, real-time and stored commands, and real-time and recorded status and mission data.

Preliminary Mass Properties. The BSRM is weighed and its CG location recorded to verify weight and balance analyses.

EMI/EMC Tests. System-level electromagnetic compatibility (EMC) tests are conducted to verify EMC between all subsystems by demonstrating an electromagnetic interference (EMI) safety margin during all modes of system operation. The EMC tests will use one or combinations of three general approaches:

- Augmentation - Inject interference at critical system points at a 6 dB higher level than normally exists, while monitoring subsystems for improper operations.

- Comparison - Compare existing system interference levels to susceptibility of critical system/subsystems for demonstration of a 6 dB margin.

- Sensitization - Sensitize the system/subsystem to render it 6 dB more susceptible to interference and monitor for improper response while operating system through normal flight modes and sequences.

Magnetic Survey. The magnetic moment of the satellite is determined by measuring interaction between its magnetic field and another calibrated magnetic field. The satellite is mounted on a torque table and placed in a uniform calibrated magnetic field produced by Helmholtz coils, with a diameter of at least twice the maximum satellite dimension. The magnetic field within the field-of-view of the experiments will be surveyed with a Hall effect magnetometer. Adjustments of the residual moment will be made as required to meet magnetic field constraints of the critical payloads.

Mechanical and Acoustic Vibration. The satellite, complete with payloads, will be subjected to flight level mechanical and acoustic vibration environments.

Deployment/Separation Shock Tests. Satellite deployment and separation tests are performed on each deliverable satellite to verify functional characteristics and to uncover possible workmanship errors and faulty parts.

Thermal Vacuum. The first BSRM will be subjected to thermal vacuum tests with solar simulation to verify operation under nominal, worst cold and worst hot orbital conditions.

Range Compatibility Testing. During satellite functional testing, real-time and recorded satellite and payload status and data telemetry will be recorded on magnetic tape. This tape will be used for an early compatibility test with the launch facility. In this test the recorded data will be transmitted from the launch pad to the tracking station and after formatting, will be used for processing and data verification.

Deployment/Alignment. All deployable items and critically aligned components and experiments are inspected to verify proper alignments have been maintained.

Weight and Balance. Weight and balance testing will be performed to balance the satellite accurately in the launch configuration to verify that design weight, center of gravity, and inertia requirements are met. The satellite will be statically and dynamically balanced and weighed and its moments of inertia measured.

Acceptance Test. The satellite acceptance test verifies that all performance objectives are satisfied prior to satellite delivery. Before starting final acceptance testing, all discrepancies occurring during previous testing are resolved and the satellite is in deliverable configuration with flight battery installed and charged.

4.3 SYSTEM OPERATION

BSRM system operation provides the experimenter with proven and versatile experiment operation. The following are the features that provide the smooth flow and versatility.

- o Tracking, data retrieval, and data processing by STDN.
- o Technical support by the BSRM design and test engineers.
- o Real time mode selections by ground command.
- o Real time or delayed execution of the initiation of satellite operations.

4.3.1 BOOST AND INJECTION OPERATIONS

Each BSRM is placed in orbit by the Scout launch vehicle. During separation of the satellite, the timer is actuated to initiate the BSRM sequence of events. The sequence of events after ejection is as follows:

- o Fire spin-up motors (for spin stabilized BSRM).
- o Deploy satellite telemetry antennas.
- o Deploy magnetometer boom.
- o Begin quarter orbit torquing to precess the satellite spin-axis to the on-orbit orientation.
- o Switch timer from the ejection mode to the orbital mode for attitude maintenance and payload control.

These events will be completed in less than three hours after separation from the launch vehicle.

4.3.2 ORBITAL OPERATIONS (SPIN STABILIZED)

Quarter orbit torquing continues for up to forty-eight hours to precess the satellite spin axis. During this period of time, ground commands are sent to the satellite to adjust the timer drive frequency to synchronize timer cycling with the orbital period. Housekeeping functions are monitored to verify proper satellite operation. Ground commands are sent, at the completion of spin axis precession, to release pyro-operated covers and other payload deployment and enabling functions.

Orbital operation of experiments is accomplished by ground command and a pre-programmer timer/ground programmed relay matrix. Real time operation of an experiment is accomplished by first sending commands to enable the experiment(s) desired and then sending a command switching the experiment power bus from pre-programmed operation to the ON configuration.

Pre-programmed operation, using the timer, provides great flexibility in experiment operations. Timer output is programmed prior to each flight for twenty one relay latching pulses per orbit; 16 of the pulses occurring at each 22.5 degrees of orbit, and 8 pulses are at the discretion of the experimenters. (Zero degrees in the orbit is referenced to any predetermined point in the orbit.) Each of the latching pulses may be used to either activate or deactivate the experiment bus during the orbit. Ground command is available to enable the use of the pulses. Ground command also is used to activate the timer during one or more of six orbits following a ground contact.

4.3.3 GROUND OPERATION

Satellite ground operations are performed by the NASA Space Tracking and Data Network (STDN). This net provides ground command of the satellite and accumulation of experiment, housekeeping, attitude and ephemeris data. Satellite operations will be performed in accordance with the Orbital Requirements Document (ORD), described in Section 4.1.3. The ORD will be updated during the mission to provide for changes in operating conditions or the desired mode of operation. Data transmitted from the satellite will be gathered by the remote tracking stations throughout the world. Real time data and commands will be transmitted to and from the satellite via land lines. Recorded data, transmitted from the satellite, is shipped from the remote tracking stations to the Goddard Space Flight Center.

Attitude and ephemeris are determined at Goddard. These data are then merged with the recorded experiment data, in a format compatible with the experimenters computer, to provide coordinated data. Raw data, including housekeeping and attitude, is available upon request.

4.4 EXPERIMENT REQUIREMENTS

This section describes typical requirements imposed on experimenters. These requirements are necessary to provide safe and reliable operation of the launch vehicle and satellite and to meet launch base criteria.

4.4.1 ENVIRONMENTAL REQUIREMENTS

4.4.1.1 Boost Environment. The experiment shall be capable of withstanding the following boost environment:

a. Random Vibration

10-100 Hz	Increase @ 6 dB/octave
100-1000 Hz	Flat @ 0.14 G^2/Hz
1000-2000 Hz	Decrease @ 6 dB/octave

Overall level of 14.2 Grms

b. Acceleration: 1) 16g axial combined with 4g lateral.
2) 22g axial combined with 2g latera.

4.4.1.2 Orbital. The experiment shall be capable of continuous operation for one year or more in standby mode with intermittent operation in normal operating mode exposed to space vacuum, with the experiment temperature in the range of +10°C to +35°C for non-deployed items.

4.4.1.3 Transportation and Handling. When packaged or otherwise prepared for shipment, the experiment shall be capable of withstanding or shall be protected for the following environment.

a. Altitude: 0 to 40,000 feet

b. Temperature: -45°F to +120°F with thermal gradients of 33°F per minute during air transportation.

4.4.1.4 Qualification. The experiment shall be subjected to a qualification program consisting of the following as a minimum:

- o Random Vibration — 2 times G^2/Hz
- o Acceleration — 1.5 times maximum flight levels
- o Thermal Vacuum — Operate -5°C to +50°C (soak and cycle)
  Survive -15°C to +60°C
- o EMI — Applicable sections of MIL-STD-461A

4.4.2 EMI REQUIREMENTS

To demonstrate an electromagnetic compatibility margin of 6 dB between the satellite equipment and the payloads, the electromagnetic generation and susceptibility characteristics of each payload package are to be provided by the experimenter. The data should be determined over the frequency range and using the test methods as shown below:

TEST METHOD	TEST DESCRIPTION	FREQUENCY RANGE
CE-01	Power Line Conducted Emissions	30 Hz to 20 KHz
CE-02	Control & Signal Lead Conducted Emissions	30 Hz to 20 KHz
CE-03	Power Line Conducted Emissions	0.02 to 50 MHz
CE-04	Control & Signal Lead Conducted Emissions	0.02 to 50 MHz
CS-01	Power Line Conducted Susceptibility	0.03 to 50 KHz
CS-02	Power Line Conducted Susceptibility	0.05 to 400 MHz
CS-06	Power Line Conducted Spike Susceptibility	(Characteristics TBD)*
RE-02	Radiated Emissions	14 KHz to 10 GHz
RE-04	Radiated Emissions, Magnetic Field	20 Hz to 50 KHz
RS-02	Magnetic Induction Field	(Characteristics TBD)*
RS-03	Electric Field Radiated Susceptibility	14 KHz to 10 GHz

4.4.3 GROUNDING AND BONDING REQUIREMENT

4.4.3.1 Grounding. The primary power return, signal return, and command return shall be isolated from the case and from each other by at least one megohm. Radio frequency coaxial returns may be grounded to the case. Exceptions to this requirement are allowed only as specifically identified in the ICD Annexes.

*Mission and payload peculiar

4.4.3.2 Bonding. All electrical/electronic equipment shall be installed and bonded in accordance with MIL-B-5087B. Bonding shall be accomplished by bare, clean metal-to-metal contact of all mating surfaces. However, suitable approved conductive finishes may be used on equipment bonding surfaces to achieve the low impedance path. Dissimilar metals shall be properly treated to avoid the long-term effects of galvanic action and corrosion. Mating surfaces shall be compatible with aluminum.

4.4.4 MAGNETIC FIELD STRENGTH CHARACTERISTICS

Each experiment package shall be tested for hard perm, soft perm, and active magnetic dipole moment along each of three orthogonal axes. Results of tests shall be submitted to Boeing in support of experiment-vehicle integration. The hard perm magnetic dipole moment shall be measured after first deperming the experiment package.

The change in magnetic dipole moment owing to soft perm shall be measured on each axis after first subjecting the experiment package to an enhanced Earth's field perm along the chosen axis. Total soft perm magnetic dipole moment shall not exceed 0.03 ampere-meter^2.

Active magnetic dipole moment shall be measured after deperming the experiment package, and shall be defined as the maximum change in magnetic dipole moment on each axis from an unpowered state through all modes of powered operation. Total active magnetic dipole moment shall not exceed 0.02 ampere-meter^2.

4.4.5 CONNECTOR AND CABLE REQUIREMENTS

4.4.5.1 Connector Requirements. Connectors shall be non-magnetic and conducting to chassis. Cadmium plating is not permitted and separate connectors shall be provided for squib circuits as discussed below. Mating connectors shall be provided by the experimenter for Boeing to install on the spacecraft wire bundle. Connectors shall be bonded to chassis in accordance with MIL-B-5087B, Class R. At least one connector pin shall be connected to chassis by a wire of 2 inches maximum length.

4.4.5.2 EED Connectors. Separate connectors shall be provided by the experimenters for EED firing circuits. The connectors shall have:

- o Provisions for peripheral termination of shields
- o Conducting back shells
- o Construction such that, when being mated, the connector shell will mate before any of the conductors and will not break contact until all inner conductors have broken contact. (Typical connector would be Bendix receptacle JTP02RE-8-44 (408) mating with plug 10-494961-8-445).

4.4.5.3 High Voltage Cable Requirements. Cables shall be non-magnetic, radiation proof (teflon satisfactory if shielded), and corona proof (after 2 hours outgassing). Cables shall be shipped with connectors attached as defined by the ICD. Boeing will define required length 60 days prior to required delivery.

4.4.5.4 Part Identification. All items furnished (cables, harnesses, etc.) shall be identified by a part number.

4.4.6 TOGGLE COMMANDS

Single channel commands that perform alternating or sequential functions are undesirable because occasions arise that require retransmittal of commands to assure receipt. This leaves the disposition of the toggle command unknown and may upset the desired sequence. This is particularly important for booms which effect the satellite stability. Sequential function commands provided by repetitive transmissions of the same command should be avoided.

4.4.7 TELEMETRY FORMAT AND INTERFACE REQUIREMENTS

Experiment processor interface characteristics and requirements of the telemetry signals are listed in Table 4.4-1. Characteristics and interface requirements of the timing signals are also shown. The gate pulse is 8, 12, 16, 24, 56 or 64 clock intervals in duration, depending on the number of bits in the digital word to be read out.

4.4.8 COMMAND CHARACTERISTICS

Commands to the experiments will be 20 to 32 VDC, 125 ± 25 millisecond pulses supplying up to 150 milliamperes, "level" commands at 28 ± 4 VDC, and ON/OFF power commands.

4.4.9 SATELLITE STABILITY REQUIREMENTS

In order to ensure dynamic stability of the spinning satellite, rigid booms shall have a lateral stiffness corresponding to greater than RPM/15 Hz bending frequency. For booms with unsymmetrical mass distribution about their axis, torsional frequency shall be greater than RPM/45 Hz.

4.4.10 ELECTRO-EXPLOSIVE DEVICES AND CIRCUITS

All experiment EEDs and circuitry shall comply with the requirements of SAMTECM 127-1 or AFTRM 127-1 and the 100 watt per square meter option of paragraph 3.3, Appendix A to AFMTGP 80-2 (issue of 1 Oct. 1963). EEDs shall be 1 watt, 1 ampere (1 ohm) no-fire and shall have been tested for RF and static sensitivity.

Table 4.4-1: Standard Payload/Processor Interface Characteristics

The following will apply to all payload/processor interfaces unless specified otherwise.

I GENERAL INTERFACE CHARACTERISTICS

A. A common ground return for both digital and analog signals will be used, unless the payload has separate digital and analog grounds.

B. Maximum signal inputs to the processor as a result of a signal anomaly in the payload shall be -.6 to +6.0 VDC.

II ANALOG INTERFACE CHARACTERISTICS (PROCESSOR INPUTS)

A.	Linear Signal Range	0 to +5.0 VDC
B.	Ground Return	Common to all analog channels
C.	Source Impedance	Less than 10K (Discrete $<$ 5K)
D.	Processor Impedance	10 Meg $\pm$ 10%

III DIGITAL INTERFACE CHARACTERISTICS (PROCESSOR INPUTS)

A.	Signal Levels: Nominal	$0 \pm .5$, $4.5 {}^{+.5}_{-2.0}$ V
	Limits	-0.6 and +6.0V
B.	Processor Impedance:	$>$100K
C.	Switching Threshold	Signals + 0.8V = Logic "0" Signals + 2.0V = Logic "1"

IV CLOCK AND CONTROL SIGNALS (PROCESSOR OUTPUTS)

A.	Logic Levels:	$0 \pm .5$, $4.5 {}^{+.5}_{-2.0}$ VDC (TTL Compatible)
B.	Rise and Fall Times:	1 Microsecond with 500 PFD Loading
C.	Phasing:	All pulses have positive going leading edges in phase with the 16.384 KHz clock positive going leading edges.
D.	Skew:	100 NS Max
E.	Source Impedance:	1 K Max
F.	Load Impedance:	25K Min
G.	Data Transfer:	Data is transferred into TLM Register in Payload on Leading Edge of Gate Pulse.
H.	Data Readout:	MSB of data is read by Processor at end of first clock period in sample interval. Data is shifted at beginning of second clock period by 16 KHz clock and read at end of second clock period, etc.
I.	Ground Return:	Common to all digital signals
J.	Main Frame Pulse:	1 Word long coincident with first word of each main frame
K.	Master Frame Pulse:	1 word long coincident with first word of each master frame
L.	Sync Pulses:	1 word long occurring at specified locations
M.	Load Pulses:	1/2 bit wide occurring in first half of first bit of sample interval
N.	1 Hz Sq. Wave:	Positive going edge coincident with leading edge of master frame
O.	262.144 KHz Clock:	Leading edge of 16 Kz clock pulse coincides with a leading edge of 262 KHz clock pulse

4.5 COMPARISON TO ASSESS STUDY

THE BSRM, USING RELATED EXPERIENCE FROM S3, IS DIRECTLY ANALOGOUS TO THE AMES AIRBORNE SCIENCES PROGRAM IN AREAS OF FULL EXPERIMENTER INVOLVEMENT, STANDARDIZED INTERFACES, AND MINIMUM DOCUMENTATION AND CONTROLS. HIGH RELIABILITY AND LOW COST CAN BE ENSURED FOR BSRM BY THE CONTINUATION OF THIS MANAGEMENT CONCEPT.

The proposed BSRM integration concept was reviewed and compared to the NASA-Ames Airborne Science Office ASSESS study program results documented in NASA TMX-62,288 dated July 1973. This Ames study identified and documented management concepts and operating procedures of the Airborne Science program as they may apply to the planning of Shuttle Spacelab operations. Certain aspects of the management concepts and operating procedures are also appropriate for "free-flier" spacecraft and are similar to those currently being implemented on the USAF Space Test Program (STP) missions.

As discussed in NASA TMX-62,288, the Ames Airborne Science Office has evolved procedures that foster scientific research yet are as informal and free of restrictions and documentation as possible, consistent with flight safety and the attainment of scientific objectives. A unique feature of the ASO operation is the active participation of experimenters in all aspects of the research program. The experimenters have the responsibility to construct and test their equipment, assist with installation in the aircraft, and participate in flights to obtain the scientific data. This one practice, more than any other, underlies the success of the Airborne Science approach. It has been enthusiastically accepted by the scientific community and is productive of research results with a minimum of preparation time, documentation, and controls, and at relatively low cost.

Boeing's experience on the STP S3 satellite program affirms the advantages of the active participation of experimenters in satellite scientific research. On the S3 program, a close working relationship with experimenters was developed early and a conscientious effort made to maintain this relationship throughout the program. While interface and integration problems did develop, there was a noticeable positive attitude on the part of all agencies and individuals to solve the difficulties and get the job done. This cooperative spirit contributed a great deal to the on-schedule delivery and flight success of S3-1. The two remaining spacecraft, currently in storage awaiting launch, are expected to duplicate the S3-1 success primarily because of the identical close working relationships developed with the experimenters on those vehicles.

Of the many conclusions of the study reported in NASA TMX-62,288, the first two, quoted below, are directly verified by the S3 program and are expected to continue into the BSRM program.

1. Full experimenter involvement and responsibility throughout the entire mission leads to low cost and high reliability of payloads through minimal documentation and controls needed for payload management, simplified procedures for experiment preparation, and the availability of experimenter expertise in operation and maintenance, which ensures a high level of experiment performance in flight.

2. Small management staff, direct interaction between principals, and rapid decisions minimize experiment development time and reduce payload costs.

These characteristics will be ensured on BSRM by the early identification and design freeze of standardized subsystem interfaces. This is an improvement over the S3 program where the first S3 spacecraft and experiments were being developed concurrently. The interface and integration problems experienced on S3 will be minimized on the BSRM by the standardization concept. The BSRM thus represents a unique opportunity to introduce an innovative, efficient and cost-effective approach to earth orbital space science research.

In the Shuttle era of the next decade it is expected that considerable opportunity will be presented to fly small "free-flier" spacecraft on many missions when excess payload capability is available. If the Shuttle achieves the high utilization rate currently projected, these opportunities may be overwhelming in number. The only effective way to provide the vast quantity of small satellites required will be to develop a modular design with standardized subsystem and Shuttle interfaces. The S3 and other DOD satellites, which fly piggyback on a USAF Host Vehicle, have successfully demonstrated the feasibility of this approach. The BSRM design, currently configured for Scout and other expendable booster systems, can further develop and perfect this concept in the interim period prior to the Shuttle becoming operational.

Table 4.5-1 summarizes and compares the key features of the ASSESS study, the S3 program and the proposed BSRM methodologies. The similarities are obvious and ensure a successful, low cost BSRM program if properly implemented.

Table 4.5-1. Methodology Comparison

	ASSESS Study	S3 Program	Proposed BSRM
Full Experimenter Involvement	X	X	X
High Degree of PI Responsibility	X	X	X
Minimum Documentation	X	X	X
Minimum Controls	X	X	X
Simplified Procedures	X	X	X
Small Management Staff	X	X	X
Standardized Interfaces	X	1	X
Direct Interaction Between Principals	X	X	X
Rapid Decision Channels	X	X	X
Experienced Spacecraft and PI Team	X	X	X
Flight Proven Subsystems	X	1	X

1 After development of first S3, remaining vehicles represent standardized, flight proven concept.

5.0 SPACECRAFT OPTIONS

5.0 SPACECRAFT OPTIONS

THE BASELINE BSRM SPACECRAFT IS READILY ADAPTABLE TO A VARIETY OF EARTH ORBIT SCIENTIFIC MISSIONS WITH THE INCORPORATION OF ONE OR MORE OPTIONAL SUBSYSTEM KITS. THE OPTIONS USE FLIGHT PROVEN EQUIPMENT AND THUS RETAIN THE QUALIFICATION STATUS OF THE BSRM SATELLITE. A COST EFFECTIVE BSRM PROGRAM IS ENSURED BY ELIMINATING THE REQUIREMENT FOR NEW DESIGN, TEST AND QUALIFICATION FOR EACH MISSION UNDERTAKEN.

The approach of providing optional kits for incorporation into a baseline modular spacecraft design has been successfully demonstrated by Boeing on the USAF S3 program. The kits described below for the BSRM provide for three-axis stabilization, improved power regulation, flexibility in experiment data rates and autonomous attitude control. The baseline subsystems will be designed to accept the options selected for any specific mission with an absolute minimum impact on the baseline spacecraft. Wiring changes, relay box module revisions and processor module changeouts are all that are required to install the options.

The BSRM reliability analysis presented in Section 3.8 of this report includes discussion of the spin-stabilized and three-axis control systems only. The incorporation of one or more of the other subsystem options discussed below would have a slight affect on the BSRM spacecraft reliability. The autonomous attitude control option may actually increase total spacecraft reliability analysis results since the ground control loop would be retained as a backup to provide redundancy. Further detailed reliability and failure modes and effects analyses will be performed under the BSRM hardware contract for the specific equipment and options selected.

5.1 CONFIGURATION KIT

The optional BSRM configurations contain the elements of the baseline. Options are achieved by simple additions or deletions and the basic S3 characteristics are maintained.

The optional three-axis stabilized version of BSRM is shown in Figure 5.1-1. The reorientation of some equipment items, the deletion of the spin motors, the deletion of the deployable antennas and earth sensor, and the addition of the scanwheel constitute the kit change. These changes result in no significant impact to the baseline BSRM.

Additional options discussed in paragraphs 5.2 and 5.5 require the addition of electronic packages (one in each case) to the basic equipment. There is no impact to the configuration as space is available and the packages are small. The changeout of the tape recorder as discussed in 5.4 is easily accommodated as the recorder housing is the same in all cases.

The shape of the BSRM may also be modified as an option. To accommodate a large payload, the configuration of the satellite could easily be modified by adding structure to form a hexagonal shape which would increase the interior

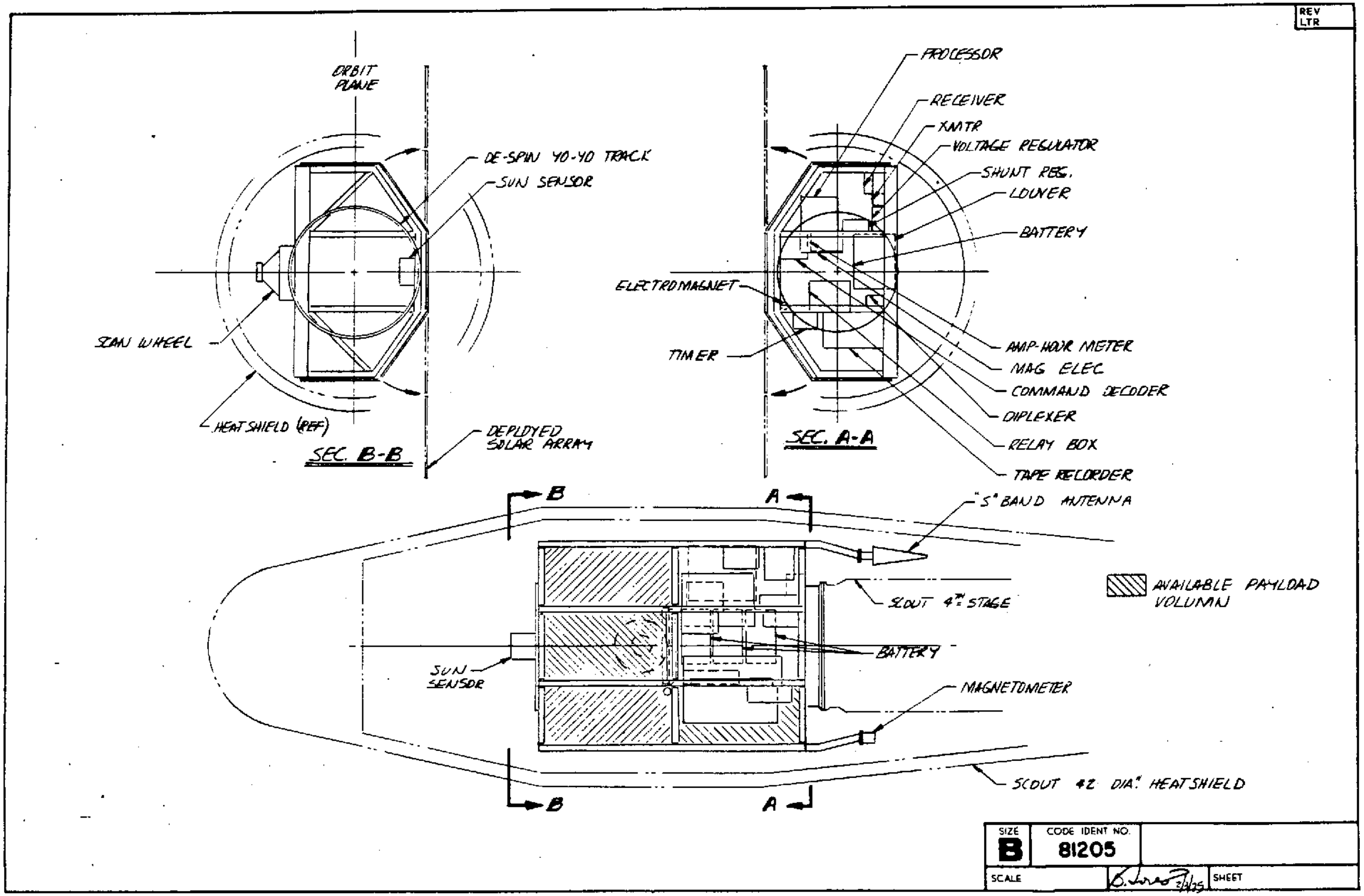


Figure 5.1-1. Three-Axis Stabilized BSRM

volume while maintaining the identical Scout interface. The solar array could be identical to the baseline or a fixed, body mounted array around the periphery of the hexagonal structure depending on mission requirements.

5.2 REGULATED POWER KIT

The baseline BSRM electrical subsystem regulates the bus power to ± 4 VDC, which is adequate for all housekeeping loads. Additional regulation, if required, is the responsibility of the payload which needs it. However, it is possible to furnish a ± 2% regulated bus for payload elements with fairly minor modifications to the existing system. Figure 5.2-1 shows this modification. A second bus, regulated to ± 2%, is created by tapping the primary bus and adding a DC-DC converter. This method has the advantage of maximizing overall system efficiency by not regulating the entire system at ± 2%, and still providing good regulation for those elements which require it.

5.3 THREE-AXIS ATTITUDE CONTROL AND DETERMINATION KIT

To meet the requirements of those missions which require a stable platform, a three-axis control system is available as a kit for incorporation into the BSRM.

The three-axis version of BSRM is dynamically identical to the baseline spacecraft but a spinning wheel replaces the spinning of the spacecraft. The momentum vector is controlled magnetically by modulating a single coil in both systems. The major difference is the need, in the three axis system, of maintaining a third axis of control. This is normally achieved by a closed loop onboard pitch control subsystem which uses sensed pitch errors to command wheel torques, Figure 5.3-1. The scanwheel performs horizon sensing functions and replaces the spinning system earth sensor.

Because the scan rate is high (~ 1800 rpm) it becomes impractical to transmit horizon crossings to earth for processing. Instead roll and pitch errors are determined on board and the errors are telemetered to ground.

No spin rate control is needed on a three axis system but there is an analogous requirement for wheel desaturation. This is performed by one of the magnets perpendicular to the spin axis as shown in Figure 5.3-1. No commutation is now necessary but the polarity requires changing every half orbit.

The spinning system is essentially two axis stabilized and requires two reference axes. The three axis system is more difficult to control because of the additional third axis and the need for a third reference. Only one basic orientation is practical. This has the spin axis aligned with the orbit normal, one axis pointed towards the local vertical and the third in the orbit plane in the general direction of motion. The orientation restricts experiment sensors to a fixed attitude with respect to rotating orbit axes but could include a variety of potentially useful pointing arrangements; away from the earth, for example, or inclined to the orbit normal so that a cone is swept out once per orbit. Maneuver in the orbit plane, i.e., rotations about the orbit normal are relatively easily accomplished.

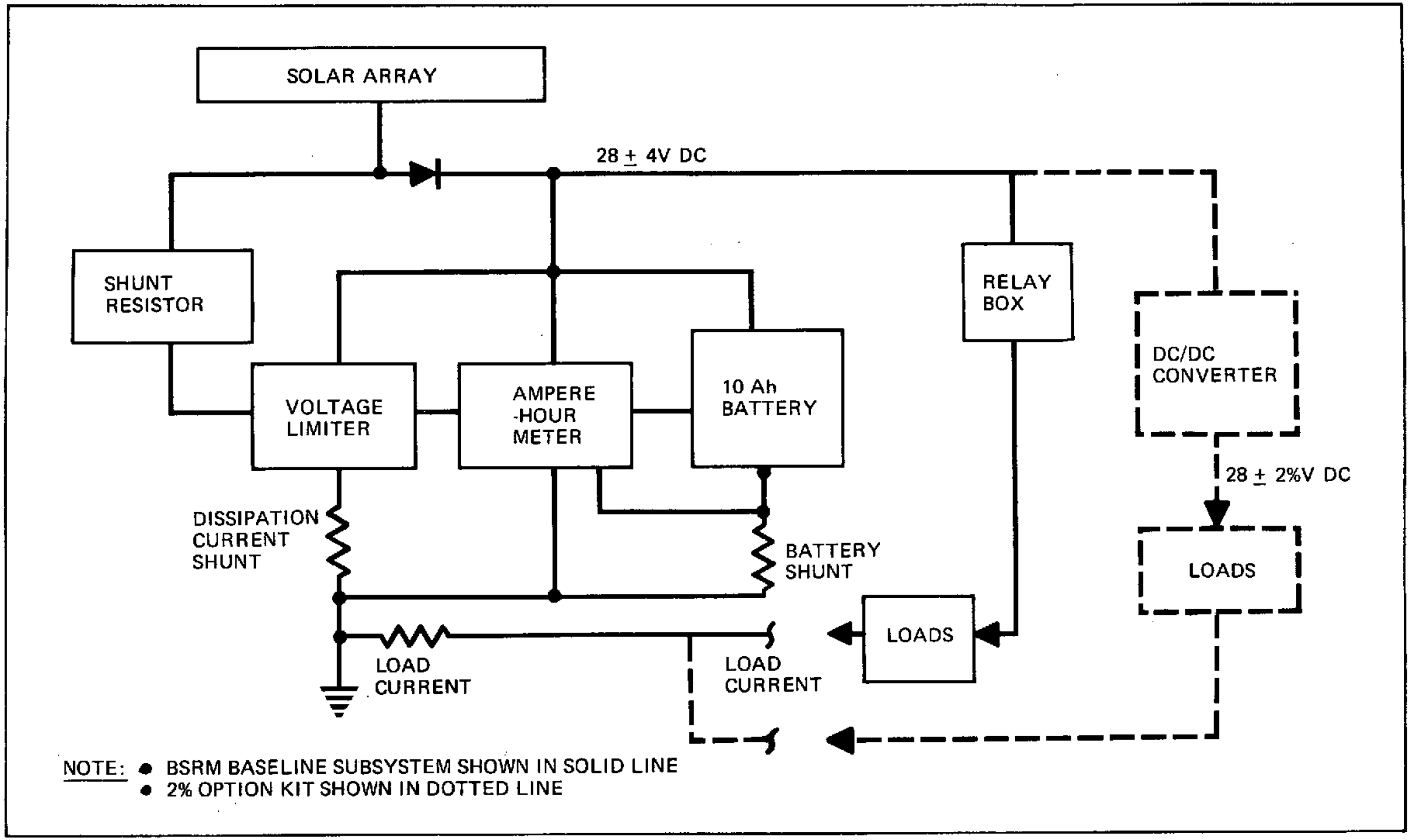


Figure 5.2-1. BSRM 2% Regulation Option Kit

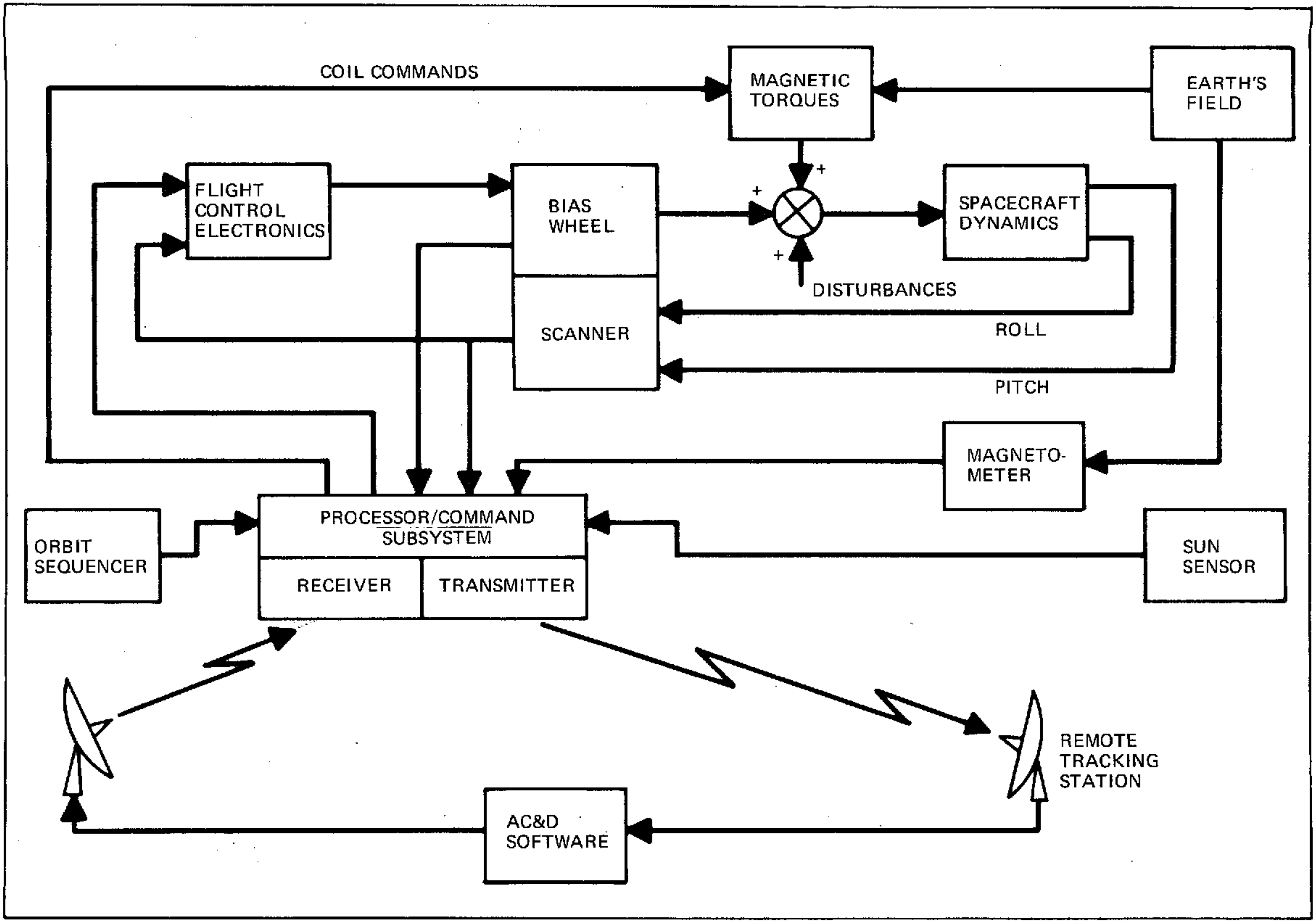


Figure 5.3-1. Three-Axis Attitude Control System Block Diagram

Orientations other than the local vertical are not impossible but can only be entertained for certain very specific cases. The problem is again the need for references. Periodic information only is necessary in the two axes which describe the angular momentum direction because of the inherent stiffness in the system. Continuous error information however is necessary to control motion about the momentum vector. Star sensors offer a possible solution in some cases but entail added weight, cost and complexity.

Again, as in the case of the spinning satellite, performance capabilities depend heavily on the orbit parameters, spacecraft configuration and maneuver requirements that only broad generalizations can be made. Under ideal conditions of a moderately low (~ 200 nm.) circular orbit at high inclination with no maneuver, the estimated performance capability of the three axis system is shown in Table 5.3-1. In general, accuracies tend to be better in circular than elliptical orbits. An increase in altitude decreases control authority because the earth's field is weaker, but in general control and determination are not degraded.

Table 5.3-1. Estimated Performance Capability of Three-Axis Control System

MODE	ATTITUDE	
	HOLD	DETERMINATION
Local vertical — roll	±1.0 deg	±0.25 deg
pitch	±0.5 deg	±0.5 deg
yaw	±2.0 deg	±0.5 deg

The equipment for the three-axis BSRM attitude control subsystem are listed in Table 5.3-2. Other subsystems are identical to the spin-stabilized spacecraft listed in Table 3.1-2 except that only one S-band antenna may be required.

TABLE 5.3-2: BSRM Three-Axis Stablilized AC&D Subsystem Equipment List

	QTY.	SUPPLIER	FLIGHT HISTORY S3	FLIGHT HISTORY OTHER
ELECTROMAGNETS	3	BOEING	X	
CONTROL ELECTRONICS	1	ITHACO 1		DELTA PAC
TIMER/SEQUENCER	1	CELESCO	X	
MAGNETOMETER SENSOR	1	SCHONSTEDT	X	
MAGNETOMETER ELECTRONICS	1	SCHONSTEDT	X	
WOBBLE DAMPER	1	BOEING	X	
SUN SENSOR (TWO AXIS)	1	ADCOLE		USAF
SCANWHEEL	1	ITHACO		DELTA PAC

1 MINOR ADDITIONS TO FLIGHT PROVEN UNIT

5.4 TELEMETRY TRACKING AND COMMAND KITS

Optional high and low data kits supplement the BSRM baseline configuration to handle any data rate up to 32 kbps. Increased command outputs can handle up to 144 discrete pulses without significant hardware impact.

Increased or decreased capacity retains the BSRM baseline concept without significantly changing the TT&C system. The most efficient systems contain only the required channels and rates. However, cost effectiveness requires the minimum change for each mission. The concept chosen for BSRM combines both, in that high and low capacity configurations are presented which more closely match the actual requirements while still using hardware of the baseline design to the greatest extent possible.

5.4.1 DATA OPTION KIT #1 (HIGH DATA CAPACITY)

Tape Recorder. The high rate data option of 32 kbps extends the data rate of the BSRM baseline by a factor of two. The selection of this rate was based upon keeping the highest cost TT&C component minimized by using off-the-shelf hardware. This component is the tape recorder used for data storage. The Odetics Model DDS-3000 used on AF/STP 72-2 has a 32 kbps capability and recording time of 400 minutes. Discussions with the supplier recommended this model for a least cost and minimum modification effort. Only three tracks of the total 5 will be needed to give 240 minutes (80 minutes/track) recording time at 32 kbps. The playback ratio will be 16:1, yielding a 512 kbps (NRZ-L) output bit rate. The size, weight and power are comparable to the baseline unit.

PCM Processor. The significant change here will be the bit rate increase from the baseline 16 kbps to the 32 kbps required for recording. Input channel types and rates will be adjusted to that required by proper jumper wiring of the processor input gates and clocking signals. The input channel capacity is expected to be very similar to that of the baseline. Where more inputs are required, trade-offs will be made between splitting up some of the higher sampled channels and the increased sample rate due to the X2 bit rate increase. This will be accomplished during later phases of this study as requirements become more clearly defined. Spacecraft measurements will maintain the baseline allocation of 1700 bps.

Baseband. The baseband section of the Processor/Baseband unit will require slight bit rate filter changes to the 1.024 MHZ and 1.7 MHZ sub-carriers. The change in bit rate (X2) is nearly offset by the NRZ-L coding. Also, in the real time mode (Mode I) the 1.7 MHZ SCO is carried as spare and could accommodate a high rate real time PCM or wideband analog channel if required.

Summary. The following summarizes the high data rate option:

Input Channel:	Baseline or as required
Mode I R/T:	32 kbps NRZ-L on 1.024 MHZ
R/T:	Spare on 1.7 MHZ

Mode II R/T:	32 kbps NRZ-L on 1.024 MHZ
P/B:	512 kbps NRZ-L on 1.7 MHZ
Record Time:	240 minutes (3 tracks)
Playback Ratio:	16:1
Total Storage:	$0.93X10^9$ bits
Playback Time:	6.0 minutes for 95 minute orbit

5.4.2 DATA OPTION KIT #2 (LOW DATA CAPACITY)

Tape Recorder. For reduced data missions a 6.4 kbps recorder is available off-the-shelf from Odetics. Previously flown on OSO/HEAO, this unit can be used without high development costs and again sets the bit rate for the low option. This Odetics model (DDS-1010) has a 220 minute record time for $0.85X10^8$ bits total storage. The size, weight and power are similar to the baseline unit.

PCM Processor. The bit rate will be reduced from the baseline 16 kbps to 6.4 kbps required for recording. The input channel capacity will again be adjusted to the incoming data by jumper wiring in the input gate and clocking circuits. The basic format and spacecraft measurements will remain the same. Measurement sample rates will be adjusted to maintain the baseline spacecraft allocation of approximately 1700 bps.

Baseband. The bit rate filter on the 1.024 MHZ and 1.7 MHZ sub-carriers will not require changing from the baseline as they fall well within the baseline channel width. The Mode I 1.7 MHZ sub-carrier is carried as spare and could accommodate a high rate real time PCM or wideband analog channel if required.

Summary. The following summarizes the low date rate option:

Input Channel:	Baseline or as required
Mode I R/T:	6.4 kbps BI-Ø on 1.024 MHZ
R/T:	Spare on 1.7 MHZ
Mode II R/T:	6.4 kbps BI-Ø on 1.024 MHZ
P/B:	128 kbps BI-Ø on 1.7 MHZ
Record Time:	220 minutes
Playback Ratio:	20:1
Total Storage:	$0.85X10^8$
Playback Time:	4.5 minutes for 95 minute orbit

5.4.3 DATA OPTION KIT #3 (WIDEBAND ANALOG - FM/FM)

Baseband. In the original design of the S3 Processor/Baseband unit, the real time Mode I incorporated wideband FM on the 1.7 MHZ sub-carrier. However, the BSRM baseline did not require this capability and the wideband FM P.C. card was removed and a dummy P.C. card installed. This capability may be retained by replacement of the FM P.C. card. The four (4) channels include one wideband analog modulated directly on the 1.7 MHZ sub-carrier with three IRIG FM sub-subcarriers (C, G, E) mixed linearly. Other IRIG channels could be substituted for those shown if required.

Summary. The following summarizes and identifies the characteristics of the four (4) wideband analog channels of this option.

INPUT	DEVIATION	RESPONSE (-3 dB)	CHANNEL
± 2.5V	± 35 KHZ	.1-10 KHZ	DIRECT
-.6 to +5.6V	± 6 KHZ	0-6 KHZ	IRIG C
-.6 to +5.6V	± 10.5 KHZ	0-10.5 KHZ	IRIG E
-.6 to +5.6V	± 18.6 KHZ	0-18.6 KHZ	IRIG G

5.4.4 COMMAND OPTION KIT #1 (ADDITIONAL PULSE OUTPUTS)

Command Decoder. The S3 command decoder utilized only seven of the eight data bits available for decoding. These provided a 3 (8 line) X 4 (16 line) X-Y matrix producing 128 outputs. However, the three P.C. Output matrix cards contain 144 circuits (48 each). Hence, a simple wiring of the 16 unused output circuits plus one additional logic decode chip (8th data bit) will expand the command discrete outputs from 128 to 144. Greater than 144 outputs could be implemented but the impact would be significantly increased beyond this simple change.

5.5 AUTONOMOUS CONTROL OPTION KIT

The three-axis earth oriented option uses ground commands to control the orientation of the wheel spin axis (the spacecraft pitch axis) and to desaturate the wheel. Both of these functions can be performed autonomously by a self contained system where the control loop is closed on-board and the spacecraft operates without assistance from the ground. Mode switching or maneuver commands would still be input from the ground but the on-going process of control in a given mode would be independent.

Autonomous spin axis orientation has not flown. Funded studies at TRW and Ithaco have shown feasibility and Boeing has examined the concept. Desaturation on-board has been demonstrated in flight.

The major advantage of an autonomous system is the elimination of ground software for attitude control and reduction in ground station contact time. The system can look after itself indefinitely in a fully hands off condition. (Software may still be necessary for attitude determination, however). Speed of response also tends to be higher and the sequencer to control magnetic switching is no longer required. Studies by the Goddard Space Flight Center conclude that total program costs are reduced.

Disadvantages are a certain loss in flexibility since man is excluded from the loop. The ground controlled approach easily accommodates an unplanned maneuver, for example, while the autonomous system will not. The autonomous system requires additional electronics to implement the on-board control laws.

In addition to spin axis orientation and desaturation, it is also possible to perform nutation damping autonomously by processing magnetometer outputs to generate appropriate magnetic torques. This scheme eliminates the passive damper to gain several pounds but at the expense of additional complexity and some loss in reliability.

The three autonomous functions, spin axis orientation, desaturation and damping, are independent and can be implemented individually. The best choice of all autonomous, all ground control or a mixture can only be determined from a detail study of specific mission requirements.

6.0 MISSION REQUIREMENTS AND SPACECRAFT COMPATIBILITY

6.0 MISSION REQUIREMENTS AND SPACECRAFT COMPATIBILITY

THE BOEING SMALL RESEARCH MODULE IS CAPABLE OF PERFORMING A SIGNIFICANT NUMBER OF SCIENTIFIC MISSIONS PROPOSED FOR SMALL (SCOUT-CLASS), LOW-EARTH ORBIT SATELLITES.

In addition to the two missions (Auroral and Aether Drift) specified in this study, Boeing utilized company-sponsored funds to evaluate some missions which have been proposed in response to Announcement of Opportunity for Scientific Definition of Scout Explorer-class Missions, A.O. #7, dated 15 July 1974. The purpose of the evaluation was to determine if the BSRM could be utilized to perform those missions. The preliminary requirements for these missions are shown in Table 6.0-1. The sections below address the individual missions and the ability of the BSRM to meet them.

6.1 AURORAL MISSION

Table 6.0-1 shows the preliminary requirements for the NASA-Ames Research Center Auroral Mission. The proposed mission can be accomplished using the Boeing Small Research Module (BSRM).

The mission will investigate some of the important unsolved problems of auroras and auroral phenomena in the atmosphere and ionosphere. Prime scientific emphasis is placed on the quantitative evaluation of atmospheric effects and the distribution of energy in time and space caused by measured particle precipitation and radiation incident on auroral regions. A better understanding of the physical and chemical processes responsible for the production of auroras is the main purpose of the mission. The mission stresses the close correlation between remote sensing optical instruments and "in-situ" optical and particles-and-fields instruments. The total passband encompassed by the optical instruments extends from 0.025 microns (EUV) to 1.66 microns (IR); the charged particle spectrum will be measured from 0.1eV to 20keV. In addition, "in situ" measurements will be made of concentrations of upper-atmospheric constituents and of the three-dimensional magnetic field.

The Auroral Mission experiment payload consists of 11-14 instruments, depending on which of four mission options is selected. The effects of the options on the spacecraft subsystems are discussed below.

The weight of the BSRM is estimated to be 97.5 kg (see Table 3.1-1, Section 3.1). With a Scout launch capability of 150 kg for the 250 km by 800 km orbit, the weight available for payload is 52.5 kg. The estimated Auroral Mission payload weights are:

Baseline Mission:	46.6 kg
Option 1 Mission:	47.5 kg
Option 2 Mission:	48.6 kg
Option 3 Mission:	54.1 kg

Table 6.0-1. Mission Requirements

Mission	Orbit altitude	Inclination	Power	Data rate	Stabilization	Axis control	P/L weight	Special requirements
Auroral	250km perigee 800km apogee	Sun-synchronous	53.5w	16KBPS	Spinner	1°	48.6kg	Spin axis pointed at sun
Aether drift	550km circular	Terminator, sun-synchronous	31.5w	2KBPS	3-Axis geocentric	1°	28.5kg	12 Month life
X-ray telescope	500km circular	Equatorial	20	Low	3-Axis inertial	1°	82kg	
Gamma burst	500km circular	Equatorial	30	2KBPS	3-Axis geocentric	5°	82-91kg	
Soft x-ray	500km circular	Equatorial	20	400BPS	Spinner	1°	68kg	Spin axis pointed at sun
Extreme ultraviolet	350km circular	Equatorial	20	Low	Spinner		45kg	Spin axis pointed at sun
Ultraviolet	500km circular	Polar	<10	Low	3-Axis geocentric	1°	37kg	
Gamma burst	500km circular	Equatorial	25	16KBPS	3-Axis geocentric	1°	82-91kg	
Electrodynamics	300km perigee 3500km apogee	75°	20	FM/FM	Spinner		As much as possible	Spin axis perpendicular to orbit plane
Cosmic background radiation	1300km circular	Equatorial	30	45BPS	Spinner	1.5°	18.2kg	Spin axis pointed at sun
Solar telescope	350-500km circular	Equatorial	20-30	1.5KBPS	3-Axis		68kg	Solar pointing
Atmospheric anomaly	250km perigee 1050km apogee	15°	65 periodically	131KBPS real-time	Spinner		52kg	Spin axis perpendicular to orbit plane

BSRM is capable of accommodating up to the Option 2 mission and still retain the 15% weight contingency. The Option 3 mission can be accommodated only if some of the weight contingency is assigned to the payload. This was not considered advisable and the Option 3 mission payload was assigned to the Aether Drift Mission for the purposes of this study.

6.1.1 CONFIGURATION

Figure 6.1-1 shows the Auroral Mission Configuration and includes the 13 instruments which would be flown on Option 2 mission. There is room available for the Option 3 mission instrument should a reduced weight contingency be acceptable. The secondary structure, including the experiment support brackets is unique for the BSRM payload mix. As with the existing S3 spacecraft, each satellite requires rearrangement of subsystems and payload installations to provide thermal and mass properties balance and to accommodate fields of view and other experiment requirements. The placement of both subsystem and payload equipments enables the maximum spin inertia to be achieved with the spin axis through the minimum satellite thickness.

Three simply-hinged booms are mounted on the satellite. Two antenna booms are hinged from opposite sides of the satellite to deployed positions roughly coincident with the spin axis. The magnetometer is mounted on one antenna boom to reduce the number of deployments. The plasma probe, if flown, would be mounted on the third boom normal to the spin axis.

6.1.2 ELECTRICAL POWER

The Auroral Mission experiment power requirements are tabulated in Table 6.1-1, for the baseline mission and the two mission options. The corresponding mission load profile for the Option 2 mission (worst case) is shown in Figure 6.1-2. The data dump is assumed to occur during the occulted period on every second orbit. The results of a system energy balance analysis are shown in Table 6.1-2. Data presented include solar array and battery sizes required and a comparison with BSRM capabilities. The difference in solar array size between the baseline mission and the worst case option (2) is only 5 watts, an insignificant amount.

Battery life is determined by the number of charge/discharge cycles and the depth-of-discharge. For a six-month mission the maximum permissible depth-of-discharge is about 32%. The predicted depth-of-discharge for the Auroral Mission is 22%, implying a cycle life margin of 82%.

The Option 3 mission would add another 1.5 watts (continuous) to the power requirements, which results in a solar array size requirement of 168 W. BSRM is capable of producing this amount of power for the Auroral Mission.

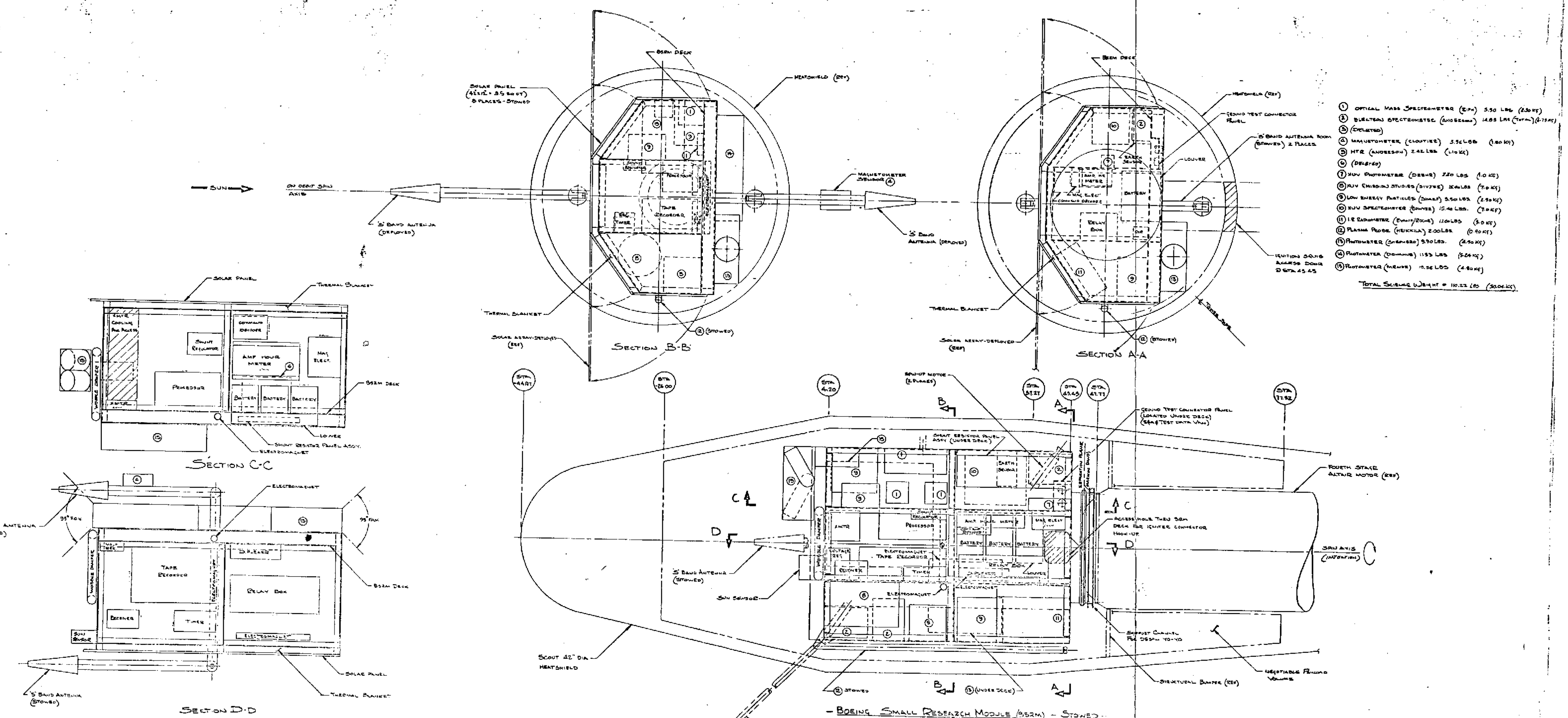


Figure 6.1-1 BSRM Configuration For Auroral Mission

NAS2-8526

BOEING SMALL RESEARCH MODULE (BSRM) — AURORAL MISSION

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Table 6.1-1. Auroral Mission Experiment Power Requirements

Baseline mission		Experimenter	Power	Duty cycle
1	Optical mass spectrometer	Zipf	4.0	100%
2	Electron/positive ion spectrometer	Anderson	3.75	100%
3	Deleted	– – –	– – –	– – –
4	Vector magnetometer	Cloutier	10.0	100%
5	Deleted	– – –	– – –	– – –
6	Deleted	– – –	– – –	– – –
7	Photometer	Deehr	3.7	Polar
8	Spectrophotometer	Sivjee	5.0	Polar
9	Soft electron spectrometer	Sharp	3.0	100%
10	EUV spectrograph	Bowyer	3.7	Polar
11	Infrared radiometer	Evans/Roche	4.0	Polar
12	Deleted	– – –	– – –	– – –
13	Photometer	Shepherd	3.0	Polar
14	Airglow photometer	Donahue	6.7	Polar
15	Photometer	Mende	4.0	Polar
		Total:	50.85 W (20.75 W continuous)	
Option 1 mission				
12	Langmuir probe	Heikkila	1.0	100%
		Total:	51.85W (21.75W continuous)	
Option 2 mission				
5	High time resolution detector	Anderson	1.6	100%
12	Langmuir probe	Heikkila	1.0	100%
		Total:	53.45W (23.35W continuous)	

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ORBIT PERIOD: 95.1 MIN
EXPERIMENT PERIOD: 70 MIN

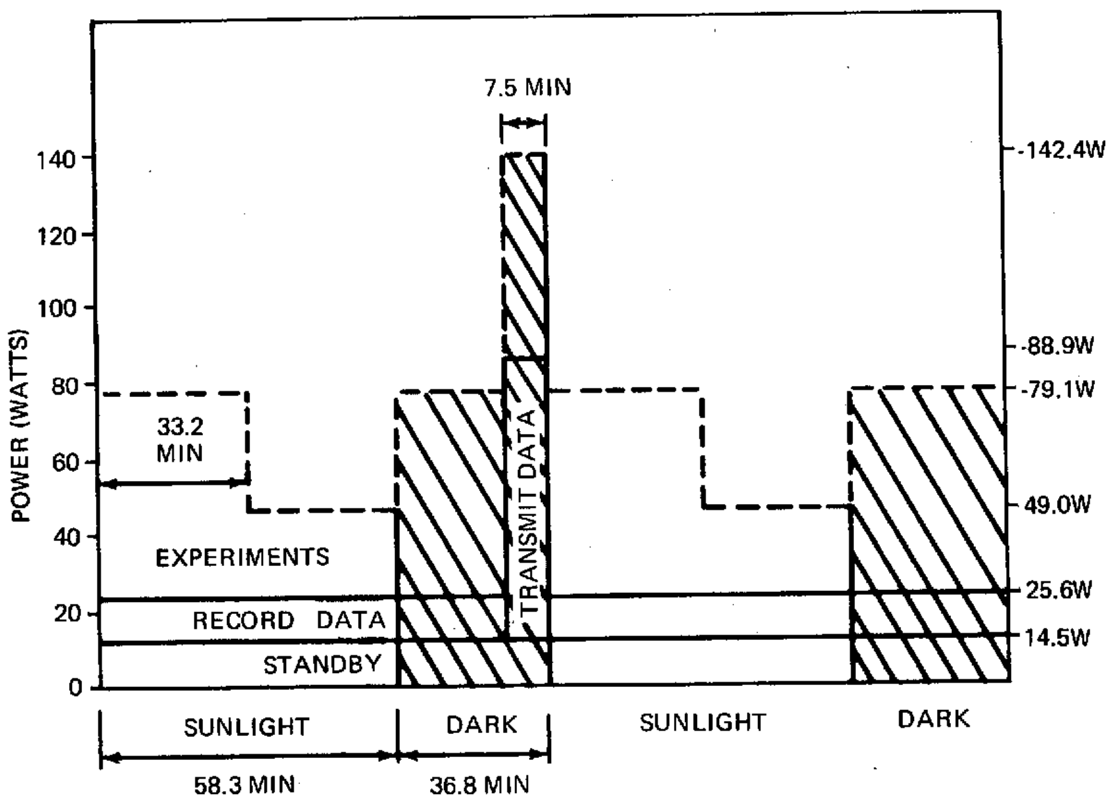

Figure 6.1-2 Auroral Mission Load Profile

Table 6.1-2: BSRM Electrical Subsystem Capabilities

SOLAR ARRAY	
o BASELINE MISSION REQUIREMENT	160 W (ASSUMES TRANSMISSION EACH ORBIT)
o OPTION 2 MISSION REQUIREMENT	165 W (ASSUMES TRANSMISSION EACH ORBIT)
o BSRM CAPABILITY	168 W*
o COMPARISON	EXCEEDS ALL MISSION REQUIREMENTS
BATTERY	
o OPTION 2 (WORST CASE)	2.2 Ah
o BSRM CAPABILITY	10 Ah
o COMPARISON	o PREDICT 22% DEPTH-OF-DISCHARGE (DOD) (PERMISSIBLE DOD IS 32%) o EXCEEDS MISSION REQUIREMENTS

*EOL (6 MONTHS) AND 20 SQ. FT. OF SOLAR PANEL

6.1.3 ATTITUDE CONTROL AND DETERMINATION

The proposed BSRM attitude control subsystem is capable of meeting the requirements of any of the Auroral Mission options. The spacecraft spin axis will be pointed at the sun. A two-axis sun sensor will replace the baseline one-axis sensor, as it will be used as the primary attitude reference device. The earth sensor now would measure the spacecraft spin rate.

6.1.4 TELEMETRY TRACKING AND COMMAND

Baseline Auroral Mission requirements as compiled from numerous contacts with the principal investigators are shown in Figure 6.1-3. Their inputs were adjusted to fit into standard telemetry channels and for a spacecraft spin rate of 5 RPM. These must be regarded as preliminary for some experiments are spin rate dependent and could impact the overall bit rate significantly should the spin rate vary from the assumed 5 RPM. A slower spin rate would lessen the telemetry bit rate requirement.

Operational requirements for the Auroral Mission specify about 95 minutes for the 250 X 800 km orbit. Analysis at Ames of STDN ground tracking coverage shows that the worst case pass time per orbit is approximately 7.5 minutes. This sets the tape recorder playback time of a single orbit to 7.5 minutes. A 2 orbit (190 minutes) tape recorder storage capacity is also necessary. Slant range for this orbit is 2250 km based on 5^{o} elevation, below which communication becomes unpredictable.

A requirement for turn-around ranging has not been established, but it is expected that at least the noncoherent capability of the BSRM baseline TT&C system will be needed for ephemeris determination.

EXPERIMENT	P.I.	TELEMETRY BIT RATE*	COMMANDS
SCANNING PHOTOMETERS (3)	DEEHR	1,024	4 PULSE
EUV SPECTROGRAPH	BOWYER	2,144	20 PULSE
FUV SPECTROGRAPH	SIVJEE	2,048	6 PULSE
PHOTOMETER, 2 CH.	MENDE/REAGAN	2,354	2 PULSE
REDLINE PHOTOMETER	SHEPPARD	771	4 PULSE
ELECT. & ION SPECTROMETER (3)	ANDERSON	1,585	2 PULSE
VECTOR MAGNETOMETER	CLOUTIER	1,330	2 PULSE
I.R. RADIOMETER	EVANS/ROCHE	178	6 PULSE
LOW ENERGY PARTICLES	SHARP	1,216	4 PULSE + 2 DIGITAL
OPTICAL MASS SPECTR.	ZIPF	552	5 PULSE
AIR GLOW PHOTOMETER	DONAHUE	210	8 PULSE
		13,412	63 PULSE + 2 DIGITAL
SPACECRAFT HOUSEKEEPING		1,705	60 PULSE + 4 DIGITAL
		15,117	123 PULSE + 6 DIGITAL

*BASED ON 5 RPM S/C SPIN RATE

Figure 6.1-3: Baseline Auroral Mission TT&C Requirements

The BSRM baseline TT&C system as described in section 3.4 meets the baseline Auroral Mission requirements. Both the telemetry bit rate and command outputs utilize better than 90% of the maximum capability of the TT&C system indicating an efficient design. A 10% design margin is normally used in sizing this type of system and represents a growth buffer for changing requirements.

The following chart summarizes the baseline Auroral requirements and baseline BSRM TT&C system capabilities based upon the original S3 concept.

SYSTEM PARAMETERS	AURORAL REQUIREMENTS	BSRM CAPABILITIES	BSRM COMPARISON
STDN COMPATIBILITY	S-BAND	S-BAND	MEETS
TLM & CMD RANGE (S.R. @ 5^{o} EL)	2250 KM	7000 KM	EXCEEDS
DATA RATES	15.1 KBPS	16.3 KBPS	EXCEEDS
DATA STORAGE	190 MINUTES (2 ORBITS)	210 MINUTES	EXCEEDS
DATA PLAYBACK	7.5 MINUTES (1 ORBIT)	6.8 MINUTES (1 ORBIT)	EXCEEDS
RANGING	NOT DEFINED	YES	COMPATIBLE W/STDN
COMMANDS	123 PULSE 6 DIGITAL	128 PULSE 7 DIGITAL	EXCEEDS

Auroral Mission Option #1 Requirements. Adding the Plasma Probe experiment (Heikkila) to the Auroral baseline as an option adds the following requirements:

TELEMETRY BIT RATE	COMMANDS
872 + (98,304 R/T)	7 Pulse + 1 Digital

The 98 kbps real time data shown are desired over the poles during an event and could be satisfied by a real time transmission over the STDN Fairbanks, Alaska station.

The BSRM baseline data system will handle the added telemetry requirements. The baseline 10% capacity will be used and effectively eliminates the growth buffer. High bit rate data (98 kbps) can be wired into the Mode I 1.7 MHZ subcarrier and transmitted in real time provided the experiment formats the data and sends it to the Baseband in serial form. Command pulse requirements exceed 128 making the Command Option Kit #1 necessary.

The following chart summarizes the differences between the BSRM baseline and Auroral Option #1.

SYSTEM PARAMETERS	AURORAL OPTION #1	BSRM CAPABILITIES	BSRM COMPARISON
DATA RATES	15,989 KBPS + (98,304 R/T)	16.3 KBPS	MEETS
COMMANDS	130 PULSE 7 DIGITAL	144 PULSE 7 DIGITAL	EXCEEDS W/CMMD KIT #1

Auroral Option #2 Mission Requirements. Adding both the (1) Plasma Probe (Heikkila) and (2) High Time Resolution Detector (Anderson) to the Auroral baseline as an option adds the following requirements:

	TELEMETRY BIT RATE	COMMANDS
1)	872 + (98,304 R/T)	7 PULSE + 1 DIGITAL
2)	1072 + (16,384 R/T)	7 PULSE
	1944 + (114,688 R/T)	14 PULSE + 1 DIGITAL

The 114 kbps real time data are both desired over the poles during an event, and could be satisfied by a real time transmission over the Fairbanks tracking station. The BSRM Data Option Kit #1 (High Data Capacity) will be required to handle the added telemetry requirements. This is not an efficient match as the capacity is nearly twice the required bit rate. A bit rate more closely tailored to the requirements (i.e., 18-20 kbps) would be much more efficient but costly in terms of modifications to off-the-shelf recorders. A possible alternative would be to use the standard NASA (GSFC) tape recorder which is now in the development stage. This unit is adjustable to various requirements but costs considerably more than present off-the-shelf units. These trades can be made in subsequent phases of this study.

The command decoder will require Command Option Kit #1 to extend the discrete pulse outputs to 144.

The following summarizes the differences between the BSRM baseline and Auroral Option #2 Mission.

SYSTEM PARAMETERS	AURORAL OPTION #2	BSRM CAPABILITIES	BSRM COMPARISON
DATA RATES	17,061 + (114,688 R/T)	32 KBPS + (> 131 KBPS R/T)	EXCEEDS W/ KIT #1
COMMANDS	137 PULSE 7 DIGITAL	144 PULSE 7 DIGITAL	EXCEEDS W/ CMMD KIT #1
DATA STORAGE	190 MINUTES (2 ORBITS)	240 MINUTES	EXCEEDS W/ DATA KIT #1
DATA PLAYBACK	7.5 MINUTES (1 ORBIT)	6.0 MINUTES (1 ORBIT)	EXCEEDS W/ DATA KIT #1

Auroral Option #3 Mission Requirements. Including the (1) Plasma Probe (Heikkila), (2) High Time Resolution Detector (Anderson), and (3) Solar Isotope Separator (Simpson) adds the following requirements:

	TELEMETRY BIT RATE	COMMANDS
1)	872 + (98,304 R/T)	7 PULSE + 1 DIGITAL
2)	1072 + (16,384 R/T)	7 PULSE
3)	106	4 PULSE
	2050 + (114,688 R/T)	18 PULSE + 1 DIGITAL

The 114 kbps real time data can be handled as described above in the Auroral Option #2 Mission.

This option is similar to Auroral Option #2 Mission and the discussion summary for that mission applies. The following summarizes differences between the BSRM baseline and the Auroral Option #3 Mission.

SYSTEM PARAMETERS	AURORAL OPTION #3	BSRM CAPABILITIES	BSRM COMPARISON
DATA RATES	17,167 + (114,688 R/T)	32 KBPS + (>131 KBPS)	EXCEEDS W/ DATA KIT #1
COMMANDS	141 PULSE 7 DIGITAL	144 PULSE 7 DIGITAL	EXCEEDS W/ CMMD KIT #1
DATA STORAGE	190 MINUTES (2 ORBITS)	240 MINUTES	EXCEEDS W/ DATA KIT #1
DATA PLAYBACK	7.5 MINUTES (1 ORBIT)	6.0 MINUTES (1 ORBIT)	EXCEEDS W/ DATA KIT #1

6.2 AETHER DRIFT MISSION

The Aether Drift mission proposes to detect and map the large-angular-scale anisotropies of the 3^o cosmic background microwave radiation with an angular resolution of a few degrees. This would then allow the detection of the motion of the earth with respect to the distant matter of the universe, as well as our overall rotation of the universe.

The mission has also been assumed to include the Solar Isotope Separater experiment which would measure the isotopic and chemical composition and spectra of solar flare accelerated particles from hydrogen to iron.

Table 6.0-1 shows the preliminary requirements for the Aether Drift mission. This mission can be accomplished using the Boeing Small Research Module (BSRM).

6.2.1 CONFIGURATION

The configuration is shown in Figure 6.2-1. The payloads and experiments are located as required to support the requirements of the mission and to achieve the proper thermal and mass properties of the vehicle. The deployable antenna assemblies are omitted and replaced with two fixed booms for the antenna and magnetometer. Additionally, the spin motors and earth sensor are deleted and a scanwheel is added.

The weight of the spacecraft, estimated to be 108 kg, and the payload weight of 29 kg are within the Scout launch vehicle capability for a 500 km, polar orbit.

6.2.2 ELECTRICAL POWER

The load profile for the Aether Drift mission is shown in Figure 6.2-2. Although the mission requires a sun-synchronous, terminator orbit, there is a potential error of about 35^o in the Beta angle due to Scout injection inaccuracies and orbit drift. This results in a possible worst case occultation period of 25 minutes, which significantly impacts the size of the solar array required to meet the power requirements of the mission.

An energy balance analysis indicates that a solar array of 107 watts is needed to meet the load requirements. This again includes a 15% contingency. Assuming a potential 35^o orbit error and a 12 month mission lifetime, the BSRM solar array is capable of producing 129 W. Thus, it would be possible to reduce the solar array size for this mission.

The energy drain on the battery is estimated to be 1.3 Ah which amounts to only a 13% DOD. An alternative would be to maintain energy balance over two or three orbits; thereby reducing the size of the solar array and increasing the depth-of-discharge. At the 13% DOD there is a 300% cycle life margin for a 12 month mission (see discussion in Section 5.1.2).

6.2.3 ATTITUDE CONTROL AND DETERMINATION

The AC&D requirements for the Aether Drift mission can be met, except for the YAW (1.5^o vs. 2^o capability), by using the 3- axis stabilization kit (see section 5.3). This YAW accuracy may not be required and further study will resolve the problem.

NOTES

EXPERIMENTS

(1) MICROWAVE HORNS (ALVAREZ) 46.5 LBS (21 KG)

(2) SOLAR ISOTOPE SEPARATOR (SIMPSON) 16.5 LBS (7.5 KG)

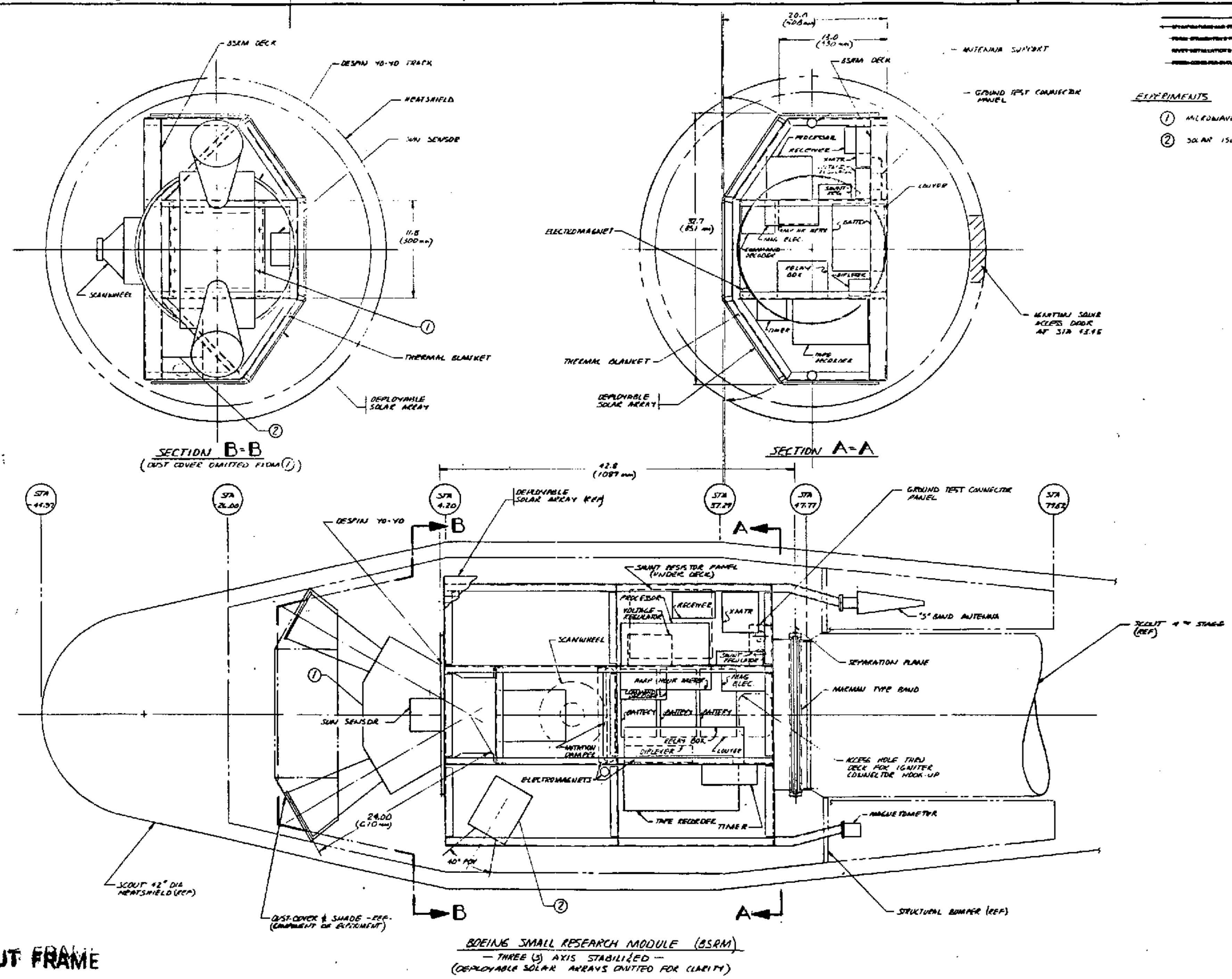

Figure 6.2-1: BSRM Configuration For Aetner Drift Mission

BOEING SMALL RESEARCH MODULE (BSRM) AETHER DRIFT MISSION

NAS 2-8526

J 81205 SK-26806-445

SK-26806-445

REQUIREMENTS:

- 30 WATTS CONTINUOUS (ALVAREZ)
- 1.5 WATTS CONTINUOUS (SIMPSON)

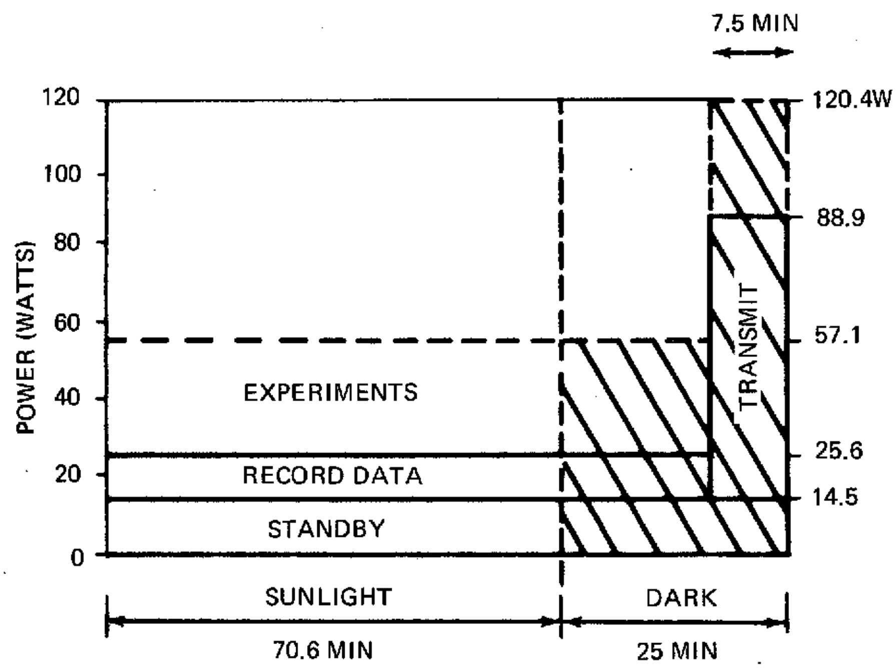

Figure 6.2-2: Aether Drift Mission Load Profile

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6.2.4 TELEMETRY TRACKING AND COMMAND

Data and commands are included for both the Aether Drift and Solar Isotope Separater and are shown in Figure 6.2-3. Operations will be similar to the Auroral Mission in that an orbit time of 95 minutes is again representative. A 7.5 minute minimum pass time again sets the playback time of a single orbit. A two orbit (190 minute) tape recorder storage capacity is assumed. Slant range for this orbit is 1800 km based on 5^0 elevation. Turn-around ranging is also assumed for ephemeris determination.

The baseline TT&C data capacity is much greater than that required; therefore Data Option Kit #2 (Low Data Capacity) will be specified to handle the reduced requirements.

EXPERIMENT	P.I.	TELEMETRY BIT RATE*	COMMANDS
AETHER DRIFT	ALVAREZ	160	12
SOLAR ISOTOPE SEP.	SIMPSON	106	4
		266	16
SPACECRAFT HOUSEKEEPING		1705	60 + 4 DIGITAL
		1971	76 + 4 DIGITAL

*NOTE: 3-AXIS STABILIZATION

Figure 6.2-3: Aether Drift Mission TT&C Requrements

Antenna requirements for a 3-axis stable spacecraft can be met by a single conical log spiral antenna and boom of Figure 6.2-1. The antenna will be mounted and pointed earthward, providing hemispherical coverage. Preliminary analysis indicates that an anomaly during orbital injection could complicate command reception somewhat with the single antenna configuration. However, considering spacecraft spin rate (slow), command word rate ($<$35 milliseconds) and a random tumble, a high probability of command reception is possible. Follow-on phases of this study will clarify this concept.

The following chart summarizes the Aether Drift Mission requirements and BSRM TT&C system capabilities.

SYSTEM PARAMETERS	AETHER DRIFT REQUIREMENTS	BSRM CAPABILITIES	BSRM COMPARISON
STDN COMPATIBILITY	S-BAND	S-BAND	MEETS
TLM & CMMD RANGE (S.R. @ 5^{o} EL)	1800 KM	7000 KM	EXCEEDS
DATA RATES	1971 BPS	6400 BPS	EXCEEDS W/ DATA KIT #2
DATA STORAGE	190 MINUTES (2 ORBITS)	220 MINUTES	EXCEEDS W/ DATA KIT #2
DATA PLAYBACK	7.5 MINUTES (1 ORBIT	4.8 MINUTES	EXCEEDS W/ DATA KIT #2
RANGING	NOT DEFINED	YES	COMPATIBLE W/STDN
COMMANDS	76 PULSE 4 DIGITAL	128 PULSE 7 DIGITAL	EXCEEDS

6.3 OTHER MISSIONS

Boeing utilized company funds and the information information generated in this study (with the permission of NASA-Ames) to evaluate some of the missions which various Principal Investigator (P.I) groups proposed under Announcement of Opportunity for Scientific Definition of Scout Explorer-Class Missions, A.O. #7, dated 15 July 1974. The ten additional missions Boeing evaluated are shown in Table 6.0-1. Preliminary assessment of the mission requirements shows that all of these missions could be performed using the BSRM. Four of the missions would require a structural modification because of the shape of the experimental payload; however, such a modification is within the capabilities and scope of BSRM.

A summary of other possible A.O. #7 missions revealed that in most cases the TT&C requirements can be met simply with Data Option Kit #2 (Low Data Capacity). Wideband analog requirements can be easily handled with Data Option Kit #3 or a derivative thereof, where real time transmission is acceptable. High rate PCM (to 256 kHz) can be transmitted real time provided it is formatted at the experiment package. Both the wideband analog and high rate PCM share the 1.7 MHz subcarrier (Mode I) and only one may be selected at a time.

The missions, for the most part, carry single experiments and do not appear to exceed the capabilities of the BSRM subsystems. However, as the majority of the interaction between Boeing and the P.I's was only by phone, a more detailed assessment would be necessary to verify the ability of BSRM to fulfill all the mission requirements. Such a study should be undertaken.

7.0 LAUNCH VEHICLE INTEGRATION

7.0 LAUNCH VEHICLE INTEGRATION

THE BOEING SRM DESIGN IS EASILY INTEGRATED WITH NUMEROUS LAUNCH VEHICLES. THE DESIGN DOES NOT REQUIRE ANY CHANGES TO THE SCOUT BOOSTER AND MINOR STRUCTURAL MODIFICATIONS FOR LAUNCH PIGGYBACK ON THE DELTA. THE BSRM CAN ALSO BE READILY LAUNCHED ON THE USAF HOST VEHICLE BY RETAINING THE EXISTING S3 GENERAL ARRANGEMENT. ALL BOOSTER INTEGRATION TASKS HAVE BEEN SUCCESSFULLY COMPLETED ON NUMEROUS PREVIOUS SPACECRAFT PROGRAMS AND ENSURE A LOW RISK BSRM INTEGRATION.

The BSRM is very compatible with the boosters examined in this study due to its evolution from the basic S3 satellite. Also, the design modifications described in this study were developed with the specific intention of minimizing launch vehicle mechanical and functional interface changes.

7.1 SCOUT

As discussed in Section 3.1 of this document, the BSRM meets all interface requirements of the existing Scout F launch vehicle. This includes the mechanical attachment, shroud clearances, separation technique and operational features. There is no electrical interface with the Scout.

To mate with the existing Scout fourth stage mounting ring, minor structural modifications to the existing S3 design are necessary including:

- o Deletion of S3 solid rocket maneuver motor mount and rotation provisions.
- o Revision of exterior shape to fit within Scout shroud.
- o Reinforcement of center structural beams to carry boost end loads.
- o Larger diameter V-band attach and separation device to accommodate existing Scout interface.

The configuration resulting from these changes is discussed in Section 3.1.

The first BSRM structure will be subjected to the structural testing to verify the flight qualification status of these changes. Modal survey and static loads tests will be conducted on the first flight structure with mass simulated components and payloads installed. A dummy Scout fourth stage or interface ring assembly will be required to perform a fit check and the structural tests.

Field processing of the BSRM with the Scout is very similar to procedures successfully used on numerous previous spacecraft missions on Thor and Atlas. The detail plan is described in Section 7.2 of D180-18450-3, BSRM Program Definition. The plan is based on performing all prepad operations at NASA Building 836 in South VAFB. Boeing has utilized NASA Building 836 for prepad processing of the STP P72-1 satellite and AFSCF compatibility testing of the three STP S3 satellites. The Boeing-owned Mobile Test Lab was used at the NASA facility for computer-controlled testing of the S3 satellites and is proposed for similar testing of the BSRM satellites. The Mobile Test Lab will be used for on-pad satellite checkout and removed from the pad prior to the final launch countdown.

After receiving inspection in Building 836, the BSRM will be positioned inside a clean room, AGE will be connected and a functional test conducted. After the functional integrity of the BSRM has been verified, the AGE will be disconnected, spin motors will be installed, and ordnance will be connected. Thermal blankets will be closed out, and solar panels will be installed and electrically verified. The BSRM will be reinstalled in its shipping container and transported to the Scout launch complex for mating with the booster.

The BSRM will arrive at the launch complex, be positioned inside the Scout moveable shelter and mated to the Scout fourth stage using the GFE Payload Handling Trailer. The V-Band will be torqued to the required specifications and the AGE will be connected to the satellite. A confidence test will be performed to verify no damage has occurred to the BSRM during the transportation and handling operations. This test will exercise the BSRM subsystems and experiments to the extent necessary to verify aliveness. After the completion of this test, the AGE will be disconnected and removed from the launch complex. Prior to heat shield installation sensor protective covers, solar panel protective covers and safing pins will be removed and final inspections of the BSRM will be accomplished.

The Scout booster contractor has been contacted in the completion of this study and numerous documents reviewed. The integration plan for BSRM is compatible with all launch vehicle requirements and relies extensively on Boeing past experience with launch vehicle integration. No changes are required to the existing Scout to accommodate the BSRM providing high confidence of an efficient and low risk integration.

7.2 DELTA

To assess the feasibility of integrating the BSRM on the Delta launch vehicle, a brief performance analysis was completed. The payload capability of the Delta 2910 Configuration was defined and the ability to carry BSRM spacecraft as piggyback or as primary payloads was determined. Figures 7.2-1 and 7.2-2 show the basic Delta polar orbit performance analyzed. Both two- and three-stage performance was considered to assess the various alternatives for the piggyback mode as discussed below.

The Delta 2910 Configuration is apparently the largest version flown to date. Nine strap-on solid rockets are used for maximum payload capability. For missions where the BSRM is the primary payload, other versions of the Delta could be used if the performance of the 2910 Configuration is not required. The integration and interface requirements imposed on the BSRM design would be almost identical to those discussed herein regardless of the Delta configuration selected.

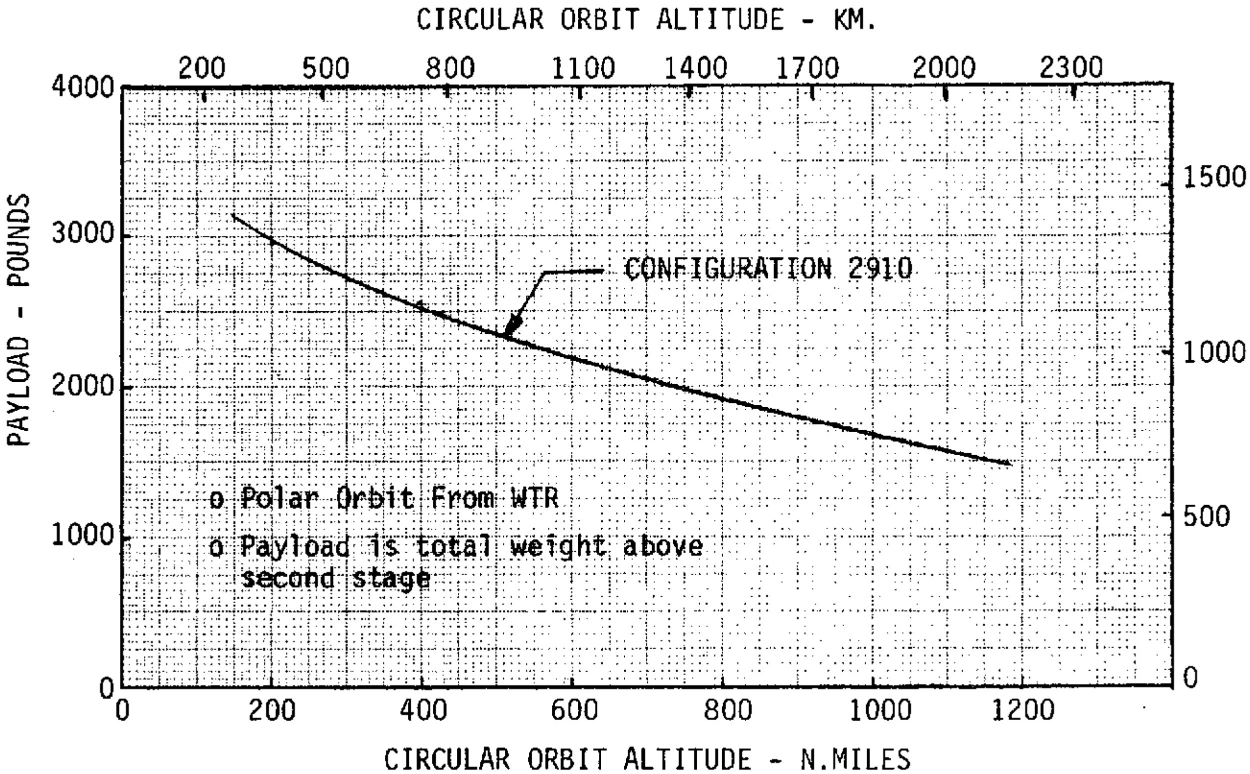

Figure 7.2-1. Delta Two-Stage Performance

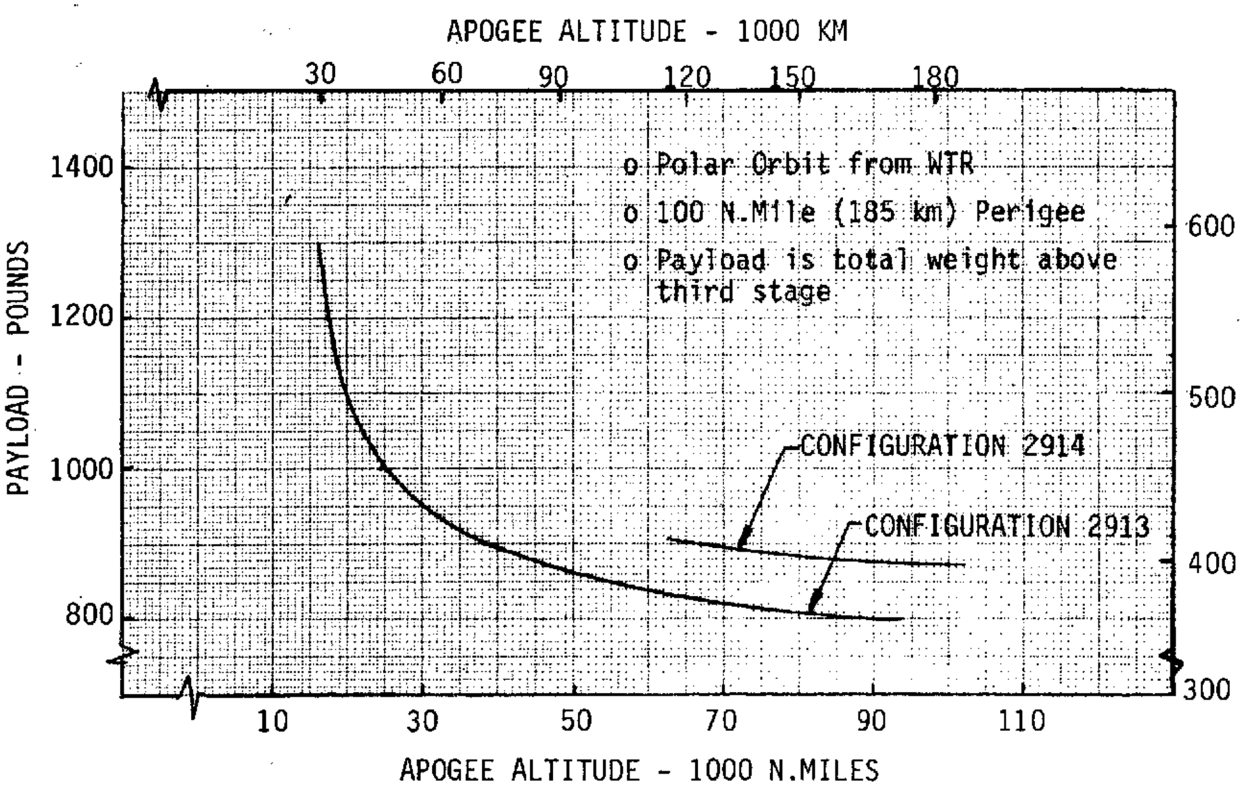

Figure 7.2-2. Delta Three-Stage Performance

7.2.1 PIGGYBACK LAUNCH

The BSRM can be carried on either the primary payload or on a booster upper stage for piggyback missions. To minimize the interface problems with a primary, higher priority payload it is believed that only piggyback on booster stages would be considered a viable approach. Boeing experience with the S3 satellite, which is carried piggyback on a USAF vehicle, indicates that the government agency responsible for the primary payload imposes severe constraints on piggyback spacecraft design requirements to ensure there is no impact on the primary mission. On S3 these constraints included:

- o Extremely high structural stiffness requirement to eliminate any coupling with the primary vehicle. A large weight penalty is thereby incurred by the piggyback spacecraft.
- o Double redundancy for all separation devices and functions to absolutely ensure jettison from the primary vehicle. Weight and complexity are increased on the piggyback spacecraft.
- o Redundant lockouts on such items as motor ignition and boom deployments on the piggyback vehicle to absolutely preclude their operation while still attached to the primary payload. Weight and complexity are increased on the piggyback spacecraft.

Integration as a piggyback spacecraft with booster upper stages would relieve many of these constraints and result in a lower cost, more efficient BSRM design. Figure 7.2-3 shows an approach for a two-stage Delta booster. The BSRM spacecraft are stowed around the primary payload. To maximize the primary payload envelope within the shroud dynamic envelope, the existing S3 general arrangement is used including the V-band attachment, ejection system and rocket motor installation flight proven on the recent S3-1 launch. Except for the slightly different shape, this version of the BSRM is identical to the subsystems, components and operational features discussed in Section 3 for the Scout launched BSRM.

The structural attachment is made to the second stage adapter to minimize structural changes to the booster. The BSRM spacecraft must be jettisoned prior to primary payload separation to provide the necessary separation clearance.

Three BSRM spacecraft are shown for the piggyback arrangement in Figure 7.2-3 to depict a typical launch. Depending on the excess performance available, any number of BSRM up to four can be physically accommodated. For a single BSRM, the primary payload cg would be offset to balance the total booster configuration or the booster would be programmed to accept the adverse cg created by the single BSRM. (The USAF launch vehicle used to launch the existing S3 has the latter capability.)

The Delta launch is two-stages to a low earth parking orbit for this piggyback mode. A sequence is then initiated to jettison the BSRM spacecraft. Rocket motors in the BSRM then ignite to achieve the required BSRM orbit as successfully demonstrated by the S3 spacecraft. The Delta parking orbit can be

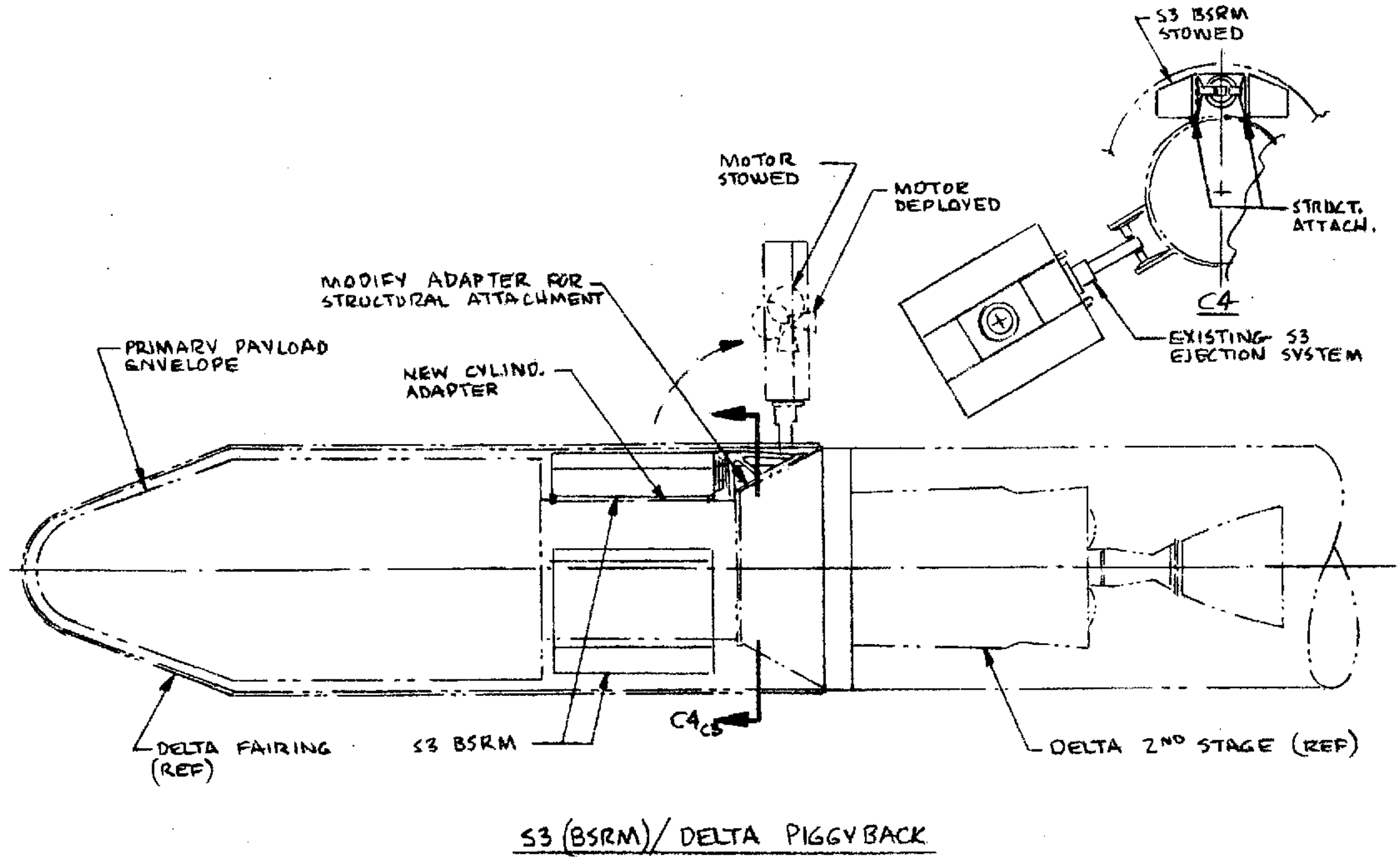

Figure 7.2-3. Two-Stage Delta Piggyback Configuration

elliptical and by proper selection of the jettison time, the BSRM can be put into a circular orbit. If a low circular parking orbit is used, an elliptical BSRM orbit is achieved. Also, two solid rocket motors can be installed on the BSRM to achieve a circular orbit different from the Delta circular parking orbit.

A variety of small solid motors is available permitting a wide selection of BSRM final orbits. On the existing S3 spacecraft, different rocket motors are installed on each of three satellites to achieve three different orbits from a common host vehicle parking orbit.

A detailed performance analysis, beyond the scope of this study contract, is required to define all the orbit alternatives possible in a piggyback mode. A definition of candidate primary payload mission requirements is necessary to undertake such an analysis. However, a simple example can be presented using the data in Figure 7.2-1 as follows.

The Delta two-stage performance for polar orbits, shows that if two BSRM are launched piggyback from WTR, the primary payload left is approximately 1700 pounds (772 kg) in a 200 n.mile (370 km) circular orbit. The BSRM orbits shown in Table 7.2-1 would be achieved using the solid rocket motors installed on the S3 spacecraft as listed in the table. It can be seen from this brief example, that a piggyback mode can undertake a variety of scientific missions from one booster launch. Since low inclination orbit performance is greater than that used in this example, more capability is available for launches from ETR permitting a larger primary payload, higher orbits, or additional piggyback BSRM.

TABLE 7.2-1

EXAMPLE BSRM FINAL ORBITS

BSRM	Motor	Orbit
-1	TE-M-479	200 x3200 n.miles (370 x1750 km)
-2	TE-M-516	200 x1510 n.miles (370 x 800 km)
-3	TE-M-521	200 x6000 n.miles (370 x3250 km)

Launching BSRM piggyback on a three-stage Delta configuration was also considered in this study. Figure 7.2-4 shows a possible configuration. The BSRM are supported from the adapter below the third stage. Stowed around the third stage motor, the BSRM spacecraft do not interfere with the primary payload envelope. The performance analysis of this configuration showed that the Delta booster cannot achieve a meaningful primary payload mission with the existing third stage. Because the BSRM must be jettisoned from a parking orbit, the Delta two-stage performance of Figure 7.2-1 must be analyzed. Subtracting approximately 2700 pounds (1226 kg) for the third stage leaves only 200 pounds (90 kg) in a 200 n.mile (370 km) circular orbit which is inadequate for the BSRM or primary payload. Jettisoning the BSRM from a non-orbital coast trajectory between second and third stage burns may improve the performance, but

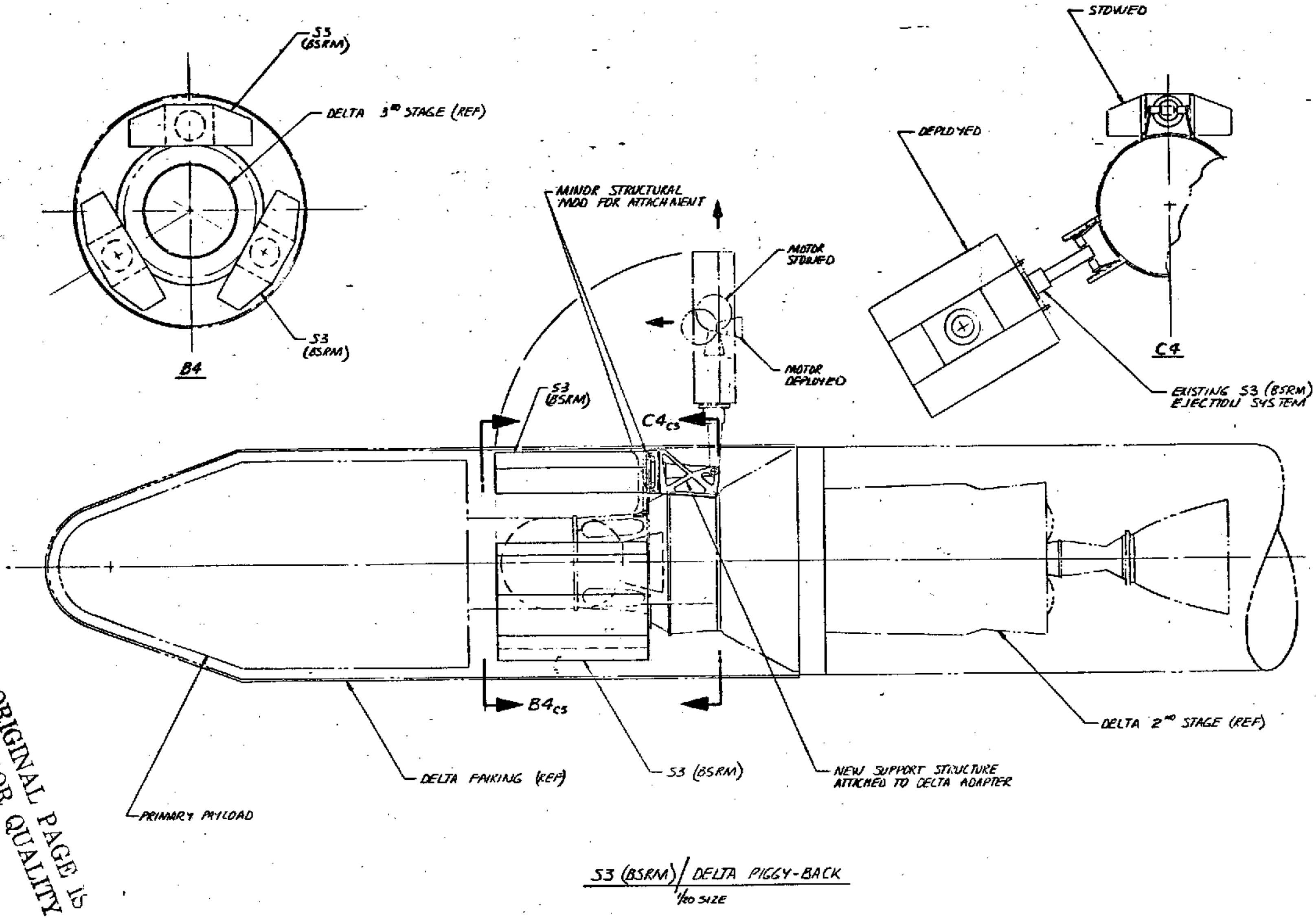

Figure 7.2-4. Three-Stage Delta Piggyback Configuration

there does not appear to be a significant performance capability of interest with this configuration and it was therefore discarded from further study.

7.2.2 Prime Payload Launch. If a Delta vehicle is dedicated to a BSRM program launch, high energy orbits and/or multi-satellite launches can be achieved. Figure 7.2-1 shows that the two-stage Delta can launch multi-BSRM satellites into circular low earth orbits from WTR. Highly elliptical orbits for multi-BSRM spacecraft can be achieved with the three-stage Delta as shown in Figure 7.2-2. By proper selection of jettison time and BSRM solid rocket motor, a wide variety of spacecraft orbits can be obtained with the Delta launch vehicle.

The synchronous payload capability of the three-stage Delta is 1540 lbs (699 kg) onto the transfer ellipse. After circularization and plane change by an apogee motor, the useful payload in the synchronous equatorial orbit is approximately 700 pounds (318 kg).

Figure 7.2-5 shows a candidate BSRM/Delta configuration with two BSRM spacecraft. This figure shows the three-stage Delta Configuration 2913 but a similar general arrangement is possible with the two-stage version.

A structural truss interfacing with the top of the appropriate Delta stage provides the support for the BSRM spacecraft. Existing S3 separation devices are used to jettison the BSRM satellite from the support structure. Rocket motors could be installed in any or all BSRM spacecraft to achieve final orbits different from the Delta parking orbit.

7.3 HOST VEHICLE

Integration of the BSRM with the USAF Host Vehicle for piggyback launch has already been completed and verified by the successful launch of the S3-1 satellite. The existing S3 general arrangement, ejection system and rocket motor installations would be retained for BSRM. All interfaces are proven and documentation, procedures, ICD's, etc., are available. This approach represents the minimum change to the S3 system for the BSRM program.

Figure 7.3-1 depicts the basic S3 installation on the USAF Host Vehicle. Room is available around the periphery of the S3 vehicle for stowage of booms and other appendages. Components and/or payloads can be mounted exterior to the basic S3 structure as long as the Host Vehicle dynamic envelope is not exceeded. There is no problem installing BSRM subsystems in the S3 general arrangement as the S3 was the basis for developing the BSRM spacecraft discussed in Section 3.

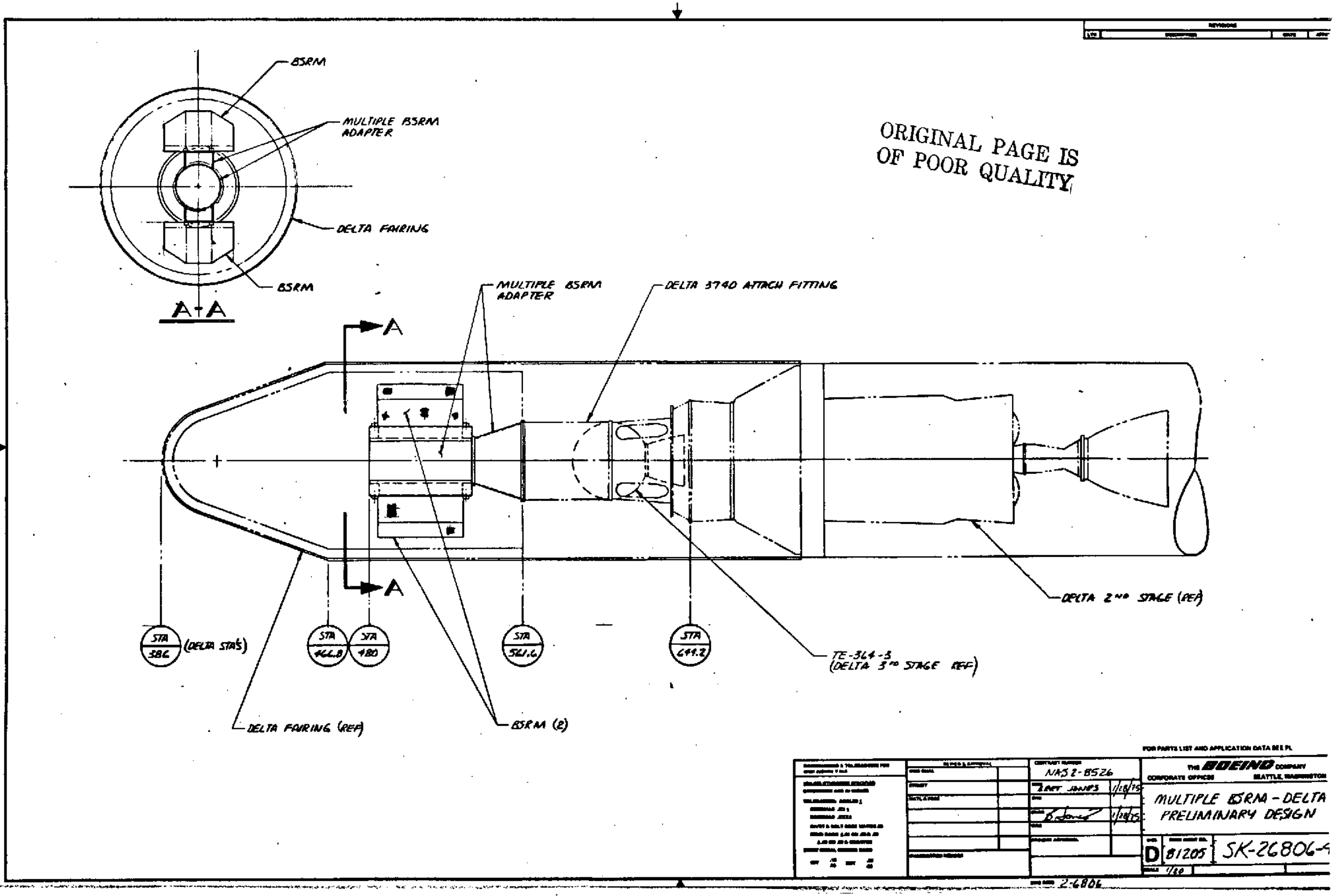

Figure 7.2-5. BSRM/Delta Primary Payload Configuration

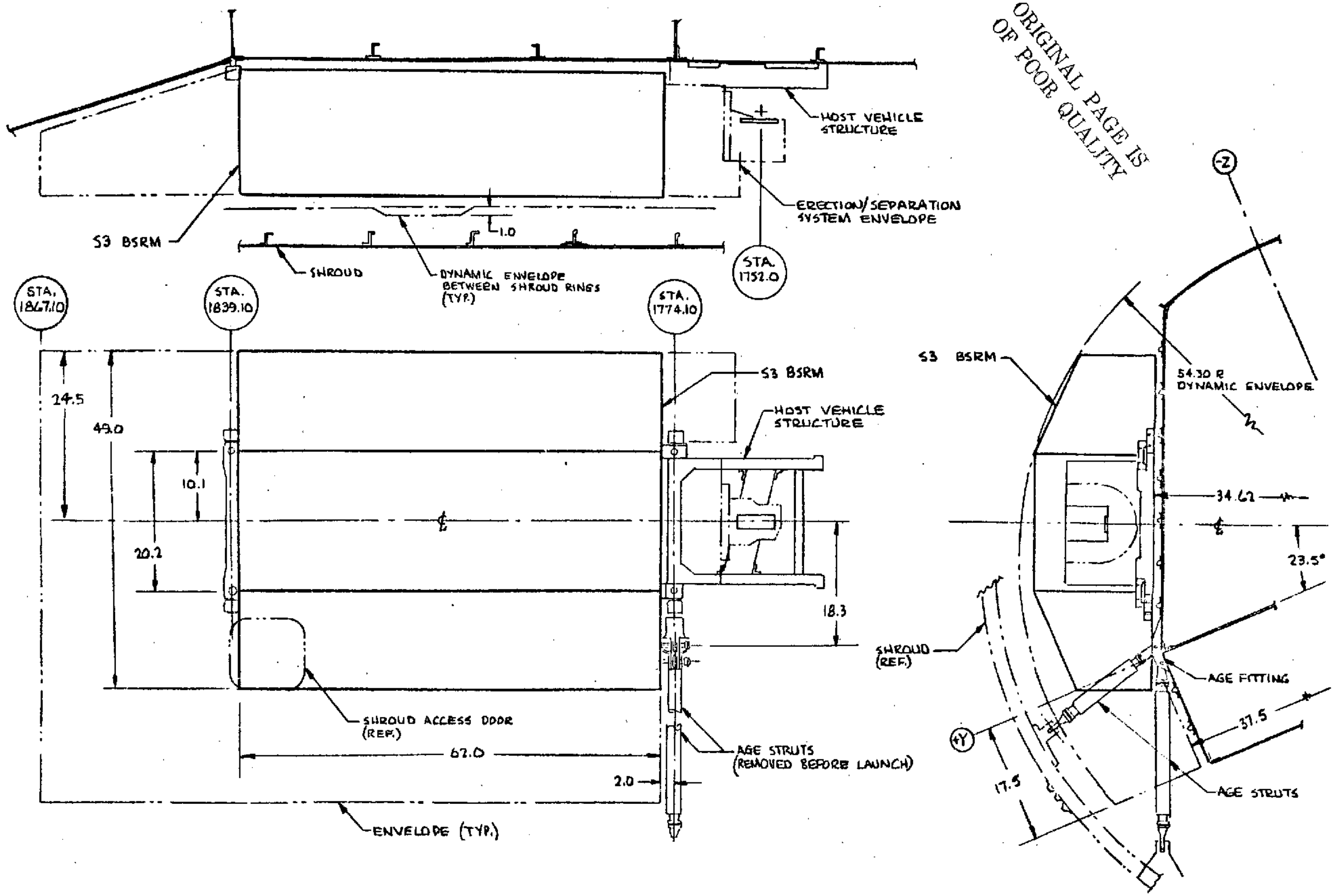

Figure 7.3-1.   BSRM Installation on USAF Host Vehicle

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FINAL REPORT

FEASIBILITY STUDY OF THE BOEING SMALL RESEARCH MODULE (BSRM) CONCEPT

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