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SPACECRAFT CONTROL SECTION FOR THE IMPROVED SMALL ASTRONOMY SATELLITE (SAS)

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Abstract

SPACECRAFT CONTROL SECTION FOR THE IMPROVED SMALL ASTRONOMY SATELLITE (SAS)

The upgraded spacecraft control section for the Small Astronomy Satellite (SAS) incorporates a remarkable amount of flexibility for a small satellite. Able to point its thrust axis to any direction in space, it can also spin or slow its outer body rotation to zero for star-locked pointing of side viewing experiments. A programmable telemetry system and delayed command system enhance the inherent capability of a spacecraft designed to be used for a variety of experiments, each of which can be built independently and attached just prior to final acceptance testing and launch.

The design of this new spacecraft, whose first launch is scheduled for 1975, is provided in sufficient detail to permit the reader to ascertain its suitability for specific experiments. Earlier versions have been used for SAS-1 (also known as Explorer 42 and Uhuru) which was launched for the United States by the Italian government from their San Marco platform on December 12, 1970, with an X-ray astronomy payload; and SAS-2 (Explorer 48), launched from San Marco on November 15, 1972, with a digitized spark chamber gamma-ray telescope.

A summary of the spacecraft characteristics, project reliability requirements, and environmental test conditions are included in the appendices.

Foreword

In the last ten years, there has been a truly remarkable flowering of high energy astrophysics including both X- and γ-ray astronomy. These high energy photons relate directly to the highest energy celestial phenomena in our galaxy and the most energetic physical processes in the universe.

The Small Astronomy Satellite (SAS) series is playing an important role in this new astronomical field. SAS-1 (Uhuru) launched in December 1970, carried into space an X-ray experiment which has now completed a full sky survey. Over one-hundred and sixty X-ray objects have already been identified, most having a totally unexpected nature. Some have variations which consist of a combination of periods and still others are seen to have burst of X-rays. SAS-2, launched in November 1972, was aimed at providing a survey of the celestial sky in gamma rays. A diffuse high energy γ-ray flux with a very steep energy spectrum has been observed, and detailed high energy photon data have been accumulated for a study of the galactic plane structure and point sources. The third satellite in the SAS series will carry a set of X-ray experiments to study in detail, over a wide range of energies and with finer time resolution than SAS-1, the types of objects emitting X-rays. Also, a star-locked pointing capability to be tried experimentally on the SAS-C spacecraft should enhance greatly the sensitivity of X-ray experiments for selected objects by permitting studies for longer continuous periods of time.

In general, the improved Small Astronomy Satellite now provides an Explorer class satellite with substantial flexibility in pointing, telemetry, commanding, and experiment type. The new SAS is, therefore, able not only to fill the gap between small or short duration experiments in many fields of astronomy and the very large experiments flown on observatories, but also to provide considerable capability for the specialized second generation experiments now under consideration.

Carl L. Fichtel
SAS Project Scientist

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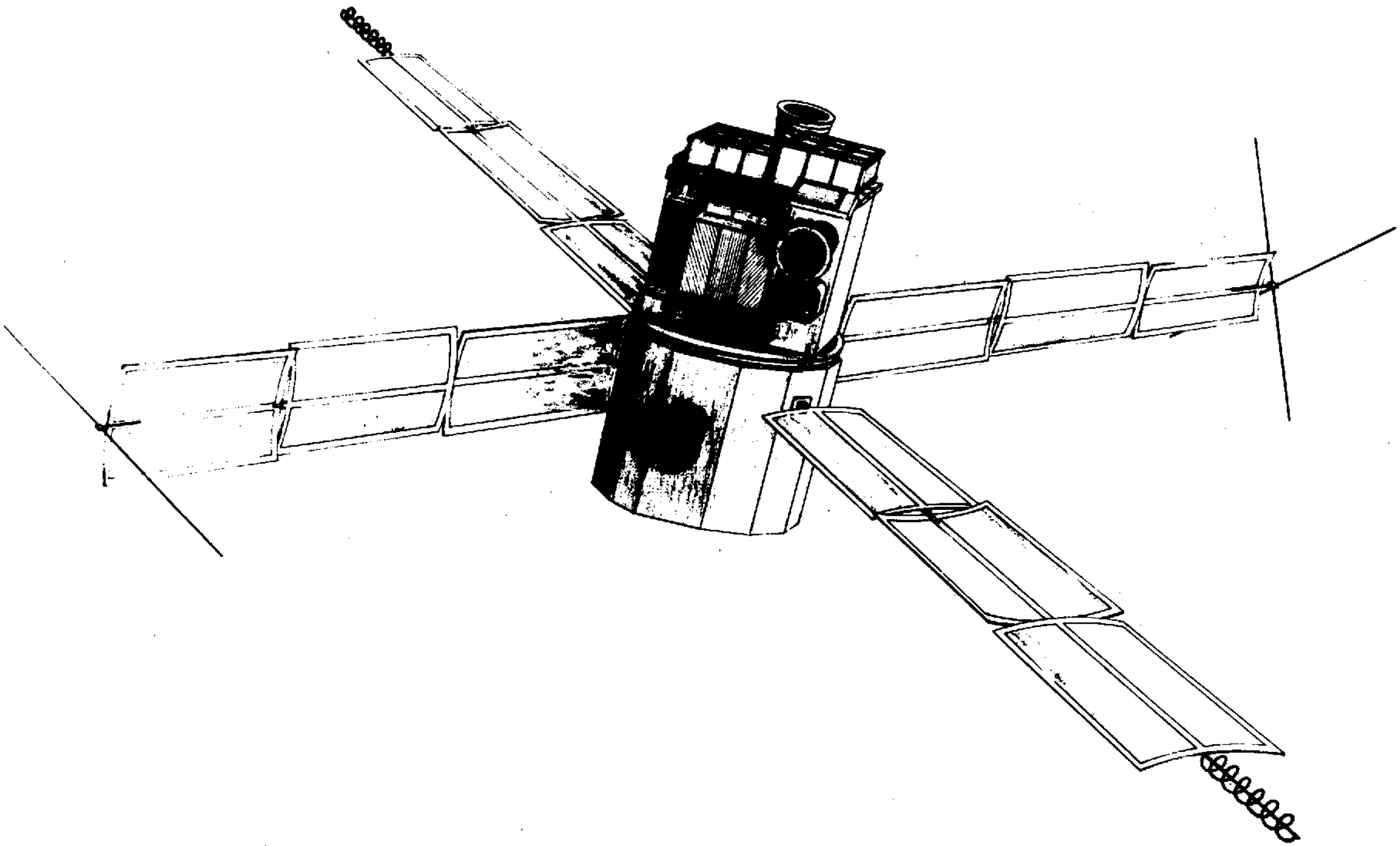

Frontispiece - SAS-C in Orbital Configuration

SPACECRAFT CONTROL SECTION FOR THE IMPROVED SMALL ASTRONOMY SATELLITE (SAS)

by

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INTRODUCTION

The objective of the Small Astronomy Satellite (SAS) program is to study the celestial sphere above the earth's atmosphere and to search for sources radiating in the X-ray, gamma-ray, ultraviolet and infra-red regions of theelectromagnetic spectrum, both inside and outside of our galaxy. Since the launch of the first of this series (SAS-1, Explorer 42, Uhuru) on December 12, 1970, the information on X-ray sources has increased immeasurably because the spacecraft has both found new sources and has obtained significant new information on the sources already known. The second in the series (SAS-2, Explorer 48) was launched on November 15, 1972, and has already collected significant data on celestial gamma radiation. It is hoped and planned that future spacecraft in the series will contribute as much and more to science.

As early survey-type experiments are concluded, it is inevitable that follow-on experiments will require additional capability from the spacecraft control section that provides the basic support needed for a space experiment, e.g., power, telemetry, commands, and control. Concurrent with these increased requirements, launch vehicle capability has been upgraded since 1970, allowing heavier, more complex experiments on low-cost vehicles. For these reasons, it was appropriate to redesign the SAS spacecraft control section for increased flexibility and provide it with the capability of flying a diverse variety of small payloads without making any changes in the spacecraft control section between missions. The design allows an "off-the-shelf" availability, wherein only the transition section between the spacecraft control section and the experiment must be fabricated to interface with any experiment configuration.

OVERALL SPACECRAFT DESIGN

The spacecraft (shown in Figure 1 in the SAS-C configuration inside the Scout heatshield) is designed to have a clean interface at STA 13.27 between the basic spacecraft control section and the experiment payload, both mechanically and thermally. Only the experiment-spacecraft transition section will be changed between missions. This will provide maximum flexibility at minimum cost, since the height of this transition section, at the top of the spacecraft control section shown in Figure 2, can be changed to allow for mission-unique hardware on the upper deck. Also conveniently located on the upper deck are the experiment-spacecraft interface connectors (Figure 3). For low, earth-orbiting missions, such

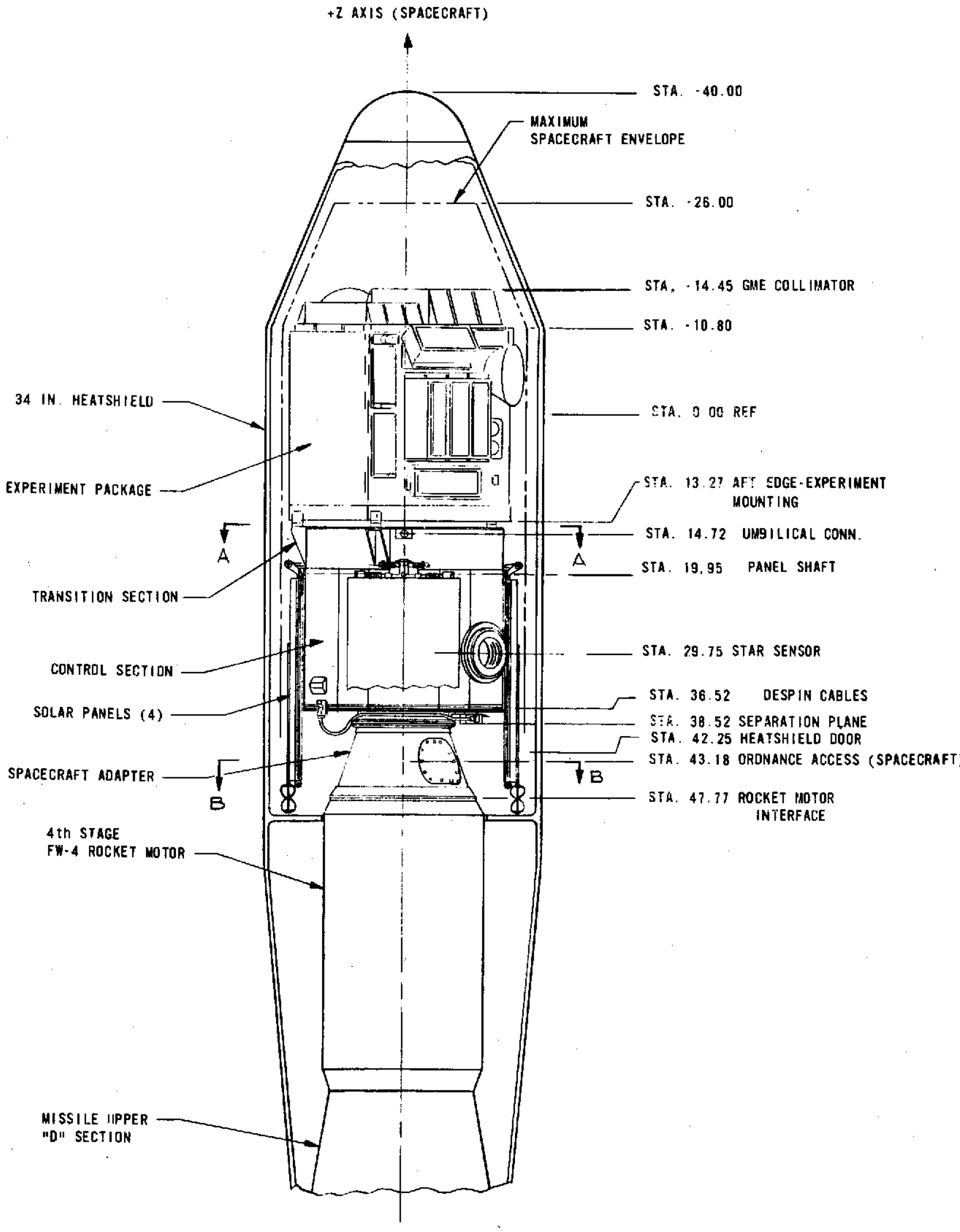

FIG. 1
SAS-C IN LAUNCH CONFIGURATION
INSIDE SCOUT HEATSHIELD

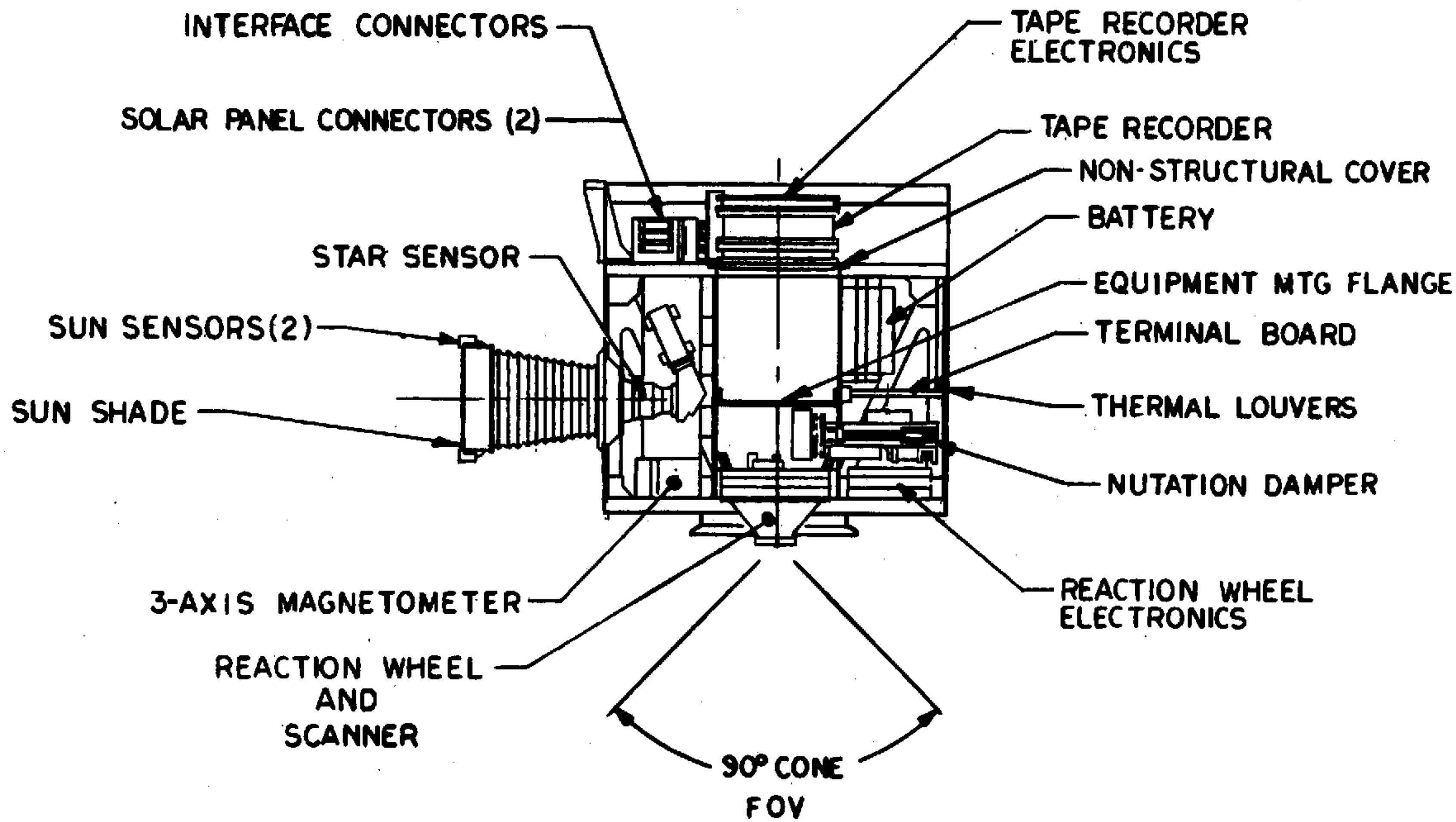

FIGURE 2 IMPROVED SAS SPACECRAFT CONTROL SECTION

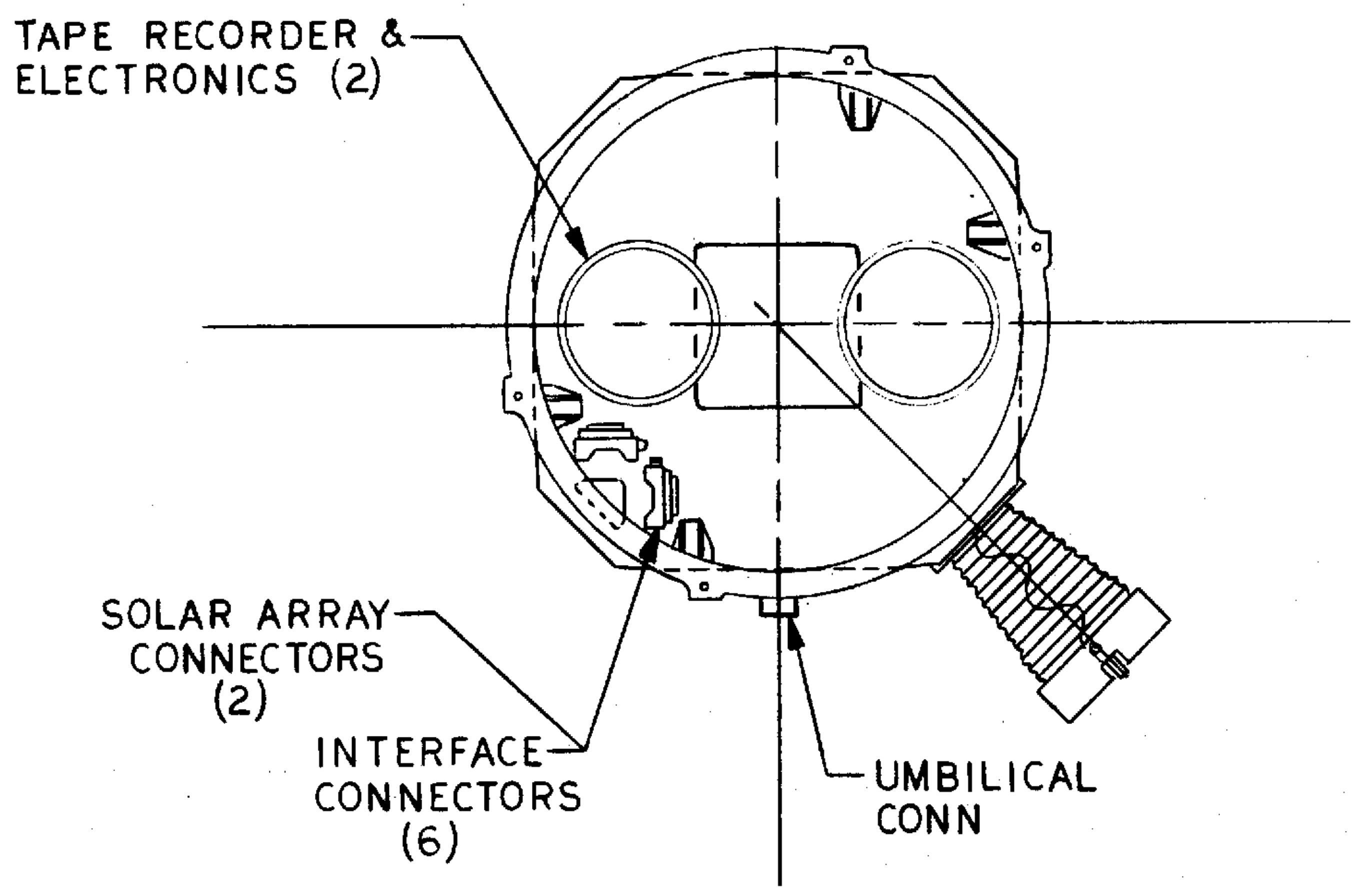

FIG. 3
PLAN VIEW OF UPPER DECK
OF IMPROVED SAS SPACECRAFT
CONTROL SECTION

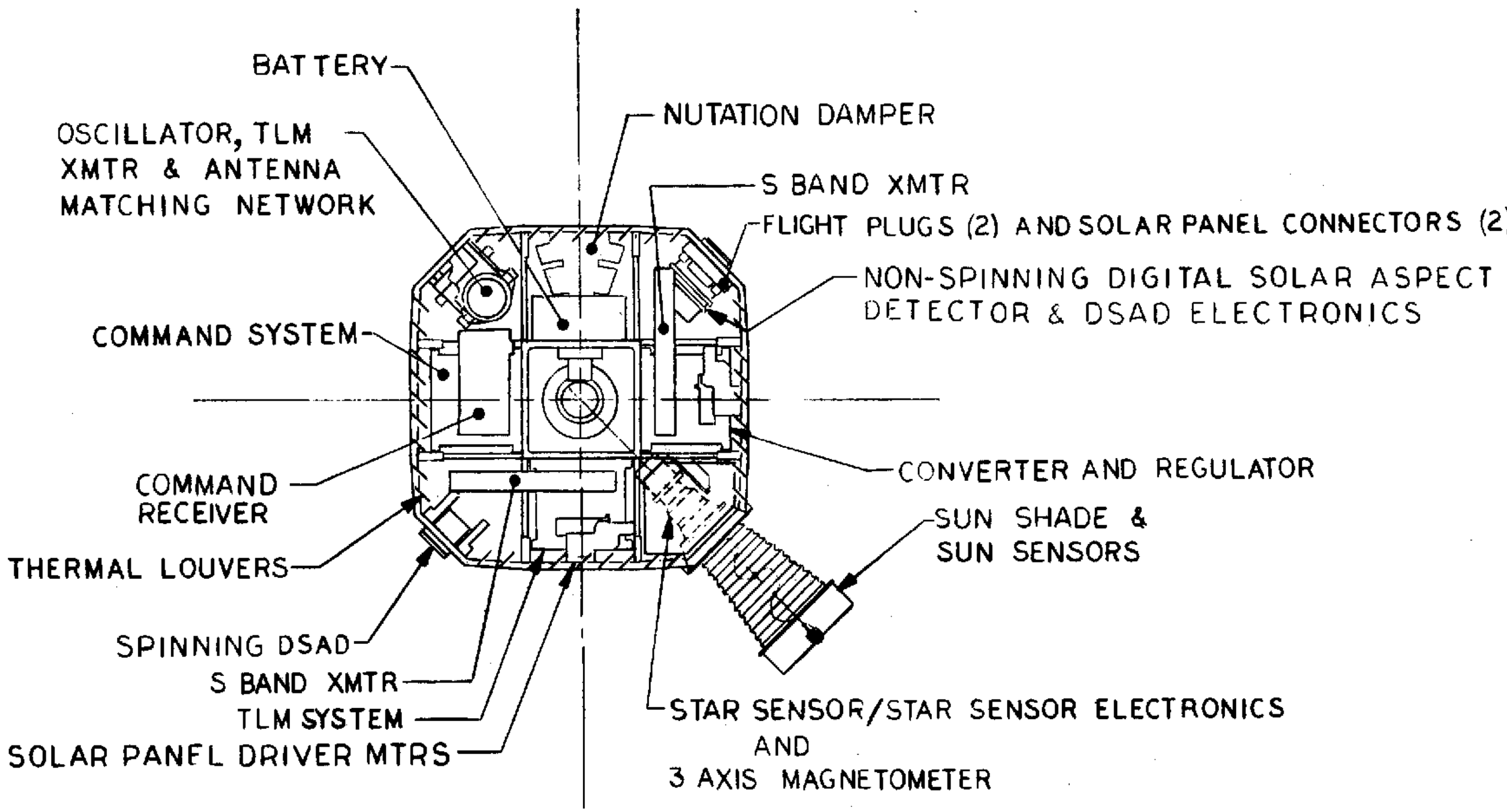

FIG. 4
PLAN VIEW BELOW UPPER DECK
OF IMPROVED SAS SPACECRAFT CONTROL SECTION

as SAS-C, redundant tape recorders will be located on this deck. For pointed missions, two star tracking cameras and two small reaction wheels can be added; the third camera, which would point along the thrust axis must be co-located with the experiment for accuracy. For synchronous orbit missions, a gas system for attitude control and station keeping would replace the tape recorders and cameras. The upper deck has been designed to support this gas system, which would thus be ideally situated near the center of gravity of the spacecraft. An apogee motor for circularizing the orbit would be located at the base of the spacecraft. For these synchronous orbit missions, space constraints would force all star cameras to be co-located with the experiment. For the various missions thus described, all changes are made only to hardware on the upper deck. Therefore, without any changes to the basic spacecraft control section, this design, normally planned for a Scout-launched low earth orbit, could be adapted for a Delta-launched synchronous orbit.

Inside the lower part of the thermally controlled spacecraft control section (Figure 4) are the systems requried to provide the basic spacecraft functions: a rechargable battery with its charge control and regulator systems; redundant command receivers and decoders with a stored command capability; a programmable telemetry system; VHF and S-band transmitters; magnetometers, sun sensors, and a star sensor for attitude determination; and a magnetically-torqued commandable control system which can point the spacecraft to any point in the sky and vary its spin rate. Stability is provided by a reaction wheel and an active nutation damper.

Hinged to the control section are four sets of solar array panels (Frontispiece) whose purpose is to provide power for the spacecraft. Each set of panels is comprised of three sections which fold against each other (see Figure 1) and are held in place by despin cables during launch. In orbit, a timer releases the cables, despinning the spacecraft and allowing the solar panels to deploy and unfold into their orbital configuration. Then, either set of opposing panels can be rotated through a 90° arc by an eight watt synchronous motor operating at 491 Hz. Each of the two motors drives one paddle directly, and the opposing paddle through a cable threaded through channels in the upper deck.

Inside a radiative outer shell, louvers, in conjunction with multilayer insulation and heaters, provide thermal control. Wound around the periphery of the lower deck is the Z-axis torque coil. The X- and Y-axis torque coils are wound vertically 90° apart.

Interconnections are provided by a spacecraft harness. The design permits the harness below the upper deck to remain the same from mission to mission. If any changes should be required between missions, the only effect is to the upper deck harness. Another feature for future missions is the hollow center of the spacecraft structure seen in Figure 2. In the middle 20 centimeter square section are the reaction wheel and part of the nutation damper. Above these, which are located at the lowest part of the spacecraft control section, a volume 20 centimeters square by about 30 centimeters high is kept clear for future missions which might require a telescope longer than the space available inside the Scout heatshield above the spacecraft-experiment interface. For this purpose, a removable 20 centimeter square panel has been built into the upper deck, and all cabling for solar panel rotation has been routed around this center section. The tape recorder and star camera positions keep this center aperture clear.

Details of the design of the SAS Spacecraft Control Section follow.

STRUCTURAL AND THERMAL DESIGN

Weighing about 120 kilograms, the basic structure of the SAS spacecraft control section is a cylinder 66 centimeters in diameter by 61 centimeters high, with upper and lower decks made of aluminum honeycomb. Loads, generated by the experiment attached to the transition section and components located on the upper deck, are carried down the center column and through hollow struts from the periphery of the transition section down to the base of the center column, from which point the loads are carried through the flared adapter section seen in Figure 1 to the upper stage of the launch vehicle. Most of the structural members are aluminum, with the sides of the center column made of Lockalloy, an alloy of aluminum and beryllium. If the weight is evenly distributed, experiments up to 180 kilograms can be supported by this structure. Specific experiment weights for various orbits can be found in Appendix A, but SAS-C can be used as an example of an experiment launched by a four stage Scout vehicle from the San Marco equatorial launch platform into a 550 kilometer circular orbit. It weighs about 75 kilograms. The available volume can be determined roughly by looking at Figure 1. The inside of the Scout heatshield is about 75 centimeters in diameter up to STA-10.00. The height above the SAS transition section is about 60 centimeters. Within the maximum spacecraft envelope, another 40 centimeters in height is available with a decreasing diameter.

Three of the four rectangular bays, seen from the top in Figure 4, contain electronics for the command, control, telemetry and power systems. This circuitry is packaged in stacks of magnesium castings approximately 16 x 16 x 5 centimeters each. The fourth bay contains the battery and the nutation damper, part of which extends into the center column. At the bottom of the center column is the wheel, whose infra-red sensor's field of view is out the base of the spacecraft. Mounted in the four corner sections are, variously, the star sensor, the crystal controlled oscillator, the three-axis magnetometer, the VHF transmitter, and the digital solar aspect detectors.

Around the periphery of the spacecraft control section, behind thin radiator panels, are louvers for active thermal control. Although the thermal design of the total spacecraft could be integrated with the experiment, it is preferred to maintain the present design, isolated from the experiment by a multilayer blanket and fiberglass bolts.

This assures that true "off-the-shelf" availability can be maintained. Thermal louvers made of thin sections of polyurethane covered with gold coated Kapton are opened and closed by bimetal actuators whose positions are determined by the ambient temperature. Banks of these three centimeter wide louvers are located in front of each of the four bays, facing silverized Teflon radiators. Multilayer insulation covers the top, bottom and four corner sections of the spacecraft control section. Heaters, internal to the control section, can be used, if necessary, and as a backup if any of the thermal louvers become stuck in the open position. However, the current design does not need heaters to maintain the spacecraft control section temperature within the range of +5° to +35°C. to meet the constraints for the battery (0° to +35°C), the tape recorders (+5 to +35°C.) and the electronics (-10° to +45°C.).

The design, without any changes, provides adequate thermal control in low earth orbits with inclinations anywhere from equatorial to polar, in synchronous orbits, and at any attitude in any of these orbits. The surface coatings assumed in this design are 5 mil silverized Teflon ($\alpha = 0.15$, $\varepsilon = 0.80$) for the space radiators, aluminized mylar ($\alpha = 0.15$, $\varepsilon = 0.80$) for the multilayer blankets, gold-coated Kapton ($\alpha = 0.04$) for the thermal louvers, and Martin hardcoat ($\alpha = 0.90$, $\varepsilon = 0.90$) for the attach ring which protrudes through the multilayer blanket. For the purposes of this design, the internal heat loads were varied from 38 to 82 watts. The set point of the bimetal actuators operating the thermal louvers was assumed to be +10°C.

The solar array panels are of thin aluminum honeycomb construction, curved to fit within the limitations of the Scout heatshield and covered with a thin film to insulate the aluminum from the solar cells. Each of three panels is approximately 30 x 180 centimeters. Antennas for the VHF command system and the VHF and S-band transmitters are located at the ends of the outer panels, which also have brass weights to provide a sufficient moment of inertia to prevent tumbling in case of a failure in the wheel system. These weights, the antennas, and necessary cabling are the only loads on the structure of these panels except for the solar cells and their cover slides.

POWER SYSTEM

The power system, shown in block diagram form in Figure 5, consists of four rotatable paddles with solar cells on both sides; redundant charge regulating and monitoring circuitry (CRAM); an eight ampere-hour hermetically-sealed, rechargeable, nickel-cadmium battery; and various regulators and converters. Orbit average power is a function of altitude, orientation, and battery size. SAS-C, in a 550 kilometer, circular, equatorial orbit, with a requirement for random orientation, represents a worst-case condition for this power system. For the purposes of the curves shown in

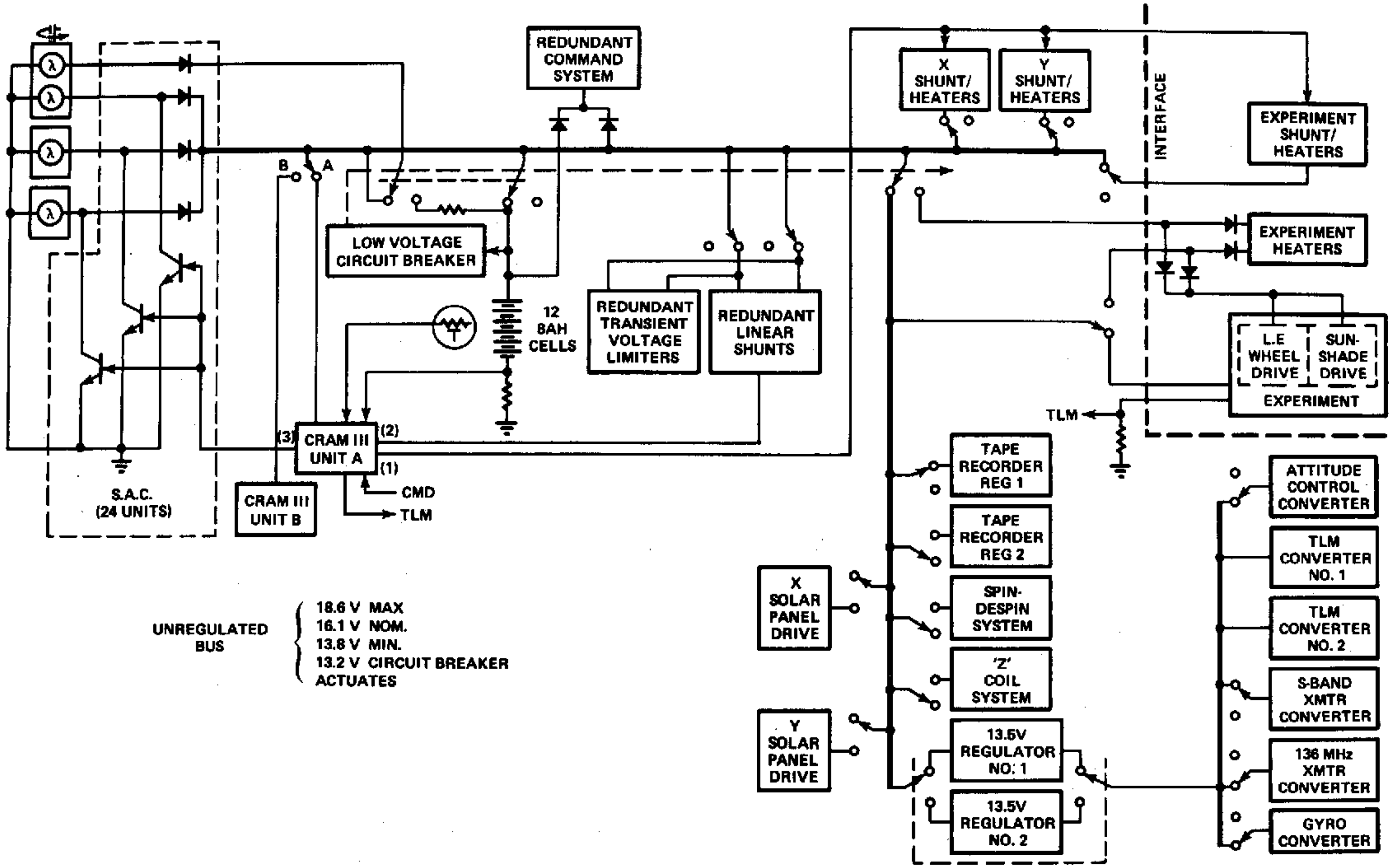

FIGURE 5 SAS POWER SYSTEM - BLOCK DIAGRAM

Figure 6, it was assumed that the two sets of solar paddles would be either parallel or at 90° with respect to each other, and that earth albedo is 10 percent. Under these conditions, calculations show that the power system will provide a minimum of 65 watts averaged over the entire orbit. This minimum power and the length of the nighttime period defines the size of the battery required for the mission. Higher average powers can be obtained for missions at higher altitudes, by restricting orientation to more favorable sun angles, and by optimizing paddle orientation between the 0° and 90° limits set for this calculation. Stored power for the nighttime portion of the orbit or for peak requirements during the day can be increased by 50 percent by changing to a 12 ampere-hour battery, without making any other changes to the system. To plan for this possibility, adequate volume has been reserved for this larger battery. Figure 6 shows the variation in available current as a function of sun angle, based on the assumptions given above. The spacecraft control section requires about 40 watts for normal operation, leaving a minimum of 25 watts for the experiment. If additional requirements are imposed on the control section, for example, star cameras for a pointed mission, this power requirement must be subtracted from the experiment allocation.

Each side of the thin solar cell substrates is covered with N/P silicon solar cells measuring 2 x 2 centimeters and having a nominal resistivity of 10 ohm-centimeters. The higher resistivity was chosen so that the design could be used in any orbit, not just the low-radiation equatorial orbit required for the SAS-C mission. The solar cells have an anti-reflectivity coating and are covered with 0.015 centimeter thick cover slides which have a blue coating on the cell side and an anti-reflectivity coating on the outside.

The Charge Regulator and Monitor (CRAM) circuit serves as a coulometer which measures and replaces the energy taken from the battery by the load. After the battery is fully charged, the current to the battery is reduced to a trickle charge level by diverting the current received from the solar arrays. If spacecraft temperatures are low, this can be used in heater circuits; if not, the outputs from the solar arrays can be switched off entirely, with only the linear shunts, acting as a vernier, dissipating power within the spacecraft.

Raw power at 16.1±15% volts is provided to the experiment and to the spacecraft control section. Redundant regulators drive converters that provide power to the attitude system, the telemetry system, the VHF and S-band transmitters, and the gyro package. The redundant command system has its own regulated converters. Separate regulators are also provided for the solar panel drive system inverters and the tape recorders. The spin-despin and Z-axis torquing coils are run from the unregulated buss for maximum efficiency.

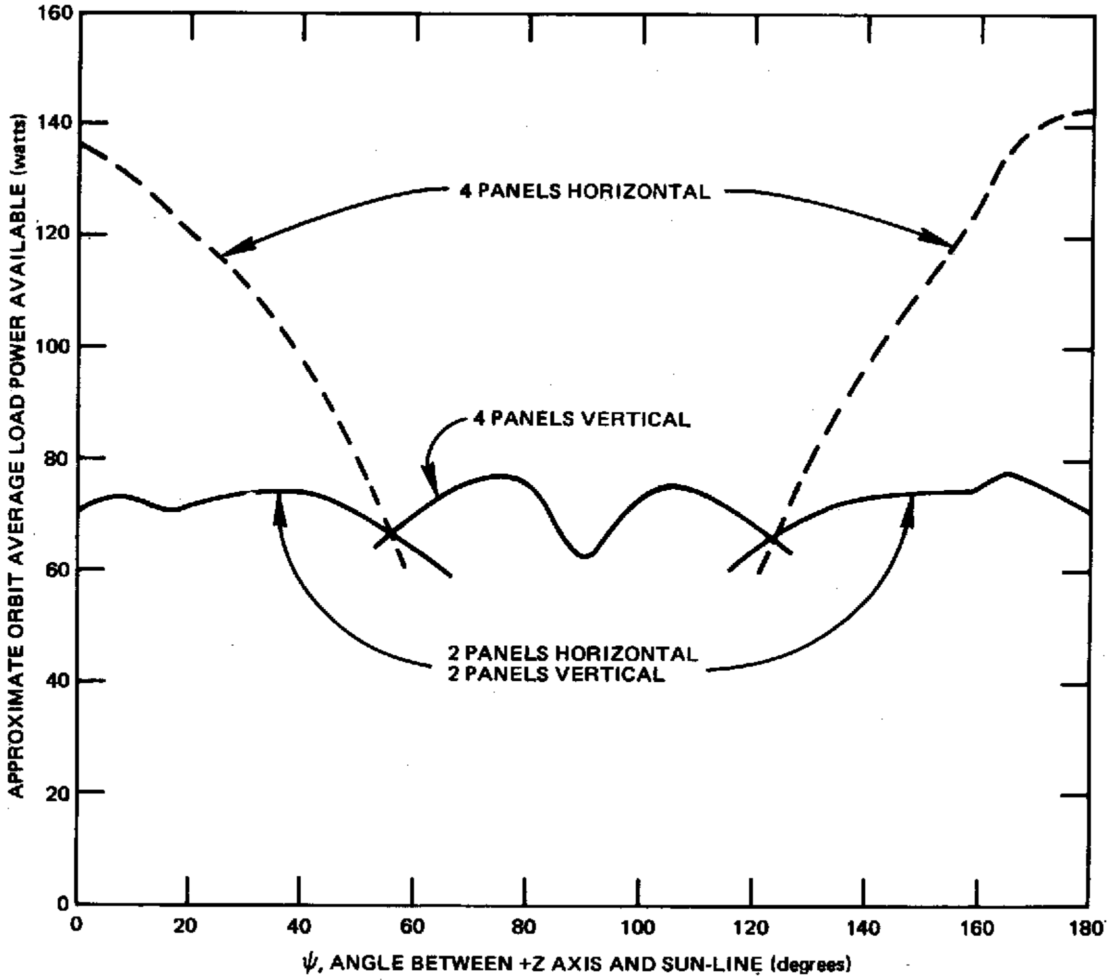

Fig. 6 POWER AVAILABLE, SAS (ASSUMING 10% INCREASE DUE TO ALBEDO)

If the battery discharge current is excessive and the voltage drops to 13 volts, an overload detector will remove the battery and all loads (except the command system receivers and selected heaters) from the main power buss, thus putting the spacecraft into a "solar only" mode. Under normal conditions, CRAM will control the upper limit of the voltage. However, in case of a dual CRAM failure, zener diodes will limit the upper voltage to 22 volts. The overload detector can be overridden, as well as set and reset by ground command.

COMMAND SYSTEM

SAS has a redundant command system (Figure 7) designed to conform to the NASA Goddard Space Flight Center Aerospace Data Systems Standards for Pulse-Code Modulated (PCM) Command Systems. Redundancy begins at the antenna and dc-to-dc converter power inputs and ends with the relay coils, using only a single set of relay contacts. At the point of delivery of a data command to a user, this redundancy also ceases. The system provides 56 relay commands, 10 of which are for the experiment.

The command signal, a pulse-code modulated, frequency-shift-keyed, amplitude-modulated (PCM/FSK-AM/AM) signal, is demodulated by either or both tuned radio frequency (TRF) command receivers, and presented to the bit detector as a non-return-to-zero (NRZ) signal which is 50% amplitude-modulated onto the data subcarrier by a sine wave bit synchronizing signal that is 180° out of phase with the bit pattern (Figure 8). A command word consists of 64 bits per second. The bit detector checks each bit as it is sent by the ground command station. If a bit does not appear at the expected time, the entire command is rejected and must be retransmitted. If accepted, the command is sent to the decoder to determine whether the transmission is a long-load command for the delayed command system, or whether it is a relay or data command to be executed immediately. Depending on this selection, it is routed for storage or execution.

Each of the redundant delayed command systems can store up to 15 commands, either of the data or relay type, providing a total capability of 30 stored commands. The significant 41 bits, a control bit, and the 12-bit time delay associated with each of the 15 commands are stored with a 12-bit "end of sequence" code in the 822-bit complementary metal oxide silicon (CMOS) storage register. The 12-bit counter counts pulses every 2.0867185 seconds until the number of stored timing pulses has been reached. At that time, which may be anywhere between 2 seconds and 6 hours, the stored command is released to the command decoder for appropriate action. The time of execution of the first command can be set to 0.5 millisecond, by accurately timing the transmission from the ground station of the real-time epoch-set command, from which all delays are measured. An epoch-set command can also be sent to one of the two delayed command systems by a pulse generated within the satellite, such as the sun crossing. This provides a timing signal for locating a precise point in space that is independent of absolute time on the ground.

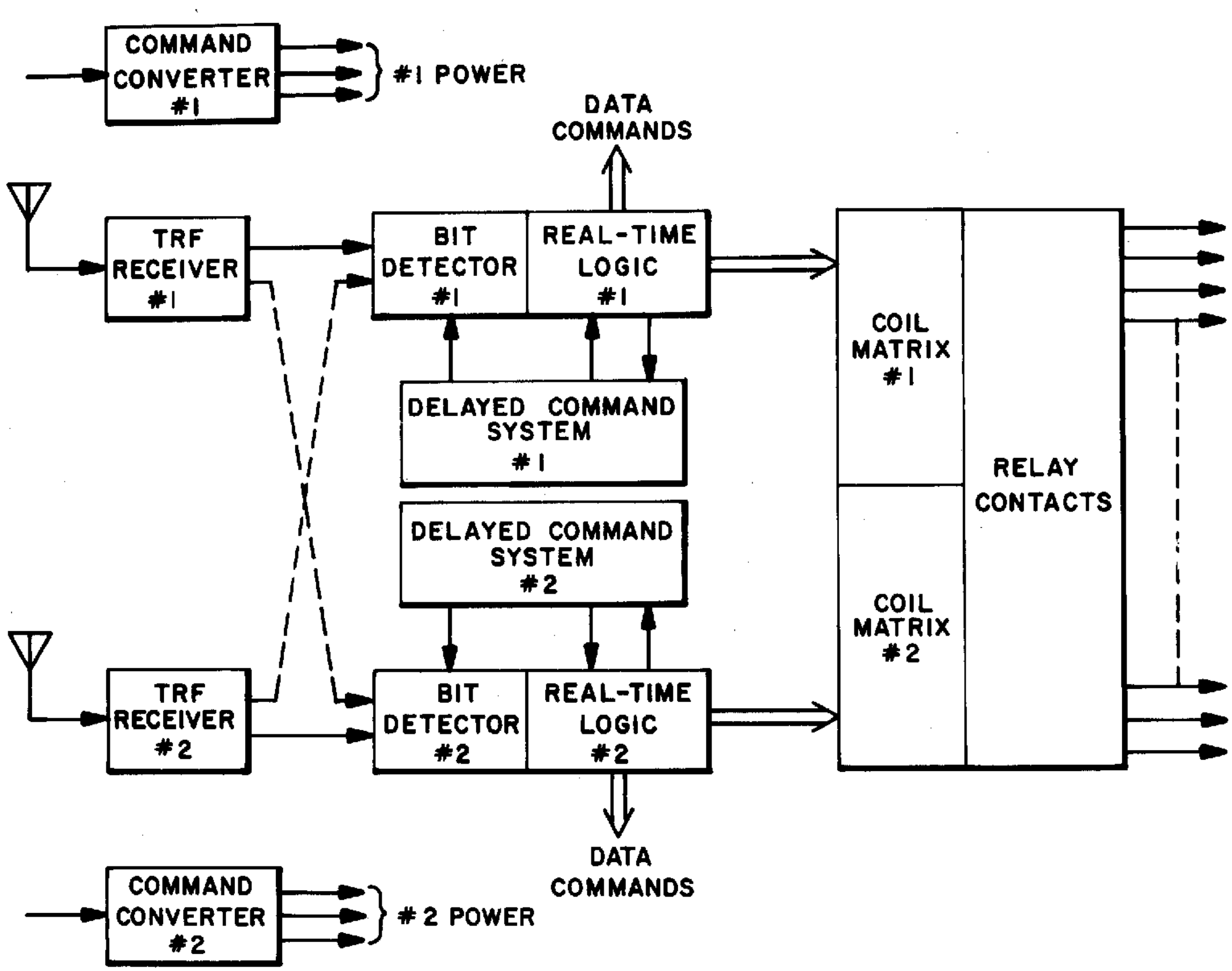

FIGURE 7 SAS COMMAND SYSTEM - BLOCK DIAGRAM

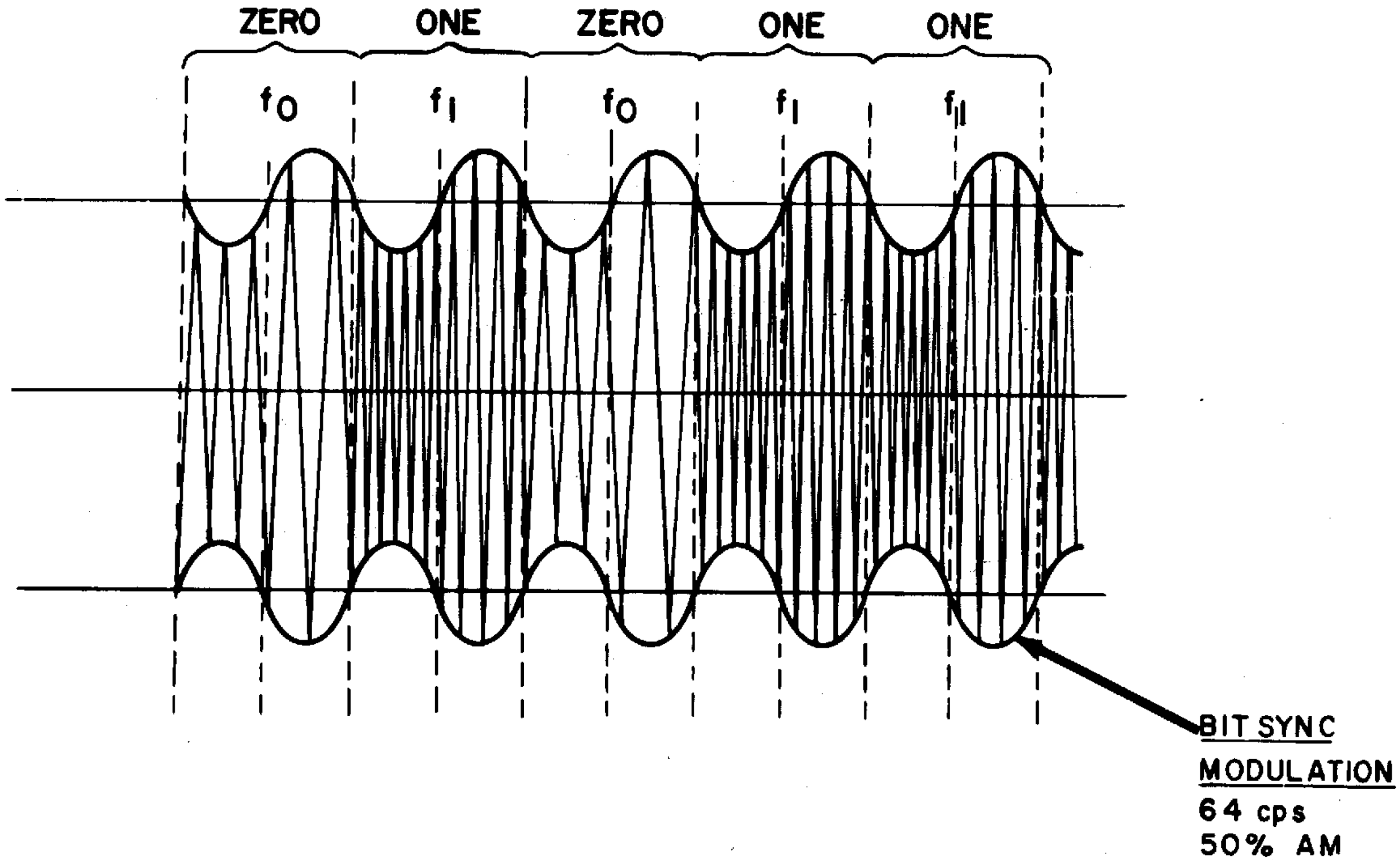

FIGURE 8 SAS COMMAND SIGNAL

Instead of being destroyed as they are used, delayed commands are recycled, permitting the same program to be executed on succeeding orbits merely by sending an epoch-set command either from the ground or from the internal source. Since it may not be desirable to use the full 15 command capability each orbit, zero-fill commands with execution times within the orbital period will allow the commands within the system to recycle to their original positions.

TELEMETRY SYSTEM

As mentioned in the discussion of the structural and thermal design, the SAS-C spacecraft control section has been designed to fly in any orbit. For its prime high energy astronomy missions, its preferred orbit is equatorial in order to avoid the South Atlantic Anomaly where the trapped particles in the radiationbelts come close to the surface of the earth. By choosing an orbit that does not pass through these belts, the lifetime of the detectors is increased, and the background counting rate is decreased. For this reason, SAS-1 and SAS-2 were launched from the San Marco Range, located at 2.9°S. latitude, 41° E. longitude. This range is owned and operated by the Centro Ricerche Aerospaziali of the University of Rome, under the direction of Professor Luigi Broglio. Future high energy astronomy experiments, such as the one planned for SAS-C will also be launched from the San Marco Range.

This presents a special problem for data collection since the only two stations in NASA's Spaceflight Tracking and Data Network (STDN) which can see a spacecraft in a low equatorial orbit are at Quito, Equador, and Ascension Island. The most effective way of getting 100 percent orbital coverage is the use of on-board storage devices. For storage of a complete orbit of data, tape recorders still provide the most compact, lightweight means available. The redundant GSFC-built endless-loop tape recorders used on SAS can store 6.0×10^6 bits each on a single track of tape approximately 90 meters long. The back of the tape is lubricated so it can be pulled off the inside of a cartridge and wound on its outside after passing over the head at 1.5 centimeters per second in record, and 30 centimeters per second in playback. The specification for wow and flutter for this recorder is less than 2 percent peak-to-peak. Requiring about one watt in the record mode and 2 watts in playback, it is designed to store 100 minutes of data whose rate is 1000 bits per second. Recorders with 10^8 bits of storage and various input data rates are being developed and will be available within the next few years. In case of failure of both tape recorders, we re on real-time data acquisition by other stations around the equator, for example, the French station at Kourou, French Guiana.

The telemetry system on SAS-C (Figure 9) is much more sophisticated than its predecessors on SAS-1 and SAS-2, although it has the same two basic modes of operation: recording and transmission of real-time data, and

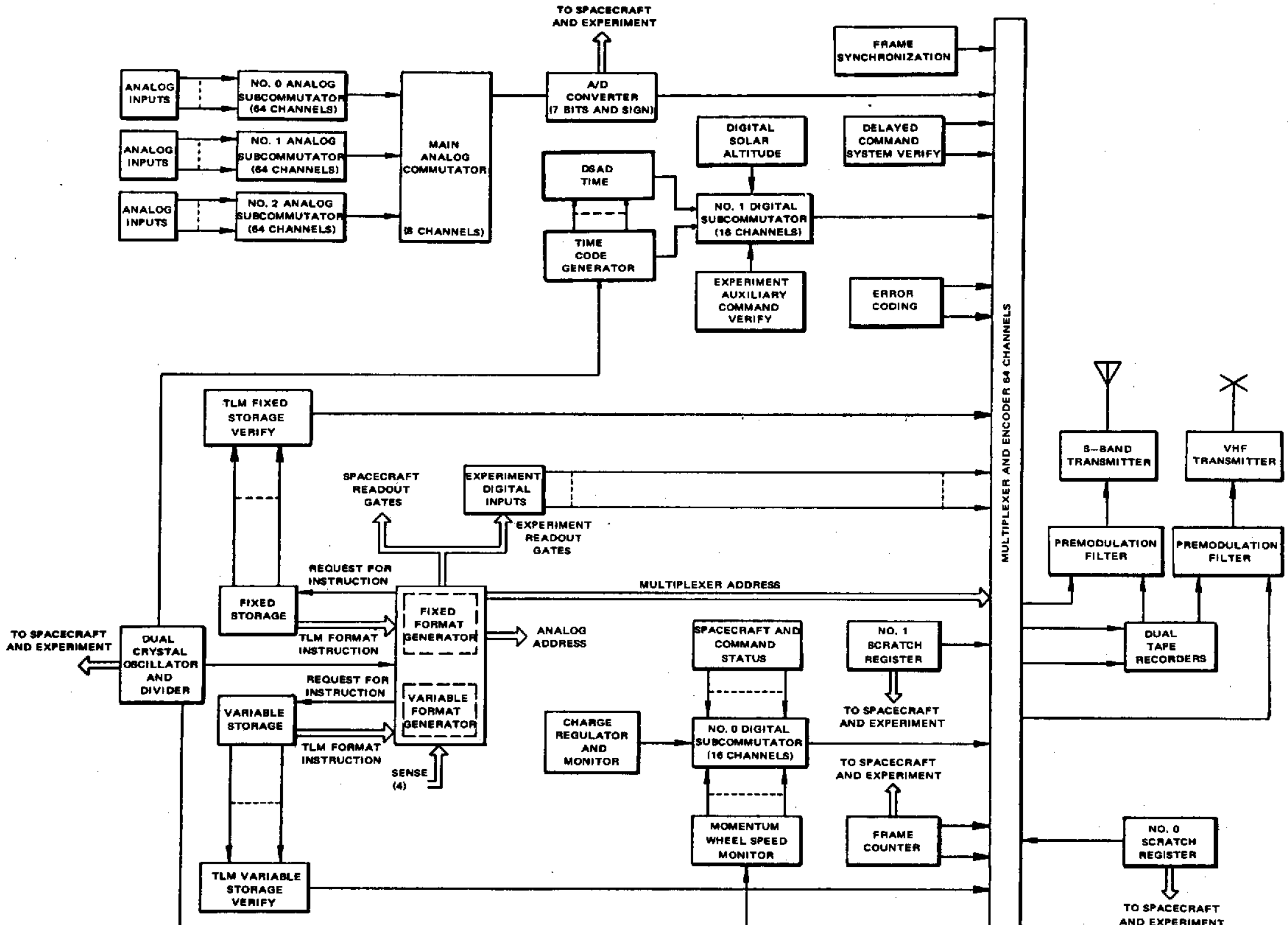

FIGURE 9 SAS TELEMETRY SYSTEM - BLOCK DIAGRAM

playback of stored data. A major difference in the improved SAS is that both of these modes can be operated simultaneously, with real-time transmitted on the VHF transmitter and playback data on the S-band transmitter, or vice versa. Real-time data can also be stored on the second tape recorder while the first is playing back data recorded on a previous orbit, thus preserving all experiment data. The addition of the higher frequency (2250MHz) S-band transmitter should improve the reliability of communications during propagation disturbances which occur regularly at Quito because of its proximity to the magnetic equator.

The most important increase in sophistication, however, is that the telemetry format is reprogrammable while in orbit. There is a choice of two hard-wire formats to which the system can always be switched by ground command. This protects the system against a power loss when the stored program might lose information, provides data while reloading the stored program, and serves as a backup. Normal operation will permit a choice of programsstored in a small core memory. It is this memory that can be reloaded from the ground whenever the satellite passes over a ground station with a suitable command encoder. The operating size of the fixed and variable storage is 256 program steps of 16 bits, or a total of 4096 bits each. With this system, the experimenter can select the desired order and frequency with which he samples the available data sources. These data sources may be either analog or digital, with digital data being prime and analog usually reserved for monitoring housekeeping functions. Although data will be stored on the tape recorders at only 1000 bits per second, stored data can be foregone if it is more desirable to send real-time data at one of the other available bit rates, nominally 125, 250, 500, 2000, 4000, 8000, or 16,000 bits per second.

With all of the flexibility built into the SAS telemetry system, one part remains invariable - the selected frame synchronization word. In addition, for simplifying ground data handling operations, the frame length used on any particular mission will also remain constant, although the format can be changed readily within that limitation. For SAS-C this will be 816 bits in a minor frame and 256 minor frames in a major frame. For ease of ground operations, also, the length of the verification data streams for both the program stored for telemetry and the program stored for the delayed command system are the same as the basic telemetry frame.

Through the use of the delayed command system, an interrupt feature in the telemetry system permits changing the format to another stored in the format memory during the orbit, even when out of sight of one of the ground stations. This can be triggered at a pre-set time or by a specific sensor located on board the satellite. Other interrupts are used to initiate the transmission through the real-time telemetry system of the verification data streams to check the contents of the delayed command system after loading and the contents of the telemetry format stored in memory. The program being used will be read out slowly through the telemetry system; the interrupt merely allows for rapid verification after loading.

The telemetry system was designed using the Texas Instruments 54L family of integrated circuits, plus some additional special components. It consists of redundant crystal oscillators and divider chains, format generators (which operate from fixed or variable storage), scratch registers, a frame counter, a time code generator, two 16-channel digital subcommutators, three 64-channel analog subcommutators (one of which is reserved exclusively for the use of the experimenter), the main analog commutator and analog-to-digital converter, multiplexer and encoders, error control, tape recorders, pre-modulation filters, phase-modulated transmitters, and antennas. It also provides time storage for the digital solar attitude detector and a monitor for momentum wheel speed.

The ultra-stable crystal-controlled oscillator employs a fifth overtone 5.025MHz crystal. Stability of the basic oscillator is 10^{-10} per hour over a temperature range of 0° to +50°C. The signal is used to synthesize the required clock rates for the telemetry system and other reference sources requiring a stable alternating signal, such as the tape recorder drive motors and the synchronous motors for driving the solar panels. With this stable clock as an input, the format generator provides the timing and control functions for all circuits within the telemetry system. These include the multiplexer address, the analog subcommutator address, and timing of the experiment and spacecraft control section readout gates, thus allowing all the data to be multiplexed into the proper sequence.

The scratch registers have three purposes: to allow a format number to be read into the telemetry data stream from storage, flagging the format being used for easy interpretation on the ground; to control the operating mode of some circuits; and to be able to read into the data stream a frame synchronization code other than the one normally provided. This provides a redundant method of inserting the proper frame synchronization code as well as substituting a different one. The frame counter is an 8-bit counter controlled by either of the scratch registers and defines the length of one major frame of data. The eight parallel output lines are available for spacecraft and experiment use.

The time code generator is used to relate spacecraft time to ground time. It is a 24 stage binary counter whose contents are stored at the beginning of every major frame. With a resolution of 2 seconds per bit, it has the capability of a non-repeating readout for approximately a year. The 24-bit stored output is transmitted via one of the digital subcommutators every sixteen minor frames, thus repeating 16 times within the major frame before being updated. The digital subcommutators multiplex not only the output of the time code generator, but also status data from various other sources that are generated in digital form, e.g., the status of the charge regulator and monitor (CRAM) for the power system, momentum wheel speed, command verification, digital solar attitude detector data, and the nutation damper angle.

The telemetry system includes three analog subcommutators, a main analog commutator, and an analog-to-digital converter. Used primarily for monitoring housekeeping information such as voltages, currents, and temperatures, it can accept input voltages between plus and minus 10 volts by attenuating them to ±0.254 volts before they reach the junction field effect transistor switches used in the main analog commutator, whose purpose is to multiplex the outputs of the analog subcommutators and provide the input to the analog-to-digital converter. The main analog commutator, like the analog and digital subcommutators, is controlled by the format generator. The analog-to-digital converter uses the dual slope technique to convert the analog inputs to an 8-bit binary output. It is clocked at 16kHz and has an aperture time of 2 milliseconds, which provides an accuracy of ±0.2% for a signal below 10Hz over a temperature range of -25° to +55°C.

The prime inputs to the telemetry system are digital. They can come from 64 separate sources in any length from 4 to 32 bits, in increments of 4 bits. Using appropriate signals from the format generator, the multiplexer accepts all the various digital telemetry data and combines them serially, adding a parity check bit for error coding. These data are then routed to the encoder where they are converted from non-return-to-zero (NRZ-C) to split phase to provide a waveform free from switching transients. Outputs from the encoder go to one of the on-board tape recorders, or to either or both of the transmitters through pre-modulation filters whose purpose is to keep the transmitted bandwidth within its allocation as prescribed by the NASA Goddard Space Flight Center Aerospace Data Systems Standards for PCM Telemetry Systems. This allocation is ±15kHz for real-time data or ±45kHz for playback data through the VHF transmitter or ±1MHz at S-band.

The 136MHz VHF transmitter transmits phase-modulated data at one of two power levels, 0.25 or 1.5 watts. The higher power mode is normally used only for playback data, although real-time data can also be transmitted at that level. The low power mode is normally on continuously, transmitting real-time 1000 bps data, and is used as a tracking beacon for orbit determination through the NASA Minitrack system. The 2250MHz S-band transmitter also has two power levels, 1 watt and 4 watts, primarily intended for real-time and playback data. Both transmitters are modulated by split-phase encoded data and set for a deviation of 65° peak. The S-band antenna pattern is nearly omnidirectional and generated by identical helices located on the ends of opposing solar panels. The VHF antenna pattern is also omnidirectional and generated by a turnstile located at the end of one of the remaining solar panels. One of the elements of this turnstile also serves as a dipole for the reception of the 148MHz command receiver signals. A second dipole, on the end of the fourth solar panel which opposes the turnstile, is positioned at 90° to the turnstile element used for the command system. This second dipole routes its received signal to the redundant command receiver.

CONTROL SYSTEM

As with its predecessors, the most unique feature of the Small Astronomy Satellite is its control system. An evolutionary system is planned which will permit the present system to progress to an accurately pointed, three-axis stabilized spacecraft - an especially valuable tool in astronomy.

The basic control system uses the earth's magnetic field for torquing the spacecraft, spinning and despinning it, and for unloading the reaction wheel. An obvious advantage of this is the elimination of the need for expendables and their inherent life limitations. Active nutation damping controls coning about the Z-axis.

Experiments can look out along the Z-axis or perpendicular to it. In the latter case, if the spacecraft is rotated slowly about its Z-axis, then a swath is swept out of the celestial sphere. If the speed of that rotation is exactly once per orbit, then a skyward-viewing experiment never sees the earth and, conversely, an experiment can be made to view the earth continuously. Thus the same control system can provide the capability of viewing anywhere in the celestial sphere or of being earth-locked. The stability required for the spacecraft is accomplished through the use of a variable speed reaction wheel whose speed is controlled by an error signal generated by an external source. If a rate gyro is used, then the rotation about the Z-axis can be made quite linear; if an earth sensor is used, then the spacecraft can be locked onto the earth to one degree as it makes its one rotation per orbit. A fixed speed mode can be provided by using the wheel tachometer output as the error signal. A fourth mode of operation is to use the error signal from a star tracking camera to hold the rotation about the Z-axis to zero and keep a side-viewing experiment fixed in inertial space. This, with the addition of two more small reaction wheels, two additional star tracking cameras, and the associated electronics, would provide us with a three-axis stabilized spacecraft; thus, the evolutionary system mentioned above.

Figure 10 shows the various components of the control system. Z-axis orientation is accomplished by energizing a torquing coil. This electromagnet, acting like a compass needle, attempts to align the spacecraft with the earth's magnetic field. At the maximum rate, using a magnetic dipole of $\pm 5 \times 10^4$ pole-centimeters, the Z-axis of the spacecraft can be moved about one degree per minute. Continuous motion, or compensation for unwanted drifts due to residual magnetic dipoles in the spacecraft, can be accomplished by a ± 1000 pole-centimeter dipole having 10 pole-centimeter steps.

Spinning and despinning is accomplished by amplifying the outputs of the X- and Y-axis coils respectively. This system, in conjunction with the earth's magnetic field, operates essentially as a motor to increase

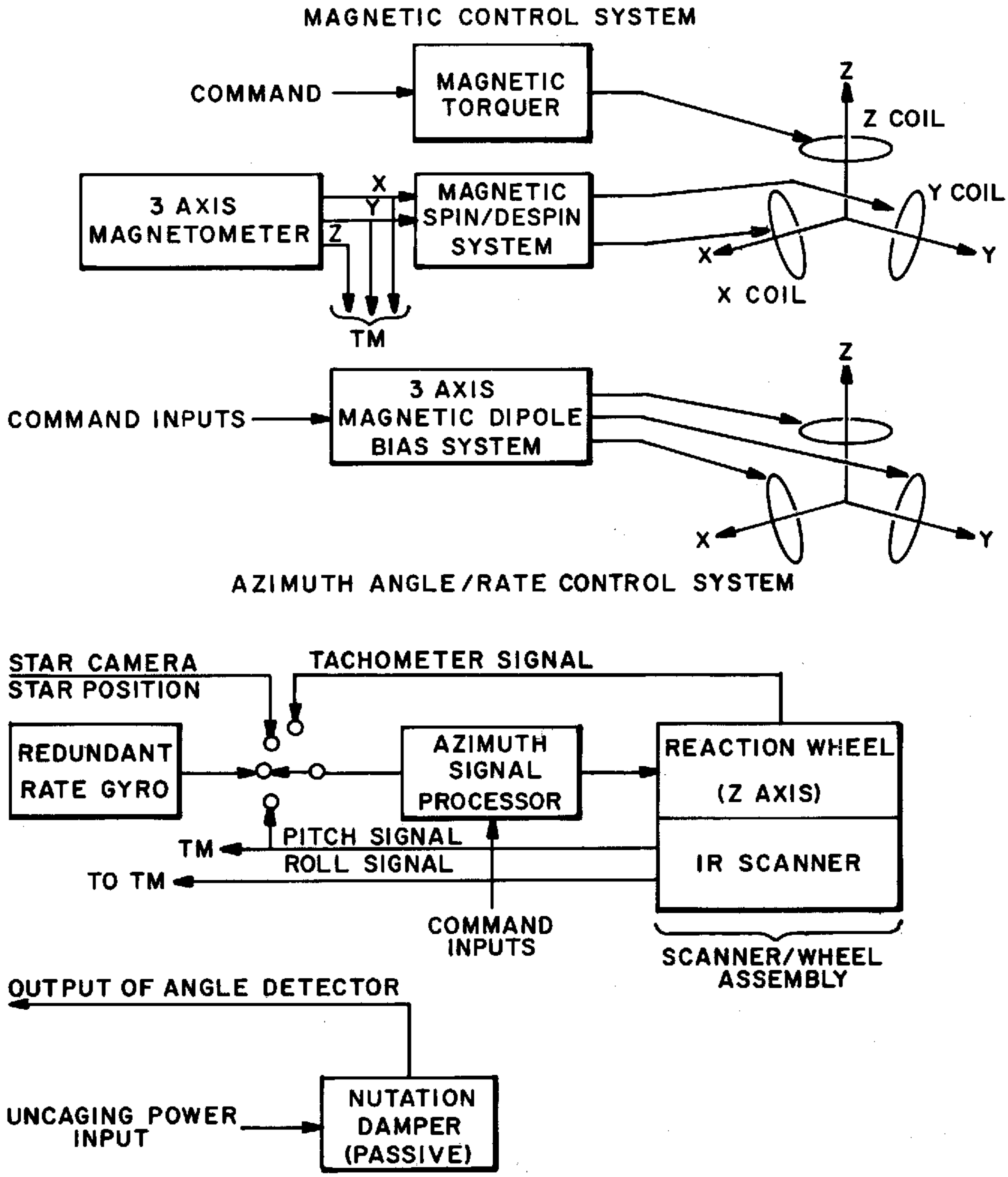

FIGURE 10
SAS CONTROL SYSTEM, BLOCK DIAGRAM

or decrease the rate of rotation about the Z-axis. With maximum dipoles of 1×10^4 pole-centimeters, the maximum rate of change of spin is about 10 rpm per day. This system also has bias dipoles for compensation of residual magnetism in the spacecraft, ±500 pole-centimeters in the X- and Y-axes in 10 pole-centimeter increments.

With the delayed command system, spinup or spindown and torquing activities can be performed out of sight of the ground stations. This provides us with the advantage of being able to plan operations for selected time periods during which there is an optimum relationship between the position of the satellite in its orbit and the earth's magnetic belt values for which are stored in a computer on the ground.

The spin-despin system is also used to inject energy when the momentum wheel has reached saturation, allowing it to operate nearer its nominal speed of 1500 rpm. This reaction wheel is driven by a d.c. induction motor and requires about 4 watts. At 1500 rpm, it produces a momentum of 30 slug-ft^2-rpm. It serves as the primary source of stability for this dual-spin spacecraft whose outer body is spinning at only one revolution per orbit or not at all. Its variable speed is a significant improvement over SAS-1 and SAS-2 whose rotation about the Z-axis was slightly non-linear because the fixed speed momentum wheel was unable to compensate at all for external torques such as those created by the gravity gradient effects of the spacecraft and its solar paddles. The speed of the reaction wheel will be controlled under normal operating conditions by a temperature-controlled air-bearing, rate-integrating gyro. Its speed is 23,264 rpm with a stability of less than one part in a million. In the spacecraft "dither" mode, the spacecraft will be required to oscillate back and forth across a source. This function will be controlled by setting the appropriate biases into the rate gyro system at times determined by the delayed command system.

A nutation damper will be used on SAS to dissipate the lateral components of satellite angular motion (coning) created by the magnetic torquing. It consists of a torsion-wire suspended arm with an end mass and a copper damping vane. If nutations occur, the arm oscillates, causing the copper vane to swing back and forth through the gap in a permanent magnet, thus inducing eddy currents in the copper vane. The plane of the pendulum is normal to the wheel momentum axis and displaced as far as possible from the center of gravity of the spacecraft. With the wheel on, the residual motion of the Z-axis should be less than 0.2°. The motion of the nutation damper pendulum is detected and telemetered by a seven bit optoelectronic system.

ATTITUDE DETERMINATION SYSTEM

The attitude of the spacecraft can be determined to a few degrees by magnetometer and sun sensor data. A finer determination (about 30 arc-minutes) can be made from the star sensor located in the spacecraft control section. Star tracking cameras in the SAS-C experiment will provide attitude determination to about 15 arc-seconds.

A three-axis vector magnetometer will be located in one quadrant of the body of the spacecraft control section. This differs from the arrangement on SAS-1 and SAS-2, where the X- and Y-axis magnetometers were located on two of the solar paddles with the Z-axis magnetometer on the body of the spacecraft. Having them in one place gives advantages and disadvantages. The orthogonality of the three is known and maintained with more precision, but there is more interaction among them.

The digital solar attitude detector consists of spinning and nonspinning sensors. The non-spinning portion has two 1/2° resolution sensors with cone angles of 128°. They are positioned on the spacecraft with the centers of their fields of view in the same place and with their cone angles overlapping. Data from these sensors consist of two eight-bit digital words plus two telltale signals indicating which, if either, sensor is illuminated. The spinning system has a single sensor with a field of view of 180° x 2° and a resolution of sun position to 1°, provided in an eight-bit digital word. The time of sun crossing is stored and telemetered with an accuracy of ±1 second. The total system requires 0.5 watt.

The star sensor planned for SAS-C and later spacecraft is the same as that used on SAS-2 and similar in design to the one used in the exeperiment on SAS-1. Designed to be sensitive to stars of +4 magnitude and brighter, it is mounted perpendicular to the spin axis of the spacecraft and sweeps out a 10° band in the celestial sphere as the satellite rotates about its Z-axis. Starlight will be focused by the sensor optics onto a reticle having an N-shaped slit behind which is a photomultiplier tube. A star passing across the N-shaped slit will produce three current pulses in the photomultiplier tube. The end pulses provide information on azimuth and the center one on elevation. The analog output from the photomultiplier tube is converted to digital form by the telemetry system's analog-to-digital converter and stored with data bits denoting the time at which a star began its transit of the slit. Ground data processing will take advantage of the sun sensor and magnetometer data for a coarse determination of attitude, which simplifies the computer program required to convert the star sensor data into fine attitude information. A sunshade permits operation in the daylight portion of the orbit, and a bright object sensor turns off the high voltage to the photomultiplier tube whenever the star sensor is within about 45° of the sun.

SUMMARY

The Small Astronomy Satellite (SAS) spacecraft control section offers the utmost in flexibility possible in a spacecraft small enough to be launched by the smallest of NASA's rockets, the Scout. This versatility makes it possible to manufacture several at a time, making them available on a quick-reaction, minimum cost basis for a variety of experiment payloads.

APPENDIX A

SUMMARY OF CHARACTERISTICS

OF

SMALL ASTRONOMY SATELLITE (SAS)

GODDARD SPACE FLIGHT CENTER

GREENBELT, MARYLAND

SUMMARY OF CHARACTERISTICS OF SMALL ASTRONOMY SATELLITE (SAS)

Launch Vehicle: Scout F

Size

Spacecraft Control Section: Cylinder 66 cm. in diameter by 61 cm. high without solar panels
4.3 meters tip-to-tip with solar panels

Experiment Payload: Cylindrical volume 76 cm. in diameter by 58 cm. high plus 39 cm. additional height with decreasing diameter. More length is available by using existing volume in spacecraft control section.
See Figure 1A.

Weight

Spacecraft Control Section: 120 kg.

Experiment Payload:

Launch Site	Altitude	Inclin-ation	Exp. Pay-Load Wgt.
San Marco	550km circular	3°	75kg
Wallops Island	550km circular	38°	63kg
Western Test Range	550km circular	90°	29kg

Power

65 watts minimum orbit average
Spacecraft Control Section: 40 watts
Available for Experiment: 25 watts
Voltage: +16.1 ± 15% volts dc
Ground: Three-wire system; separate chassis, signal and power grounds

Telemetry

Type: Reprogrammable PCM split-phase with variable bit rates
Real time bit rates: Binary submultiples of 1.005MHz
Nominal: 981 bps
Other: 123, 245, 491, 1962, 3925, 7851, and 15,703 bps
Storage method: Endless loop tape recorder
Capacity: 6×10^6 bits
Record rate: 981 pbs
Playback rate: 19629 bps (20:1) for approx. 5 min.
Transmitters: 136 MHz (90 kHz bandwidth), with two power modes (0.25W and 1.5W.)
2250 MHz (1 MHz bandwidth) with two power modes (1W. and 4W.)

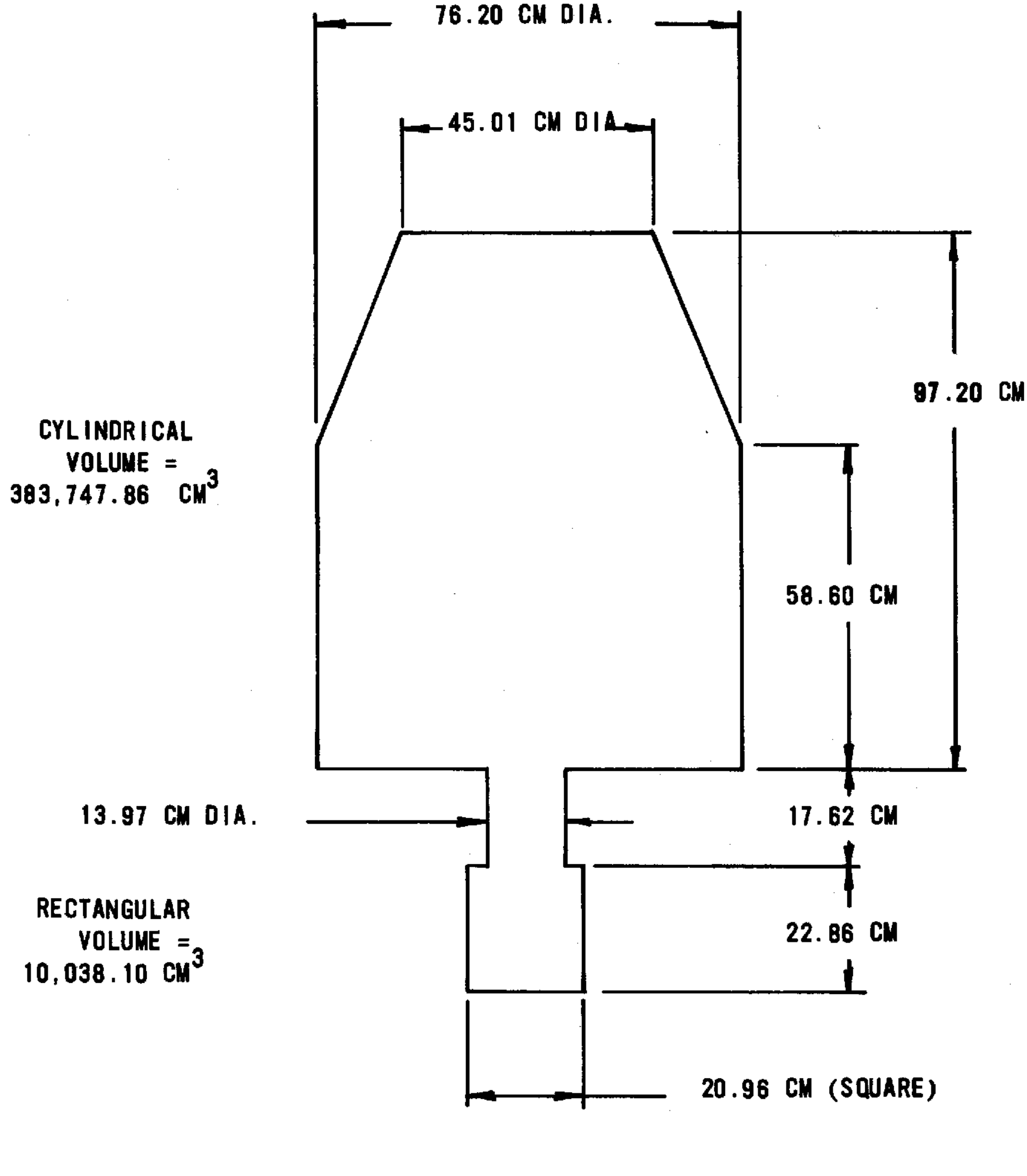

FIG. 1A

OUTLINE DRAWING OF SAS EXPERIMENT ENVELOPE

Encoder: 8 bit analog-to-digital converter will provide better then ±0.2% accuracy for signals up to 10 Hz

Input Signal Levels: Analog: ±10.0v., ±7.5v., ±5.0v., ±2.5v., ±1.25v., ±0.5v., ±0.25v.

Digital: +4v. for a "1" and +0.3v. for a "0" into a standard Texas Instruments Series 54L TTL input.

Program: One hard wired general program (can be changed between missions by changing one card)
One reprogrammable, non-volatile memory

Tracking
Positional accuracy from Minitrack System: 10 km.

Command System
Conforms to NASA/GSFC Aerospace Data Systems Standards for PCM Command Systems
Spacecraft control section: 46 "on" and "off" relay commands
Experiment: 10 "on" and "off" relay commands
24 bit data commands are routed to control system, experiment and second user. Second user will reroute 24 bit commands using the first four bits for routing and 20 bits for commands. Data commands must be decoded by the user.
Stored commands: Any 15 relay or data commands can be stored for later activation in each of two redundant command memories for a total of 30 delayed commands. One of these command systems can be triggered by an on-board pulse to start its timing sequence. The other must be sent its epoch pulse by ground command.

Control System
Positioning of Z-axis: <1° relative to known position
Maneuver Rate: ~1° per minute
Average drift of Z-axis: ⩽2° per day (assumes residual magnetic moment of experiment to be <1000 pole-cm.)
Spin rate: 0 to 10 rpm
Nutation angle (coning): ⩽0.1°
Nutation period: ≫1 minute
Wheel momentum: 30 slug-ft.^{2}-rpm at 1500 rpm
Wheel speed control: Error signals from
(1) Internal technometer for fixed speed
(2) Earth sensor for 1° earth lock
(3) Rate gyro for controlling linearity of rotation to ±0.3° per hour
(4) Star tracking camera for star-lock mode
(1 arc-minute design goal)

Attitude Determination System
Three-axis magnetometers and sun sensors: ±3°
Star sensor: ±1 arc-minute

APPENDIX B

GENERAL DESIGN CONSIDERATIONS FOR SMALL ASTRONOMY SATELLITE (SAS) EXPERIMENTS

GODDARD SPACE FLIGHT CENTER

GREENBELT, MARYLAND

GENERAL DESIGN CONSIDERATIONS FOR SMALL ASTRONOMY SATELLITE (SAS) EXPERIMENTS

1. SCOPE

This General Design Considerations Document describes special requirements imposed by the Small Astronomy Satellite (SAS) Project on flight hardware. It includes parts selection and screening design considerations, and environmental test requirements. In addition, the SAS Project has available a Xerox Data Systems Sigma 5 computer for use during the integrated testing of the experiment and the spacecraft control section. This equipment is also used at the launch site for pre-launch testing and can be used for immediate post-launch evaluation of experiment and spacecraft systems. It is assumed that the experimenter will build a structural and thermal model, a proto-flight model and a flight unit of the experiment.

2. APPLICABLE DOCUMENTS

The current issues of the following documents are to be used as references and form a part of the SAS Project requirements for spaceflight hardware, where applicable. The first document, the GSFC Preferred Parts List (PPL), includes specific requirements for parts screening that are directly applicable.

- GSFC PPL XX, current edition
- GSFC S-320-1, "General Environmental Test Specifications for Spacecraft and Components," current edition
- GSFC S-250-P-18, "Contractor-Prepared Monthly, Periodic, and Final Reports with Amendment #2," November 14, 1970
- GSFC S-311-P-12A, "Inspection Criteria for Scanning Electron-Microscope Inspection of Semiconductor Device Metallization"
- GSFC S-713-P-5A, "Printed Wiring Board Design, Fabrication, Inspection and Handling"
- GSFC X-713-72-296, "Design of Welded Cord Wood Modules"
- GSFC X-713-72-369, "Design of Welded P-C Stick Modules"
- GSFC X-735-70-73, "Outgassing Studies of Some Polymer Systems for GSFC Cognizant Spacecraft," February 1970
- GSFC X-764-71-314, "A Compilation of Low Outgassing Polymer Materials Normally Recommended for GSFC Cognizant Spacecraft," July 1971

- GSFC-W3, "Welded Electronic Modules"
- GSFC S-723-P5A, "Quality Assurance Requirements for Standard Industrial Equipment," October 1966
- NASA NHB 5300.4(3A), "Requirements for Soldered Electrical Connections," May 1968, as applicable
- MIL-STD-480, "Configuration-Control-Engineering Changes, Deviations and Waivers," October 30, 1968
- MIL-HDBK-217A, "Reliability Stress and Failure Rate Data for Electronic Equipment," December 1, 1965
- MIL-M-38510, "General Specification for Microcircuits"
- MIL-STD-750, "Test Methods for Semiconductor Devices"
- MIL-STD-461A, "Requirement for Electromagnetic Interference Characteristics"
- MIL-STD-462, "Measurement of Electromagnetic Interference Characteristics"
- MIL-STD-463, "Electromagnetic Interference Technology, Definitions and System of Units"
- MIL-STD-1290, "Marking for Shipment and Storage"
- MIL-STD-794B, "Procedures for Packaging and Packing of Parts and Equipment"
- MIL-STD-SS9, "Dissimilar Metals"
- MIL-STD-826, "Electromagnetic Interference Test Requirements and Test Method"
- MIL-STD-810B, "Environmental Test Methods," June 1967
- MIL-STD-130, "Identification and Marking of Components"
- MIL-STD-105D, "Sampling Procedures and Tables for Inspection by Attributes"
- MSFC-STD-136, "Parts Mounting Design Requirements for Printed Wiring Board Assemblies"

- FED-STD-209A, "Clean Room and Work Station Requirements"
- GSFC X-673-64-1C, "GSFC Engineering Standards Design Manual," January 1972
- GSFC X-325-67-70, "Magnetic Field Restraints for Spacecraft Systems and Subsystems"
- GSFC X-560-63-2, "Aerospace Data Systems Standards"

3. PARTS AND MATERIALS

All parts, wire, cable, connectors and material used in the SAS spacecraft shall be approved by the GSFC Project Manager. Preference shall be given to those in GSFC Preferred Parts List; items not listed therein must be submitted for approval with sufficient information to prove their acceptability for use in flight hardware.

In general, parts screening will be as described in the current GSFC Preferred Parts List. Only high reliability parts will be used in the SAS spacecraft control section and experiments. Specifically, for all semi-conductors, i.e., diodes, transistors, and integrated circuits, the following screening sequence is required as a minimum:

- Visual inspection before sealing or X-ray examination after sealing, if the latter can be shown to be effective for the specific type of semi-conductor
- Temperature cycling from -65°C. to maximum rated storage temperature
- Centrifuge test
- Electrical test with variables data recorded
- 336 ±36 hours burn-in at 100°C. and 80% of part-rated power
- Electrical test with variables data recorded (parts will be rejected if outside of acceptable variables limits)
- Fine and gross leak tests
- Final inspection

In submitting the total parts list for approval, the following information must be included as a minimum:

- Type of component
- Value and rating
- Manufacturer
- Manufacturer's type and model
- Manufacturer's screening process specifications to which part is being bought
- Maximum anticipated electrical stress level

In submitting the total materials list for approval, the following information must be included as a minimum:

- Purpose for which material is to be used
- Location in spacecraft or experiment
- Quantity to be used
- Outgassing characteristics, if known

No cadmium plating shall be used anywhere in the SAS spacecraft.

4. DESIGN CONSIDERATIONS

A specific design requirements document for each individual experiment will be written. However, certain general design considerations can be defined herein.

4.1 Power System

In the design of the power system and its experiment loads, consideration must be given to protection features for the spacecraft control section, for the experiment section and for the power system. The experiment must not feed back into the power system surges greater than 2 amperes for more than 0.1 second. It must tolerate voltages up to 25 volts indefinitely, although the anticipated voltage provided by the SAS spacecraft control section is 16.1 ±15% volts. It must tolerate ripple up to 250 mv at any frequency up to 20kHz.

High voltage components are to be potted and must be able to survive, operating, in the corona region for an indefinite period. It is not a requirement that they meet their operational specifications when operating in the corona region, but they must survive and be able to meet their operational specifications after they are at the pressure levels expected in flight.

A three-wire grounding system must be provided. This means that chassis ground, signal ground, and power ground are to be kept separate and made available at the interface connector between the spacecraft control section and the experiment section.

4.2 Noise Protection

The experiment must protect itself against radiated pulses, e.g., from the spacecraft clock, and must also assure that it is not radiating pulses that might interfere with operation of the spacecraft control section, e.g., from its clocking circuits or dc-to-dc converter transients. Tests will be run to determine whether there is any RFI. The experimenter is responsible for eliminating such unnecessary transients and noise from his lines by means of suitable filters, shielding, etc.

If an open on a telemetry output would develop a large voltage surge to the multiplexer input (more than a factor of 2 over nominal), the output must be paralleled with a protective circuit (Zener diode). Isolation shall be provided from the subsystem monitoring circuits in case of telemetry shorting.

4.3 Command System

Ten relay commands are provided to the experiment by the spacecraft control section. In addition, 24-bit data commands are routed to the experiment. To be used, these must be decoded by the experiment. This provides an additional capability of 2^{24} commands.

4.4 Telemetry System

Out of the 816-bit minor frame, about 720 bits are available for prime experiment data. In addition, one 64-channel analog subcommutator is allocated to the experiment for housekeeping and other data. One sample of this 64-channel subcommutator appears in each minor frame, so each position is sampled at a rate of about once per minute, unless the telemetry system is programmed to sample a specific position more frequently.

Calibration curves shall be defined for each telemetered function. They are to be derived from sufficient data to allow smooth curves to be plotted.

Protection for open or short circuits in the telemetry system was discussed under Paragraph 4.2 on Noise Protection.

4.5 Control System

The current design provides a control system that will point anywhere in the sky from a low earth orbit. The Z-axis can drift up to 2° per day. Attitude determination can be provided by the star sensor on the spacecraft control section, after ground data analysis, to one arc-minute. Stable three-axis pointing to one arc-minute requires a development program for the spacecraft. More accurate attitude determination with the present system requires more accurate sensors on the experiment. A development program would also be required for synchronous orbit missions, although many of the required characteristics have been designed into the present spacecraft control section.

4.6 Thermal Design

Current plans are for independent thermal designs for the spacecraft control section and the experiment. If a preliminary experiment thermal design indicates great difficulty in achieving the required thermal range, then consideration can be given to an integrated thermal design.

Coatings to be used on the experiment must be approved by the SAS Project Manager. Areas of prime consideration are susceptibility to damage from ultraviolet radiation and outgassing properties.

5. ENVIRONMENTAL TEST REQUIREMENTS

5.1 Vibration Testing

In order to assure survival of the launch environment, vibration testing to the requirements of GSFC S-320-S-1, "General Environmental Test Specifications for Spacecraft and Components", must be performed on the experiment. While exact levels are dependent on the amplifications inherent in the experiment structural design, Table B1 provides a good estimate of qualification vibration levels which might be expected by an experiment bolted to the SAS spacecraft control section.

TABLE B-1 VIBRATION QUALIFICATION LEVELS

SINUSOIDAL VIBRATION

Axis	Frequency (Hz)	Level
Thrust	10-70	0.11 in. d.a.
	70-130	±28.0 g
	130-170	±20.0 g
	170-200	±10.0 g
	200-2000	± 5.0 g
Lateral	5-20	0.6 in. d.a.
	20-26	±12.0 g
	27-200	± 4.0 g
	200-2000	± 5.0 g

Sweep rate: 2 octaves per minute

RANDOM VIBRATION

Axis	Frequency (H_z)	APSD (g^2/H_z)
Thrust	20-150	+6db/oct. up to 0.045
	150-500	+6db/oct. 0.045 to 0.12
	500-2000	-3db/oct. from 0.12
Lateral	20-300	+3db/oct.
	300-2000	up to 0.045

Duration: 2 minutes per axis

5.2 Vacuum Thermal Testing

Each part of the spacecraft shall be designed to operate, unsealed wherever possible, under near-Earth space conditions at pressures down to 10^{-10} mm of Hg. While it is not always possible to test the entire experiment down to these pressures, the design must consider the actual environment. Each part of the spacecraft shall be tested, operating, through partial pressures such as those found in the ionisphere, where corona can occur, to assure its survival in case of accidental turn-on prior to its reaching operational pressure levels. This vacuum test will be conducted for a period of at least two weeks, during which the temperature will be varied over the anticipated operating range plus and minus 10°C. beyond the worst case limits on each end. Details can be found in GSFC S-320-S-1, "General Environmental Test Specifications for Spacecraft and Components"

5.3 Humidity

Each part of the spacecraft must be designed to survive operating conditions of relative humidity of up to 95%. However, experiments having sensors that are particularly sensitive to humidity can request a waiver on those items. If it can be shown that proper precautions can be taken to protect those items in a high humidity environment, the waiver will be granted.

5.4 Magnetic Testing

All magnetic moments, either permanent or variable, should be less than 1000 pole-centimeters as a design goal. Values higher than this will compromise the operation of the control system. Any nickel component leads should be aligned perpendicular to the spin axis. Magnetic materials should be avoided, e.g., only non-magnetic fasteners should be used. A test will be performed at the GSFC Magnetic Test Site to assure that these requirements are met.

5.5 Shipping and Handling

The design must consider the shipping and handling requirements to transport the experiment to the spacecraft control section contractor's plant and to the launch site, including the San Marco Launch Site, Kenya.

6. GROUND SUPPORT EQUIPMENT

The ground support equipment is intended to provide a system for the complete checkout of the integrated spacecraft during final checkout in the integration contractor's plant and at the launch site. GSFC will provide a dual van arrangement consisting of telemetry receivers and a PCM front end working into an XDS Sigma 5 computer. Digital-to-analog converters are available to drive analog pen recorders for direct data readout, in addition to a printer driven directly by the computer. A command encoder and transmitter compatible with the GSFC Aerospace Data Systems Standards are available for transmitting commands to the spacecraft by rf link. Any unique equipment required to determine the suitability of the experiment for launch must be provided by the experimenter. It can be housed in available rack space inside the van. Portable equipment can be located at the checkout room near the spacecraft. Two sets of such equipment must be provided, one for use at the contractor's plant and one for use at the launch site.

6.1 Support During Integration, Testing and Launch

The Integration Contractor is responsible for integration of the experiment with the spacecraft control section, and for testing the integrated spacecraft. The experimenter will provide personnel at the Integration Contractor's plant and/or GSFC during all phases of testing involving the experiment.

6.2 Data Reduction and Analysis

The experimenter must prepare programs to handle post-launch experiment data. These programs must be prepared and checked prior to launch. The experimenter will be provided with a computer formatted tape containing experiment and housekeeping data taken from the integrated spacecraft during testing. The experimenter must prepare a Data Reduction and Analysis Plan, defining how the data will be reduced and analyzed after launch, that will serve as the basis for a post-launch data analysis contract.

7. RELIABILITY AND QUALITY ASSURANCE

The quality and reliability of the SAS spacecraft and experiments will be assured by appropriate testing, careful selection and screening of approved parts, and GSFC reviews consisting of Design Review, Pre-environmental Readiness Review, and Flight Readiness Review. Complete documentation of all schematics, block diagrams, structural design, thermal design, interfaces with the spacecraft control section, a reliability assessment, and experiment-unique ground support equipment must be provided for the appropriate reviews. Configuration management will be imposed after Design Review.

7.1 Test Plans and Procedures

Detailed test plans and procedures will be required. The test plans shall contain a listing of each test to be performed together with the test procedure to be used. The procedure shall include the test objective, step-by-step procedures, personnel responsibilities, applicable safety requirements, plus the test equipment to be used, its calibration requirements, required facilities, and data processing techniques to be used.

7.2 Safety Plan

The experimenter is required to provide a Safety Plan setting out the procedures he plans to use to protect the experiment and the personnel working on the experiment from harm.

7.3 Reporting and Documentation

Malfunctions will be reported using standard GSFC Malfunction Reporting Forms. The experimenter will establish a Materials Review Board for the disposition of nonconformances. Reports on the actions of this board will be provided. A Reliability Program Plan must be submitted by the experimenter.

8. DOCUMENTATION

The following reports and documentation are required.

- Parts List
- Materials List
- Reliability Program Plan
- Safety Plan
- Schematics and Block Diagrams
- Reliability Assessment
- Qualification Test Plan
- Ground Support Equipment Instruction Manual

- Malfunction Reports
- Engineering Change Orders
- Structural and Thermal Model Test Data
- Protoflight Model Test Data
- Flight Unit Test Data
- Launch Support Plan
- Data Reduction and Analysis Plan
- Schedule (to be updated biweekly)
- Monthly Technical Progress Reports including Quality Status Reports
- Monthly Financial Reports