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# THE STANDARDIZED FUNCTIONAL SUPPORT SECTION FOR THE SMALL ASTRONOMY SATELLITE (SAS)

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16. Abstract

The standardized functional support section for the improved Small Astronomy Satellite (SAS) spacecraft, which can be used virtually without change for a wide variety of experimental packages and missions, is described. This functional support section makes the spacecraft remarkably flexible 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- or earth-locked pointing of side-viewing experiments. It features a reprogrammable telemetry system, a delayed command system, and an improved control system. Experiments can be built independently and attached to the SAS spacecraft just prior to final acceptance testing and launch. The spacecraft subsystems are described in detail. Included are a summary of the spacecraft characteristics, special design considerations, project reliability requirements, and environmental test conditions. It is intended that this new functional support section afford virtual off-the-shelf availability of the SAS spacecraft to independently built experiments, thus providing quick response time and minimum cost in meeting a wide variety of experimenter needs.

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## FOREWORD

In the last 10 years, there has been a truly remarkable flowering of high-energy astrophysics including both X- and $\gamma$-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 160 X-ray objects have already been identified, most having a totally unexpected nature. Some objects have variations which consist of a combination of periods, and still others are seen to have bursts 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 $\gamma$-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 for detailed observations over a wide range of energies and with finer time resolution than SAS-1. Also, a star-locked pointing capability to be tried experimentally on the SAS-C mission should enhance greatly the sensitivity of some X-ray experiments by permitting continuous observations of selected objects over longer periods of time than previously possible.

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 E. Fichtel
SAS Project Scientist

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# THE STANDARDIZED FUNCTIONAL SUPPORT SECTION FOR THE SMALL ASTRONOMY SATELLITE (SAS)

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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 infrared regions of the electromagnetic spectrum, both inside and outside our galaxy. The SAS spacecraft have been launched by the Italian Government for the United States from the San Marco Launch Platform into low-altitude equatorial earth orbits. This range is owned and operated by the Centro Ricerche Aerospaziali of the University of Rome and is under the direction of Professor Luigi Broglio. The first satellite of this series, SAS-1 (Explorer-42, Uhuru), was launched on December 12, 1970; since then, the information on X-ray sources has increased immeasurably. The spacecraft experiment has discovered 125 new X-ray sources and has obtained significant new information on the sources already known. The second spacecraft in the series, SAS-2 (Explorer-48), was launched from the San Marco Platform on November 15, 1972, with a digitized spark-chamber gamma-ray telescope. This experiment has collected significant data on celestial gamma radiation. Future SAS spacecraft will be able to carry even more complex experiments and hopefully will contribute substantially to the science of astrophysics.

In anticipation of second-generation experiments requiring improved spacecraft capabilities, particularly in the basic support needed for a space experiment (power, telemetry, commands, and control), it was found appropriate to redesign the SAS standardized functional support section. The new design allows increased flexibility and provides the capability of flying a diverse variety of small payloads without making changes to the standardized section. Thus, the spacecraft functional support section can have an off-the-shelf availability; only the transition ring, to which is bolted the experiment section, will be fabricated for each mission.

As experiment requirements have increased, launch vehicle capability has likewise been upgraded since 1970, allowing heavier, more complex experiments to be placed in orbit by low-cost launch vehicles.

## OVERVIEW OF SPACECRAFT FUNCTIONAL SUPPORT SECTION

The SAS spacecraft (shown in figure 1 in the SAS-C configuration inside the Scout heat shield) is designed to have a clean interface, both mechanically and thermally, between the basic spacecraft functional support section and the experiment payload. This interface occurs at STA 13.27 for SAS-C. The experiment-spacecraft transition ring, which is attached to the top of the spacecraft functional support section shown in figures 2 and 3, will be changed between missions, thus providing maximum flexibility at minimum cost. The height of this transition ring can be changed to allow for mission-unique hardware on the upper deck and its bolt-hole configuration can be changed for easy mounting of each individual experiment.

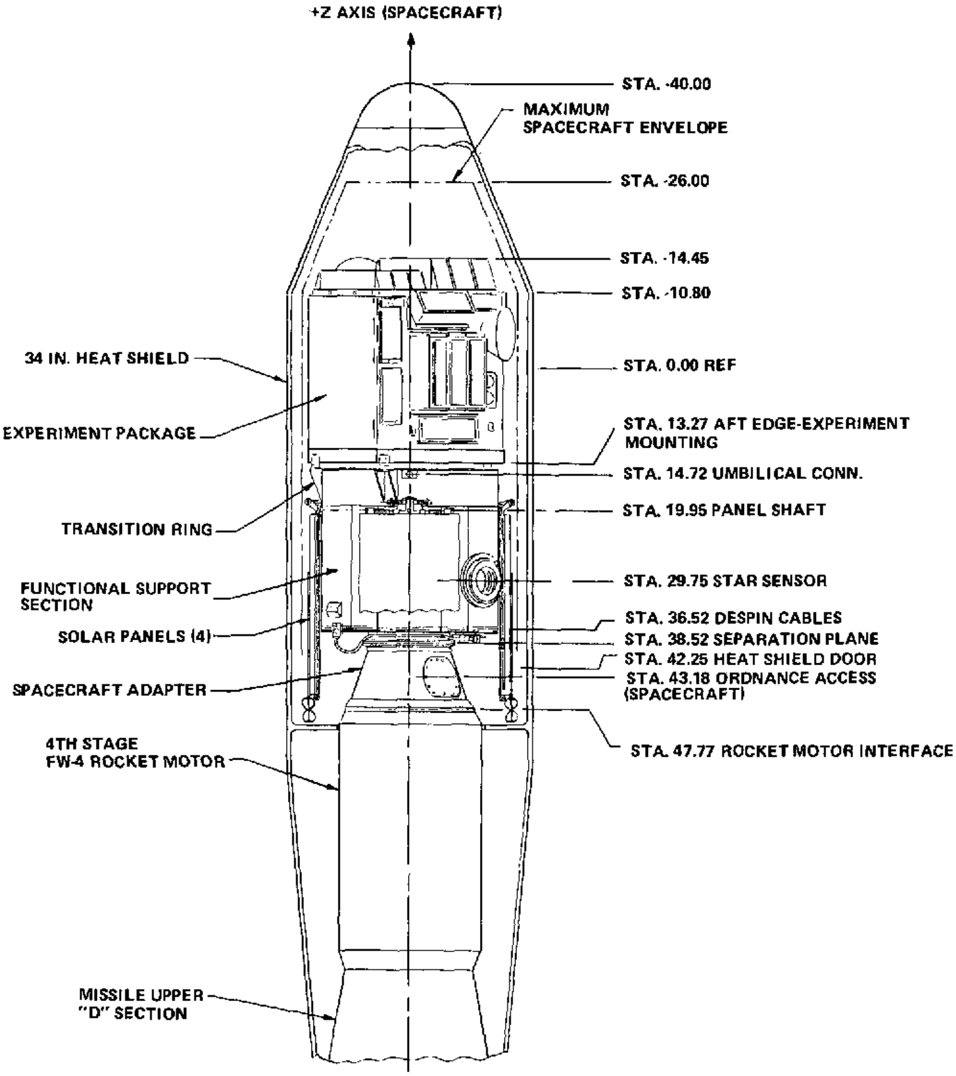



Figure 1. SAS-C in Launch Configuration Inside Scout Heat Shield

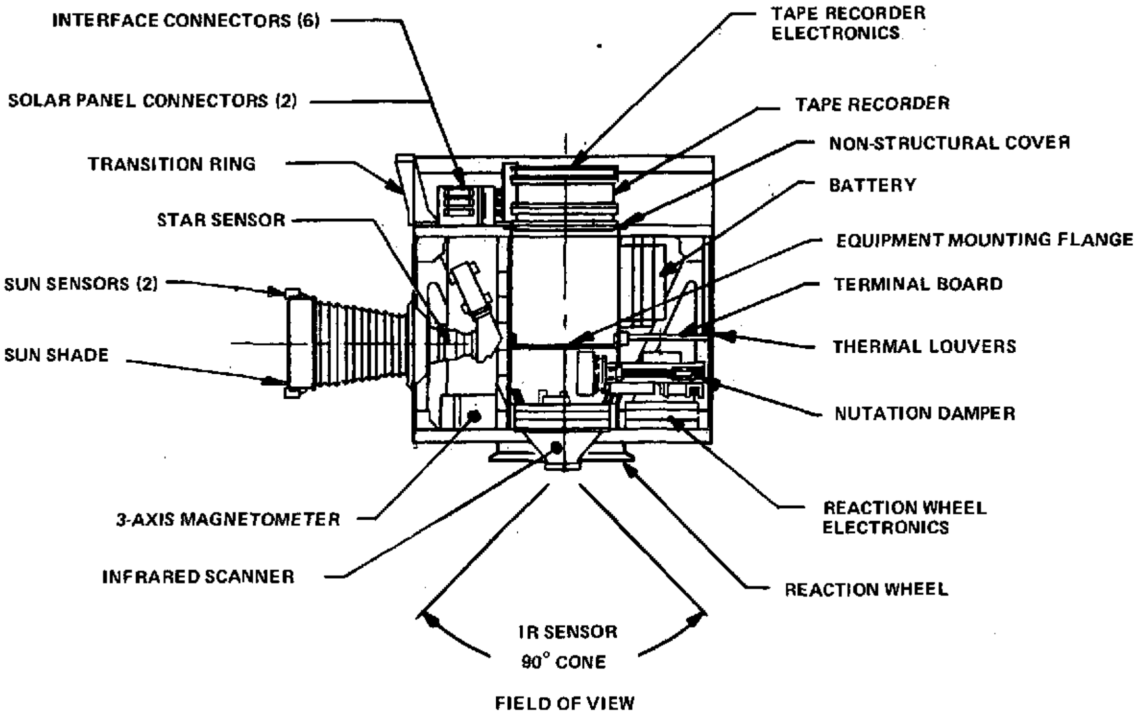



Figure 2. SAS Spacecraft Functional Support Section

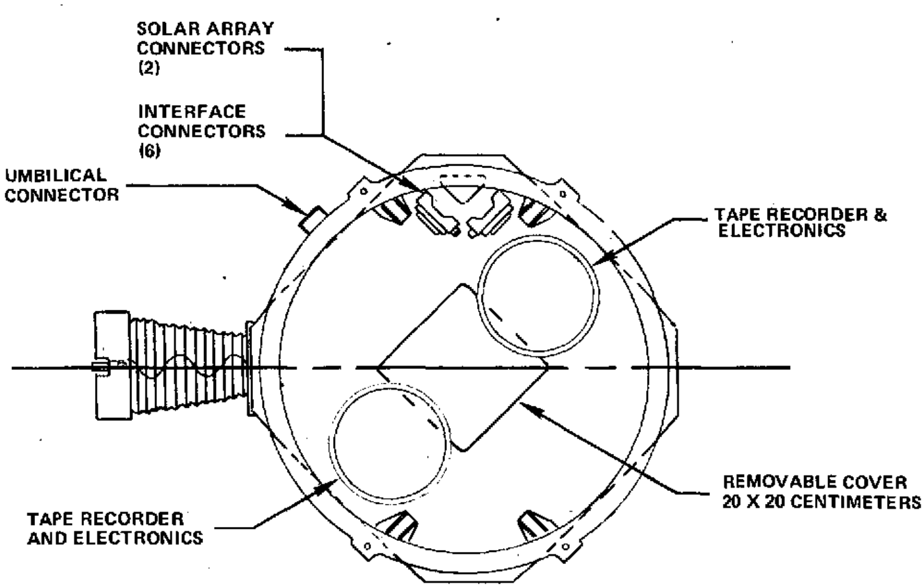



Figure 3. View of Upper Deck of SAS Spacecraft Functional Support Section

Conveniently located on the upper deck are the experiment-spacecraft interface connectors (figure 3). For low, earth-orbiting missions, such as the SAS-C mission, 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 collocated with the experiment for accuracy.

Without any change to the basic spacecraft functional support section, this design, normally planned for a Scout-launched low-earth orbit, can be adapted for a Delta-launched synchronous orbit; 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 collocated with the experiment. For the various missions thus described, all changes are made only to hardware on the upper deck.

The lower part of the thermally controlled spacecraft functional support section (figure 4) contains the standardized systems required to provide basic spacecraft functions. They are:

- Power system with a rechargeable battery with charge control and regulator systems;
- Redundant command system with receivers, decoders, and a stored command capability;
- Programmable telemetry system with VHF and S-band transmitters;
- Magnetically-torqued commandable control system; and
- Attitude determination system with magnetometers, sun sensors, and a star sensor.

The commandable control system 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 upper deck of the functional support section are four sets of solar-array paddles. These can be seen in figure 5 which shows the SAS-C spacecraft in orbital configuration. Each set of paddles is comprised of three panels which fold against each other in the shroud during the launch phase (see figure 1) and are held in place by despin cables. After orbit insertion, the cables are released and the spacecraft is despun, allowing the solar panels to deploy and unfold into their orbital configuration. Either set of opposing panels can be rotated through a 90-degree arc by a synchronous motor. Each of two motors drives one paddle directly and the opposing paddle through a cable threaded through channels in the upper deck.

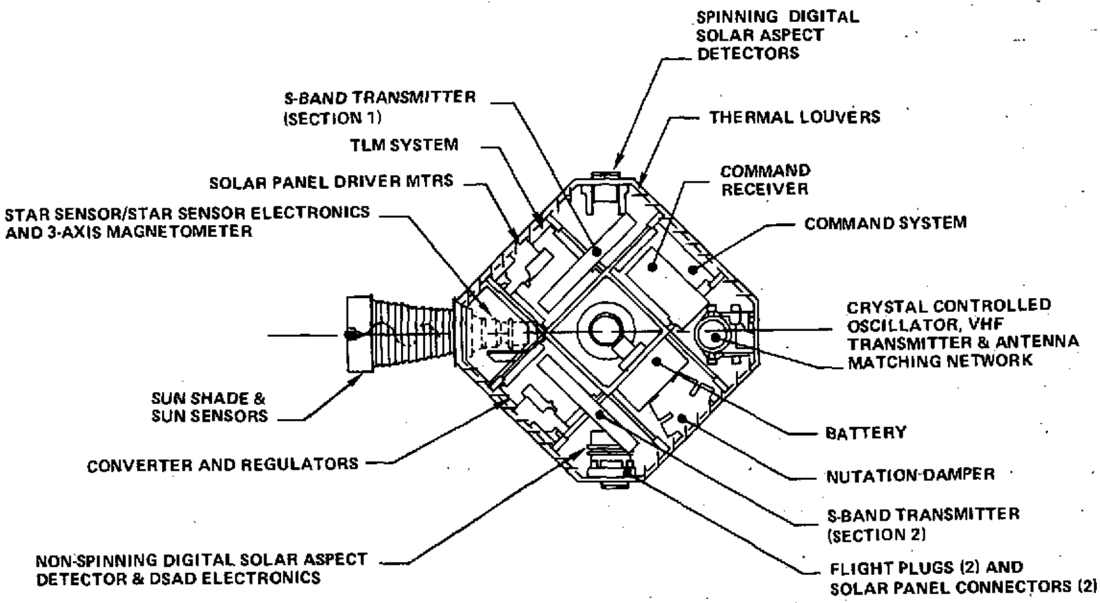



Figure 4. Lower Section of SAS Spacecraft Functional Support Section

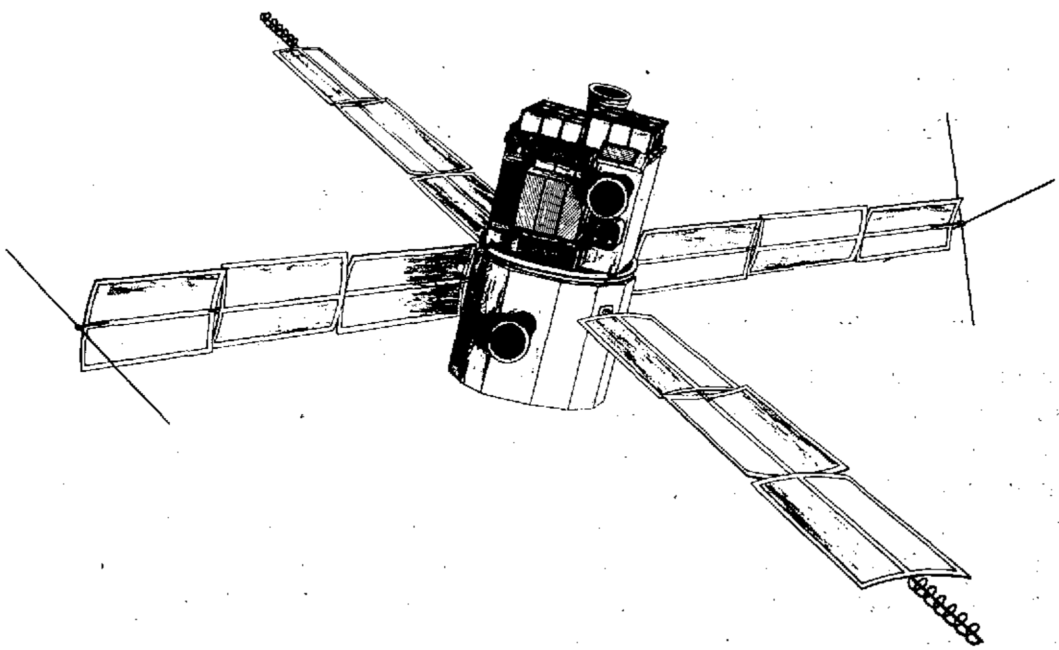
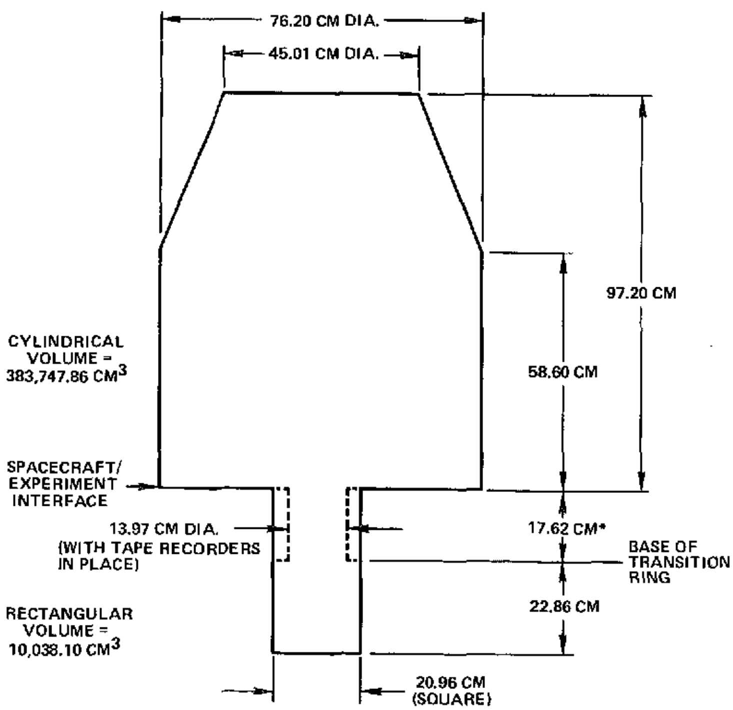


Figure 5. SAS-C in Orbital Configuration

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

Interconnections between experiments and the functional support section are provided through a spacecraft harness. The SAS design permits the harness below the upper deck to remain the same from mission to mission. If any changes should be required for particular missions, only the upper deck harness is affected.

Another feature of the standardized functional support section, which allows for a variety of missions, is the hollow center of the spacecraft structure seen in figures 2 and 7b. At the lowest part of this center 20-centimeter-square section are the reaction wheel and part of the nutation damper. A volume above them, 20 centimeters square by about 40 centimeters high, is kept clear for future missions which might include a telescope longer than allowed by the space inside the Scout heat shield above the spacecraft-experiment interface. To preserve this additional space for such a telescope, a 20-centimeter-square opening has been provided in the upper deck, and all cabling for solar-paddle rotation has been routed around this center space. The available center volume is shown in figure 6.




* THIS HEIGHT CAN VARY FROM MISSION TO MISSION.

Figure 6. SAS Experiment Envelope Dimensions

## STRUCTURAL AND THERMAL DESIGN OF THE FUNCTIONAL SUPPORT SECTION

The basic structure of the SAS functional support section is a cylinder 66 centimeters in diameter by 61 centimeters high, weighing about 120 kilograms. The upper and lower decks are made of aluminum honeycomb. Figures 7a and 7b show the basic spacecraft structure. Loads, generated by the experiment attached to the transition ring 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 this 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 table 1 (pp. 14-15). For example, a 75-kilogram experiment can be launched by a four-stage Scout vehicle from the San Marco equatorial launch platform into a circular orbit with an altitude of approximately 550 kilometers. The volume available for Scout-launched experiments can be determined roughly by looking at figure 6.

Three of the four rectangular bays, seen from the top in figures 4 and 7b, contain electronics for the command, control, telemetry, and power systems. This circuitry is packaged in stacks of magnesium castings approximately 16 by 16 by 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 reaction wheel which has an infrared sensor whose field-of-view is outside the base of the spacecraft. Figure 4 shows the locations of the crystal controlled oscillator; S-band and VHF transmitters and antenna matching network; spinning and nonspinning digital solar aspect detectors; three-axis magnetometer; star sensor; flight plugs; solar panel connectors; solar panel drive motors; battery; nutation damper; telemetry system; command system; and command receiver.

Around the periphery of the spacecraft functional support section, behind thin radiator panels, are the 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: isolation of the support section from the experiment. This is accomplished by a multilayer blanket, while the attach points are isolated by fiberglass standoffs and bolthead insulators. The separation of the functional support section and experiment section assures that true off-the-shelf availability of the spacecraft can be maintained.

The thermal louvers, made of thin sections of polyurethane open cell foam covered with gold coated Kapton, are opened and closed by bimetal actuators whose positions are determined by the ambient temperature. Banks of these 3-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 functional support section. Heaters, internal to the functional support section, can be used as necessary. They can also provide a backup if any of the thermal louvers should become stuck in the open

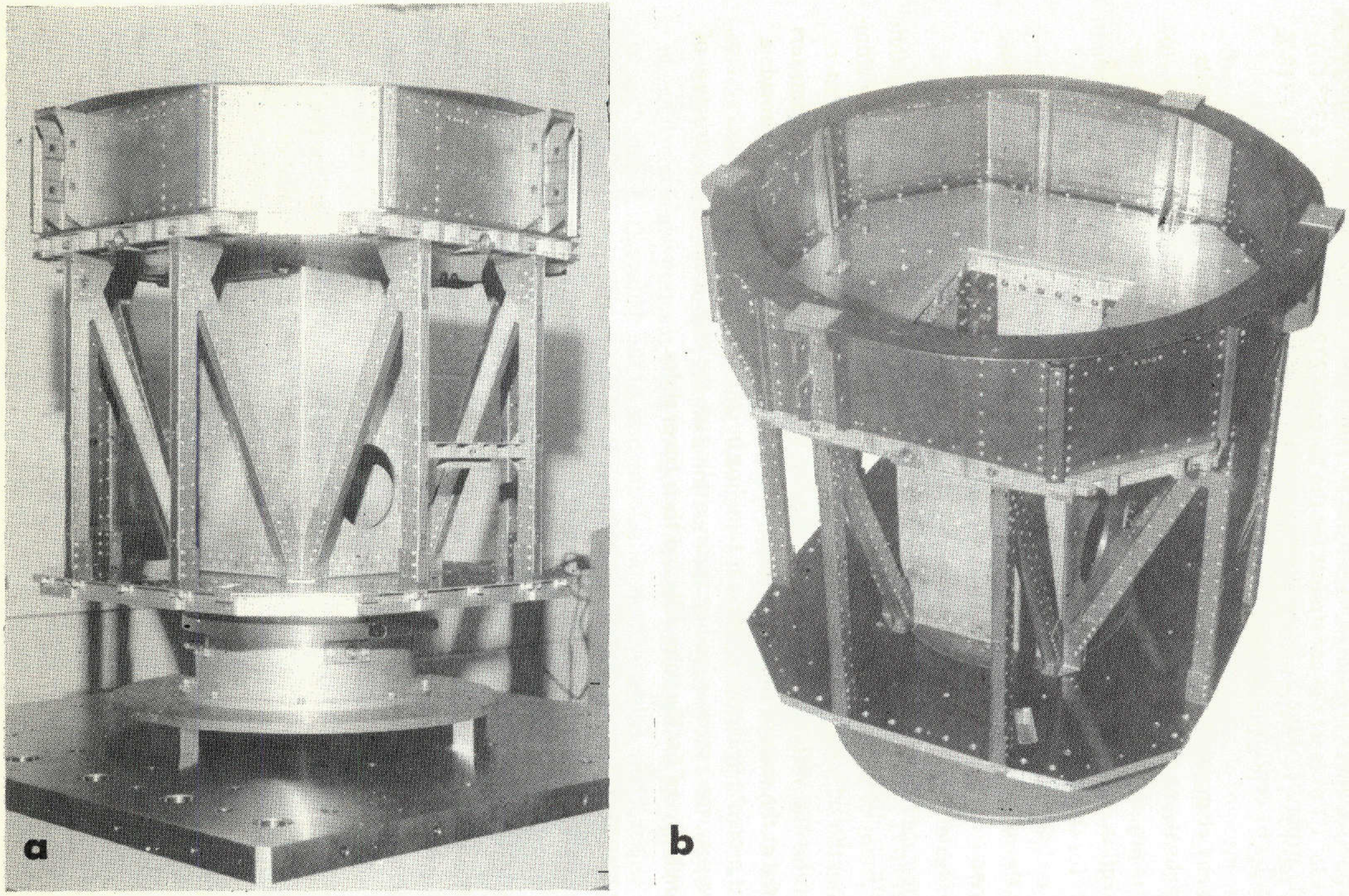



Figure 7. SAS Spacecraft Structure

position. However, the current design would not need to utilize these heaters to maintain the spacecraft functional support section temperature within the range of +278 to +308 K (+5° to +35° C) to meet the constraints for the battery (273 to +308 K (0° to +35° C)), the tape recorders (+278 to +308 K (+5° to +35° C)) and the electronics (263 to +318 K (-10° to +45° C)) in a low-earth equatorial orbit.

Without any changes, the design provides adequate thermal control in low-earth orbits having 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 ($\alpha = 0.15$, $\epsilon = 0.80$) for the space radiators, aluminized mylar ($\alpha = 0.15$, $\epsilon = 0.80$) for the multilayer blankets, gold-coated Kapton ($\alpha = 0.04$) for the thermal louvers, and Martin hardcoat ($\alpha = 0.90$, $\epsilon = 0.90$) 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 +283 K (+10° C).

The solar-array panels are constructed of thin aluminum honeycomb, curved to fit within the limitations of the Scout heat shield, and covered with a thin film to insulate the aluminum from the solar cells. Each of three panels is approximately 36 by 195 centimeters. Antennas for the VHF command system and the VHF and S-band telemetry transmitters are located at the ends of the outer panels, which also have integral 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 8, consists of four rotatable paddles with solar cells on both sides; redundant charge regulating and monitoring circuitry (CRAM); an 8-ampere-hour hermetically sealed, rechargeable, nickel-cadmium battery; and various regulators and converters. Orbit average power is a function of altitude and orientation. SAS-C, in its circular, equatorial orbit of less than 500 kilometers in altitude, with a requirement for random orientation, represents a worst-case condition for this power system.

Figure 9 presents representative curves of power as a function of sun angle for this orbit based on the assumption that the two sets of solar paddles are either parallel or at 90 degrees 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 mission operations when the spacecraft is at higher altitudes, when spacecraft orientation is restricted to more favorable sun angles, or when paddle orientation is optimized between the 0- and 90-degree limits set for this calculation. These procedures allow a higher total of incident sunlight. Without making any other changes to

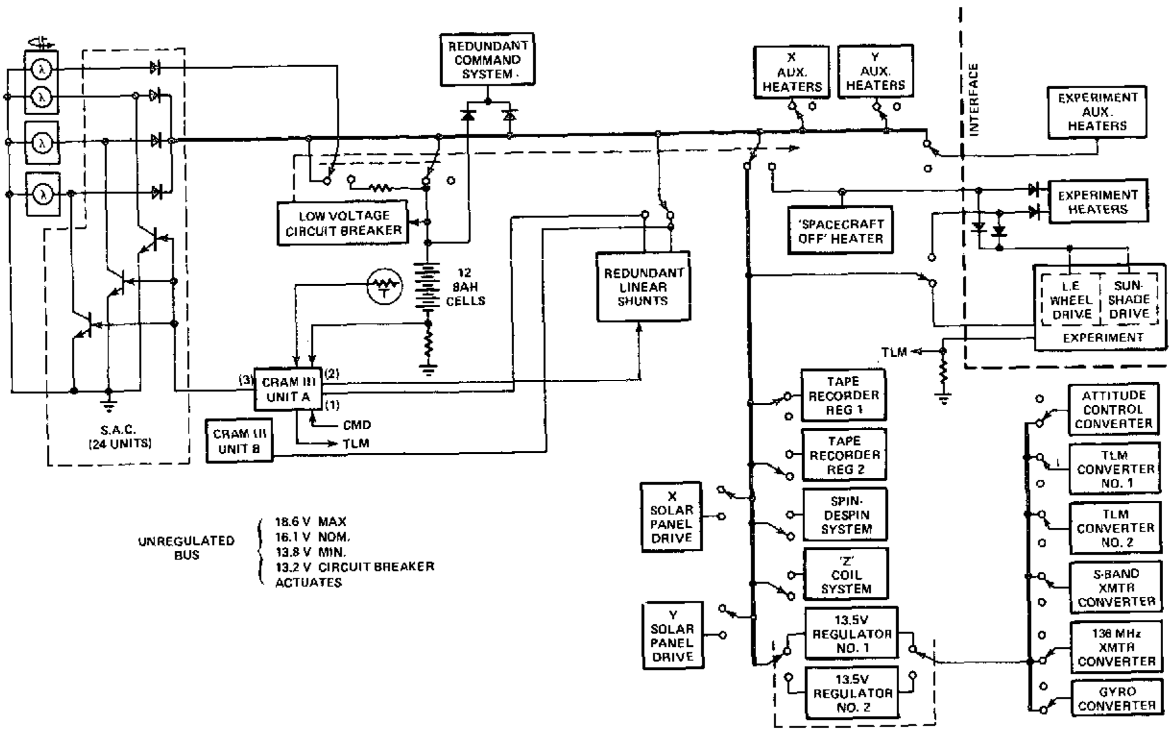



Figure 8. SAS Power System - Block Diagram

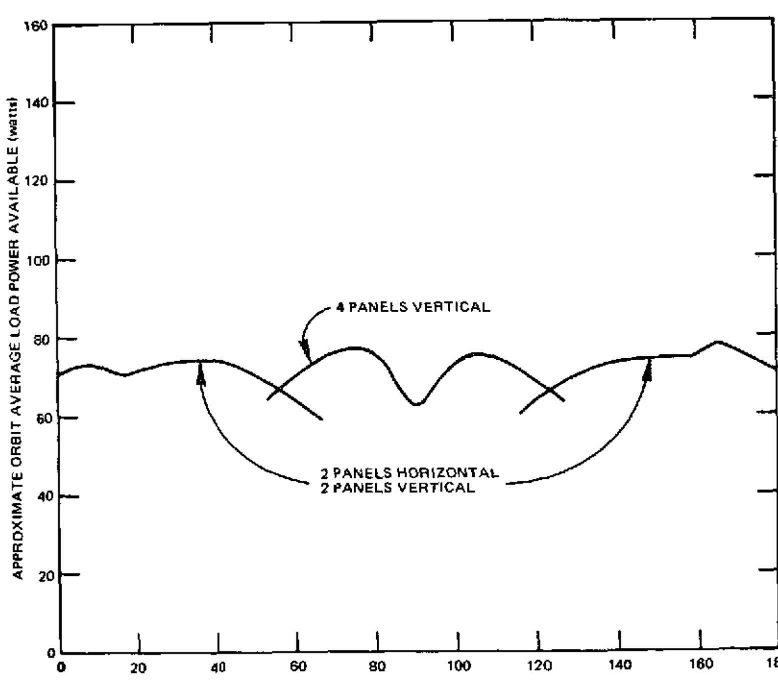



Figure 9. Power Available, SAS (Assuming 10% Increase Due to Albedo)

the system, stored power for the nighttime portion of the orbit or for peak requirements during the day can be increased by 50 percent by using a 12-ampere-hour battery rather than an 8-ampere-hour battery. To plan for this possibility, adequate volume has been reserved for this larger battery.

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 functional support section, such as star cameras for a pointed mission, this additional 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 by 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 reflecting 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 the energy taken from the battery by the load and allows it to be replaced. After the battery is fully charged, the current to the battery is reduced to a trickle charge level by shunting the current received from the solar arrays. If spacecraft temperatures are low, this current can be used in heater circuits; or, the outputs from the solar arrays can be switched off entirely, leaving only the linear shunts, acting as a vernier, to dissipate power within the spacecraft.

Raw power at 16.1 ± 15% volts is provided to the experiment and to the spacecraft functional support section. Redundant regulators drive converters that provide power to the attitude system, the telemetry system, the VHF and S-band telemetry 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.

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 CRAM failure in the solar-only mode, 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 pulse-code modulated (PCM) command system (figure 10) designed to conform to the NASA/Goddard Space Flight Center Aerospace Data Systems Standards

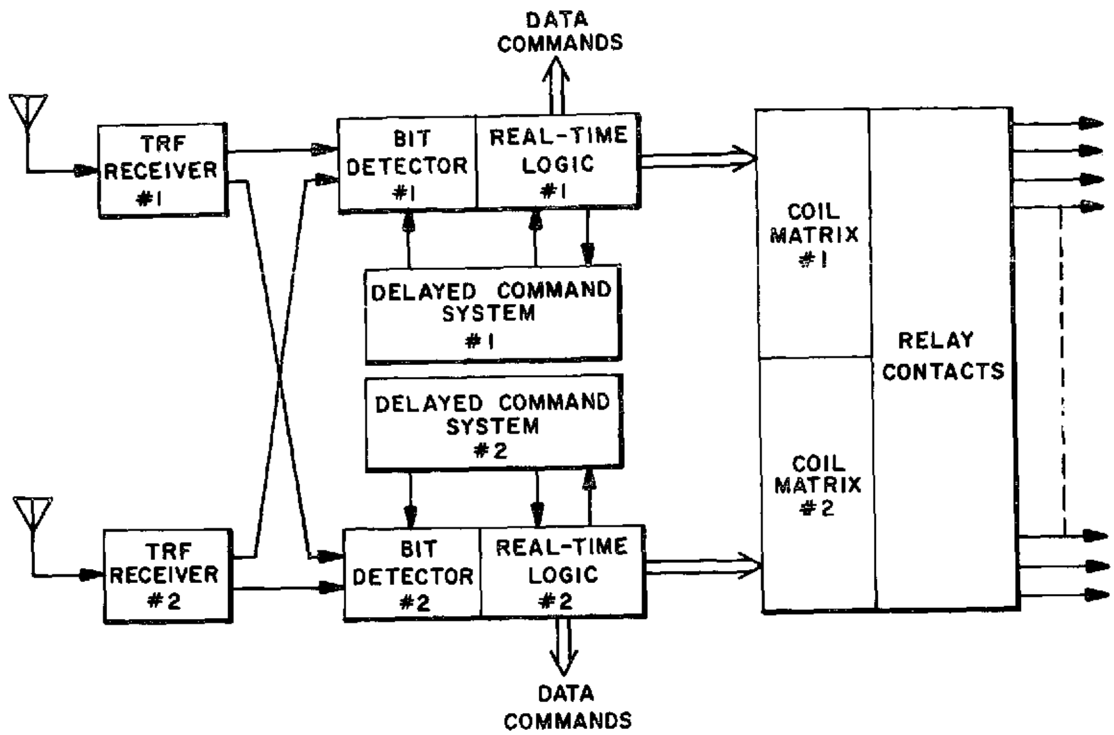



Figure 10. SAS Command System - Block Diagram

(X-560-63-2). Redundancy begins at the antenna and dc-to-dc converter power inputs and ends at the relay contacts. At the point of receipt of a data command by 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) FSK signal which is 50 percent amplitude-modulated by a sine wave bit synchronizing signal that is 180 degrees out of phase with the bit pattern. A command word consists of 64 bits in one 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 telemetry system or whether it is a relay or short (24 bit) 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 of either data or relay commands, 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 3 seconds and 2.4 hours per command, 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 the first delay is 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.

Delayed commands are not destroyed as they are used, but are recycled into storage. This permits 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 spacecraft functional support section has been designed to fly in any orbit. For high-energy astronomy missions, the preferred orbit is equatorial in order to avoid the South Atlantic Anomaly where the trapped particles in the radiation belts 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. Future high-energy astronomy experiments, such as the one planned for SAS-C, will also be launched from the San Marco Range.

An equatorial orbit presents a special problem for data collection because the only two stations in NASA's Spaceflight Tracking and Data Network (STDN) that can view a spacecraft in a low-equatorial orbit are at Quito, Ecuador, 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 $6.0 \times 10^6$ 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 mode and 30 centimeters per second in playback mode. 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 at a rate of 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, it is necessary to rely on real-time data acquisition by stations,

## Table 1
## Summary of Characteristics of Small Astronomy Satellite (SAS)

### SIZE

| | |
|---|---|
| Spacecraft Functional Support Section: | Cylinder 74 cm in dia. by 64 cm high, without solar panels<br>4.7 m tip-to-tip with solar panels |
| Experiment Payload using Scout F launch vehicle: | Cylindrical area 76 cm in diameter by 58 cm high, plus 39 cm additional height with decreasing diameter.<br>More length is available by using existing volume in spacecraft functional support section. (See Figure 6.) |

### WEIGHT

| | |
|---|---|
| Spacecraft Functional Support Section: | 120 kg |
| Experiment Payload using Scout F launch vehicle: | |

| Exp. Pay-Load Wgt. | Launch Site | Altitude (Circular Orbit) | Inclination |
|---|---|---|---|
| 75 kg | San Marco | 550 km | 3° |
| 63 kg | Wallops Island | 550 km | 38° |
| 29 kg | Western Test Range | 550 km | 90° |

### ORBIT AVERAGE POWER

| | |
|---|---|
| Spacecraft Functional Support Section: | 40 watts |
| Available for Experiment: | 25 watts minimum |
| 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 1.005 MHz | |
| Nominal: | 981 bps | |
| Other: | 123, 245, 491, 1962, 3925, 7851, and 15,703 bps | |
| Storage Method: | Endless loop-tape recorder | |
| | Capacity: | $6 \times 10^6$ bits |
| | Record rate: | 981 bps |
| | Playback rate: | 19,629 bps (20:1) for approx. 5 min. |
| Transmitters: | 136 MHz (90 kHz bandwidth) with two power modes (0.25 W and 1.5 W)<br>2250 MHz (1 MHz bandwidth) with two power modes (1 W and 4 W) | |
| Encoder: | 8-bit analog-to-digital converter (provides better than ±0.2% accuracy for signals up to 10 Hz) | |

## Table 1 (continued)

| | | |
|---|---|---|
| Input Signal Levels: | Analog: | ±10.0 v, ±7.5 v, ±5.0 v, ±2.5 v, ±1.25 v, ±0.5 v, ±0.25 v |
| | Digital: | +4 v for a "1" and +0.3 v 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)<br>One reprogrammable, nonvolatile memory | |

TRACKING

| | |
|---|---|
| Positional Accuracy from Minitrack System: | 10 km |

COMMAND SYSTEM—Conforms to NASA/GSFC "Aerospace Data Systems Standards" for PCM command systems using VHF carrier

| | |
|---|---|
| Spacecraft Functional Support Section: | 46 "on" and "off" relay commands |
| Experiment: | 10 "on" and "off" relay commands<br>24 bit data commands are routed to the experiment, and to attitude control system, power system, telemetry system, and delayed command system. The attitude control system can reroute 24 bit commands using the first four bits for routing and 20 bits for commands. Data commands are 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: | $<1^\circ$ relative to known position |
| Maneuver Rate: | $\sim 1^\circ$ per min |
| Average drift of Z-axis: | $\leqslant 2^\circ$ per day (assumes residual magnetic moment of experiment to be $<1000$ pole-cm) |
| Spin Rate: | 0 to 10 rpm |
| Nutation angle (coning): | $\leqslant 0.1^\circ$ |
| Nutation period: | $\geqslant 1$ min |
| Wheel momentum: | 40 kg-m$^2$-rpm at 1500 rpm |
| Wheel speed control: | Error signals from (1) Internal tachometer for fixed speed<br>(2) Earth sensor for $1^\circ$ earth lock<br>(3) Rate gyro for controlling linearity of rotation to $\pm 0.3^\circ$ per hour<br>(4) Star tracking camera for star-lock mode (1 arc-minute design goal) |

ATTITUDE DETERMINATION SYSTEM

| | |
|---|---|
| Three-axis magnetometers and sun sensors: | $\pm 3^\circ$ |
| Star sensor: | ±1 arc-minute |

other than STDN, around the equator, for example, the French station at Kourou, French Guiana.

The telemetry system on the improved SAS spacecraft (figure 11) is much more sophisticated than its predecessors on SAS-1 and SAS-2, although it has the same two basic modes of operation: (1) recording and transmission of real-time data and (2) playback of stored data. A major improvement is that both of these modes can be operated simultaneously; real-time data are transmitted on the VHF telemetry transmitter and playback data on the S-band telemetry 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. A higher frequency (2250 MHz) S-band telemetry transmitter has been added to the SAS telemetry system and 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 attribute of this sophisticated telemetry system is that the telemetry format is reprogrammable while in orbit. There is a selection of fixed formats to which the system can always be switched by ground command. This capability permits acquisition of data while reloading the stored program and serves as a backup. Normal operation will permit a choice of programs to be stored 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 of sampling of the available data sources. These data sources may be provided in either analog or digital form, with digital data being prime and analog usually reserved for monitoring housekeeping functions. Data normally will be stored on the tape recorders at only about 1000 bits per second; this data will not be stored when 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 the SAS-C mission, there will be 816 bits in a minor frame and 256 minor frames in a major frame. To facilitate ground operations, 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 one 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 preset 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

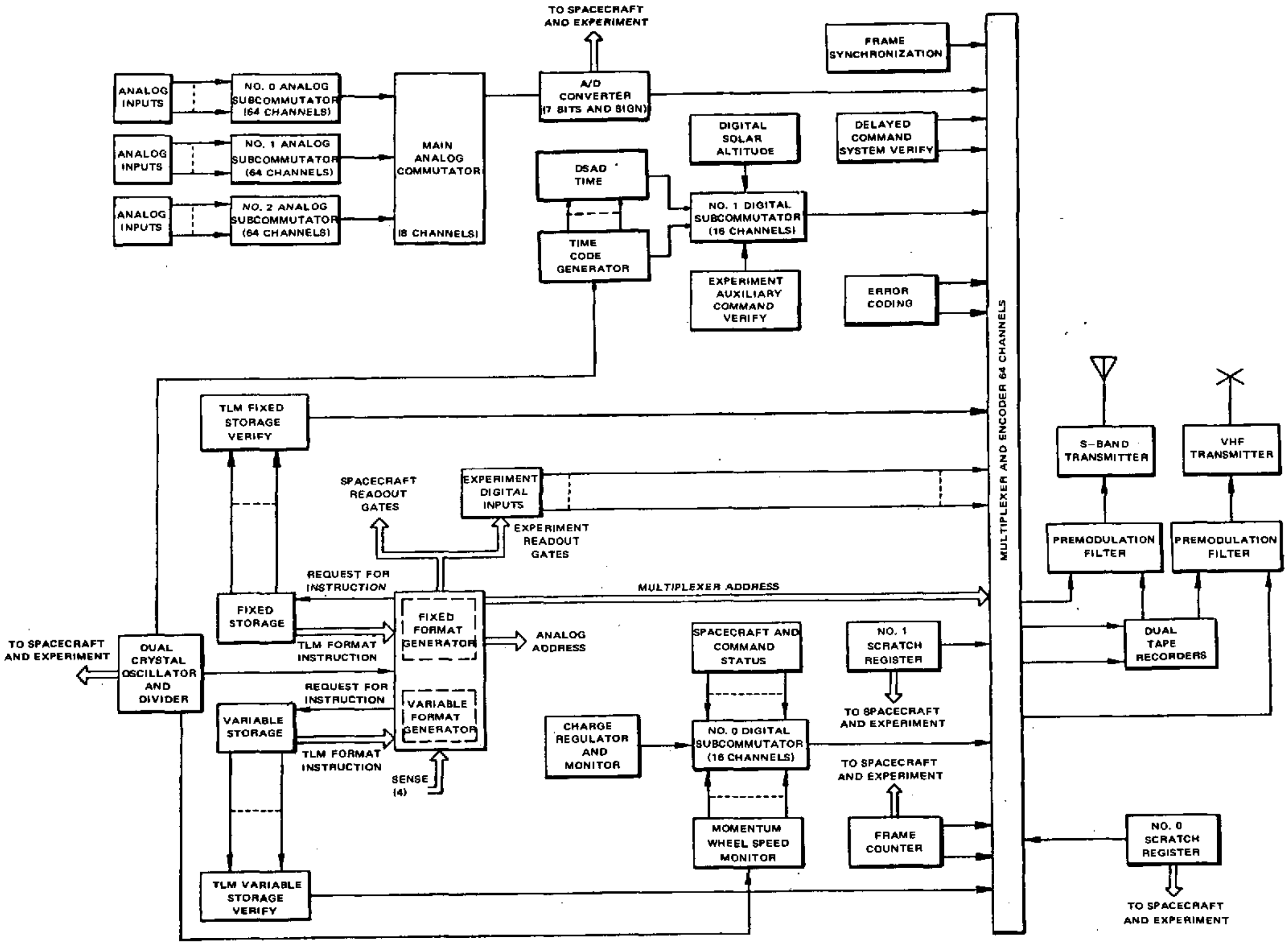



Figure 11. SAS Telemetry System - Block Diagram

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, designed using the Texas Instruments 54L family of integrated circuits plus some additional special components, 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 coding, tape recorders, premodulation 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 5.025-MHz crystal. Stability of the basic oscillator is $10^{-10}$ per hour over a temperature range of 273 to 323 K (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 functional support section readout gates, thus allowing all the data to be multiplexed into the proper sequence.

The scratch registers have three purposes: (1) 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; (2) to control the operating mode of some circuits; and (3) to make it possible 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 nonrepeating readout for approximately one year. The 24-bit stored output is transmitted via one of the digital subcommutators every 16 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, for example, 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. The analog inputs are used primarily for monitoring housekeeping information such as voltages, currents, and temperatures, although they can be used for other low-sampling-rate data. The system can accept a variety of input voltages (see table 1) between plus and minus 10 volts by attenuating them to ±0.254 volt before they reach the junction field effect transistor switches used in the analog subcommutators and the main analog commutator. These switches multiplex the input data and the outputs of the analog subcommutators to 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 16 kHz and has an aperture time of 2 milliseconds, which provides an accuracy of ±0.2 percent for a signal below 10 Hz over a temperature range of 248 to 328 K (-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) to a split-phase waveform, free from switching transients. Outputs from the encoder go to each of the on-board tape recorders or to either or both of the transmitters through premodulation filters. The filters keep the transmitted bandwidth within its allocation as prescribed by NASA/Goddard Space Flight Center Aerospace Data Systems Standards. This allocation is ±15 kHz for real-time data or ±45 kHz for playback data through the VHF telemetry transmitter or ±1 MHz at S-band.

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

## CONTROL SYSTEM

As with its predecessors, a very important feature of the SAS spacecraft is its control system. An evolutionary development is planned which would 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 system is the elimination of the need for expendables, such as gas systems with their inherent life limitations. Passive nutation damping controls coning about the Z-axis.

Experiments can view along or perpendicular to the Z-axis. In the latter case, if the spacecraft is rotated slowly about its Z-axis, then a swath is swept on 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 on 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 (figure 12) whose speed is controlled by an error signal generated by an external source. When a rate gyro is used, as will be done on SAS-C, the rotation rate about the Z-axis can be made quite constant; 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 wheel speed control 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 refinement of the basic control system, with the addition of two more small reaction wheels, two additional star-tracking cameras, and the associated electronics, will provide a three-axis stabilized spacecraft; thus, the evolutionary development mentioned above.

Figures 13a and 13b show the various components of the control system. Z-axis orientation is accomplished by energizing a torquing coil. This electromagnet interacts with the earth's magnetic field to precess the spin axis. 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 that has 10-pole-centimeter steps.

Spinning and despinning is accomplished by amplifying the outputs of the X- and Y-axis magnetometers and driving the Y- and X-axis coils, respectively. This system, in conjunction with the earth's magnetic field, operates essentially as a motor to increase or decrease the rate of rotation about the Z-axis. With maximum dipoles of $1 \times 10^4$ pole-centimeters,

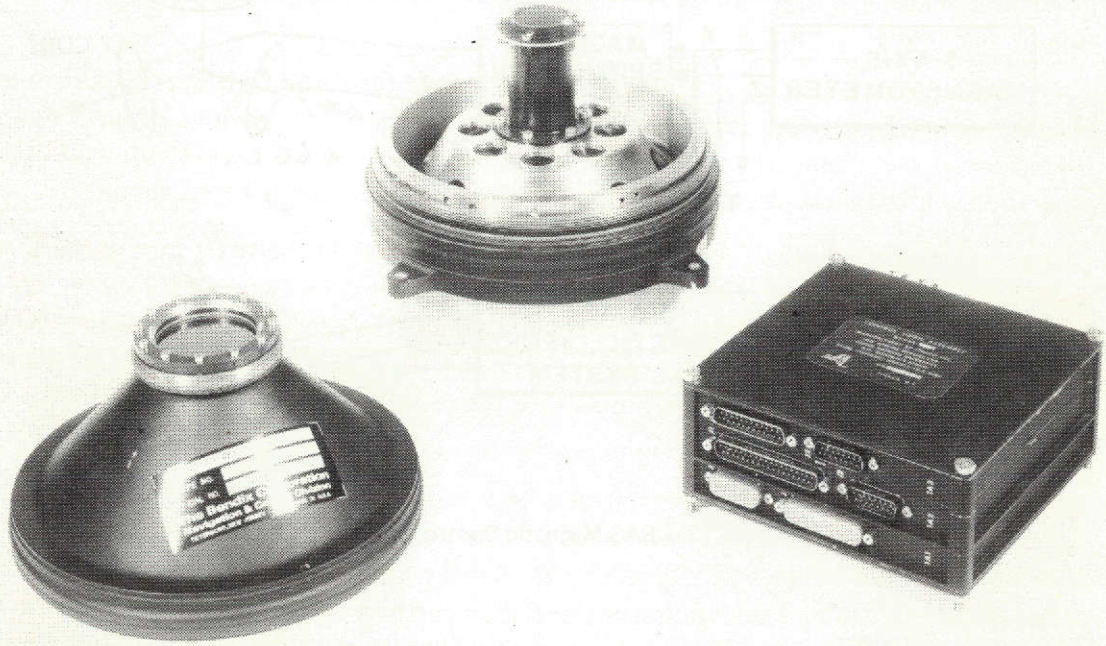


Figure 12. The SAS Variable-Speed Reaction Wheel, Disassembled
(Cover, Wheel, and Breadboard Electronics)

the maximum rate of change of spin is about 10 rpm per day. This system also has bias dipoles for compensation of residual magnetism along the X- and Y-axes of the spacecraft, ±500 pole-centimeters 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 permits the planning of 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 field. The advantage of using this capability is to minimize the time and power required for torquing maneuvers. Magnetic field data needed for this calculation are stored in a computer on the ground. Commands are subsequently relayed to the spacecraft and stored by the command system for execution at times during the orbit determined by the computer calculations.

The spin-despin system is also used to exchange momentum when the reaction wheel has reached saturation. The primary source of stability for the SAS spacecraft is the variable-

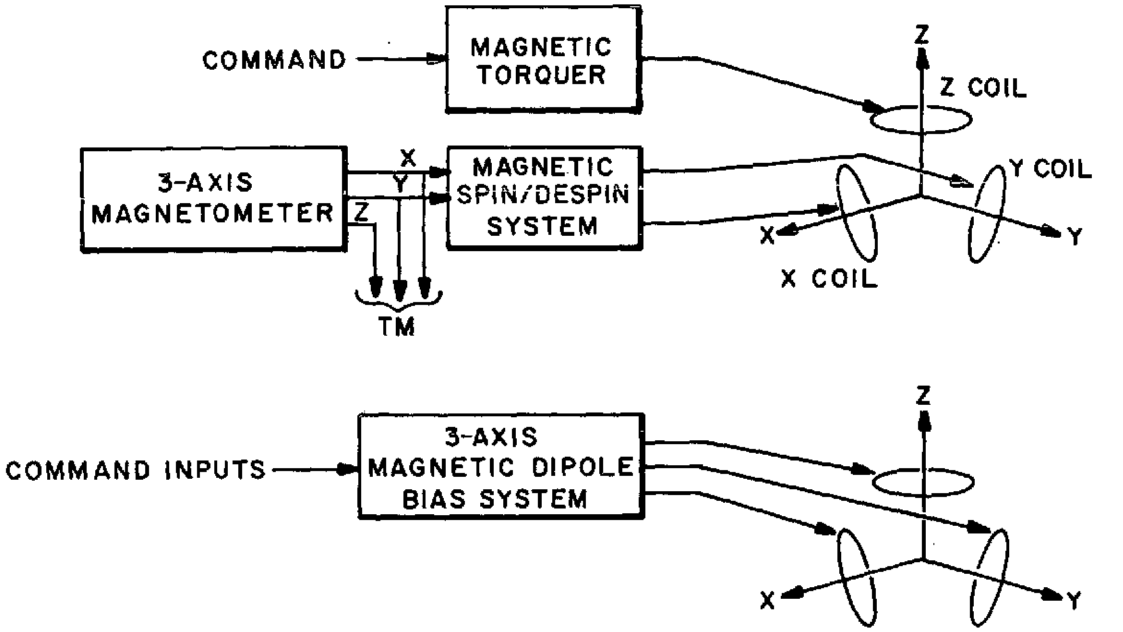



Figure 13a. SAS Magnetic Control System

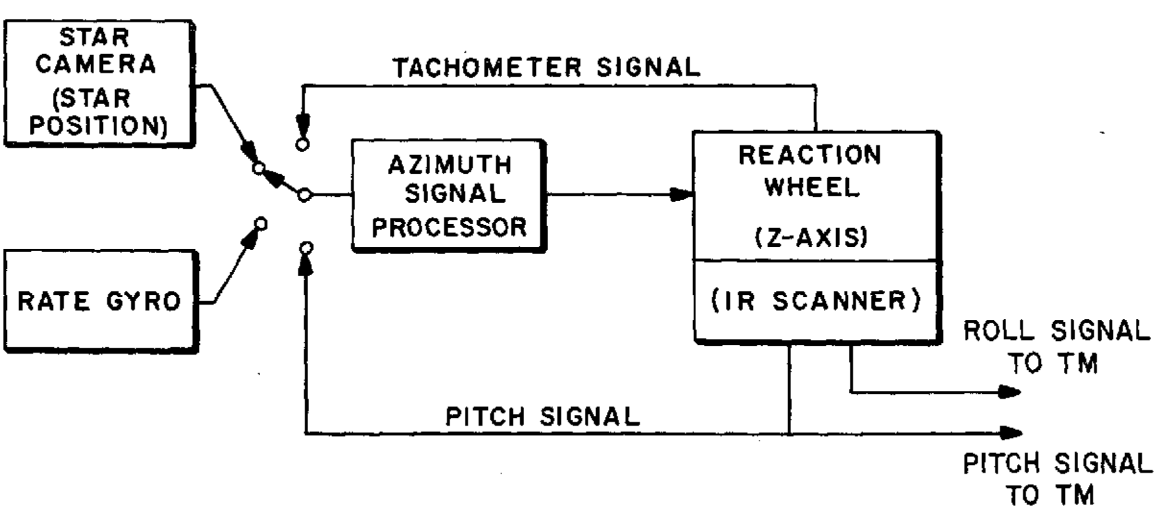



Figure 13b. SAS Azimuth Angle and Azimuth Rate Control System

speed reaction wheel (figure 12). Driven by an ac induction motor, this reaction wheel produces a momentum of 40 kg-m$^2$-rpm at its nominal speed of 1500 rpm. This momentum obviates the need for spinning. Thus, the spacecraft can rotate slowly to scan the celestial sphere or stop its rotation in order to provide a longer data collection period. Having a variable speed is a significant improvement over the SAS-1 and SAS-2 satellites. Their rotation about the Z-axis was slightly nonlinear because the fixed-speed momentum wheel was unable to compensate for external torques such as those created by gravity-gradient and aerodynamic effects on the spacecraft and its solar paddles.

The speed of the SAS reaction wheel will be controlled under normal operating conditions by an air-bearing, rate-integrating gyro with good temperature control. The speed of this gyro is 23,264 rpm with a stability of more 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 (figure 14) is 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 is 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 degree. The motion of the nutation damper pendulum is detected and telemetered by a 7-bit optoelectronic system.

## ATTITUDE DETERMINATION SYSTEM

The attitude of the spacecraft can be determined to an accuracy of a few degrees by magnetometer and sun sensor data. A finer determination of about 30 arc-minutes can be made with data from the star sensor located in the spacecraft functional support 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 functional support section. This differs from the arrangement on SAS-1 and SAS-2; the X- and Y-axis magnetometers were located on two of the solar paddles, and the Z-axis magnetometer was located on the body of the spacecraft. With all three magnetometers together inside the functional support section, there are both advantages and disadvantages; the orthogonality of the three is known and can be maintained with more precision, but they are affected by biases created by other spacecraft systems.

The digital solar attitude detection (DSAD) system provides for both spinning and non-spinning operation. The nonspinning portion has two 0.5-degree resolution sensors with cone angles of 128 degrees. These two sensors are positioned on the spacecraft with the

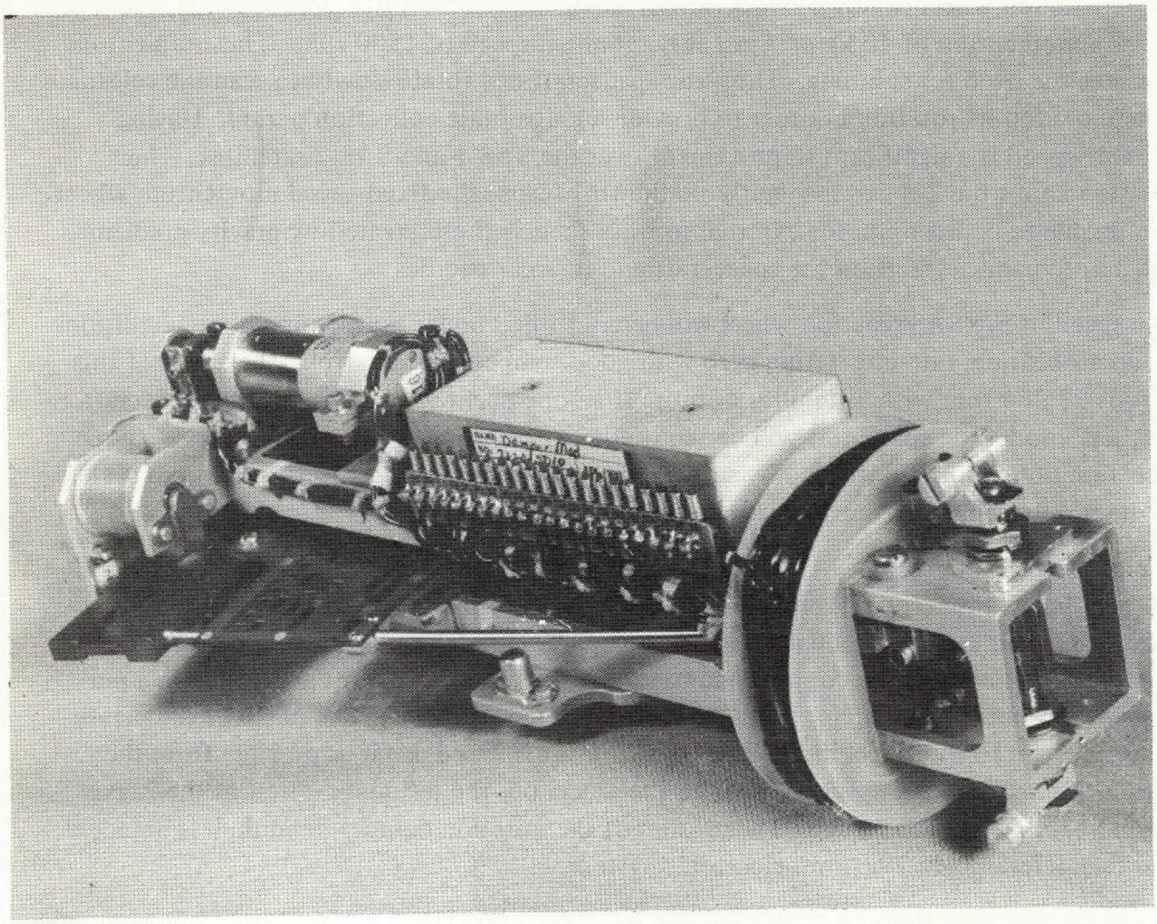


Figure 14. The SAS Nutation Damper

centers of their fields-of-view collocated and with their cone angles overlapping. Data from these sensors consist of two 8-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 by 2 degrees and a 0.5-degree resolution of the sun's elevation, which is provided in an 8-bit digital word. The time of sun crossing is stored and telemetered with an accuracy of ±1 second.

The star sensor planned for SAS-C and for use on later spacecraft is the same as that used on SAS-2; it is similar in design to the one collocated with the X-ray experiment on SAS-1. Designed to be sensitive to stars of 4th magnitude and brighter, it is mounted perpendicular to the spin axis of the spacecraft and sweeps out a 10-degree band on 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 photomulitplier 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 pulse provides information on elevation. The analog output from the photomultiplier tube is converted to digital form by the telemetry system's analog-to-digital converter. These values are stored with data bits that denote the time at which the viewed star began its transit of the slit. Ground data processing will use sun sensor and magnetometer data for a coarse determination of attitude; this procedure 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 field-of-view is within about 45 degrees of the sun.

## CONCLUSION

The Small Astronomy Satellite standardized functional support section offers the utmost in flexibility possible in a spacecraft small enough to be launched by the smallest of NASA's launch vehicles, the Scout. The ability to accommodate experiments of various sizes and weights is not new in spacecraft design; however, to provide a spacecraft that can support a variety of missions with such little change in fabrication is unique. The standardization of the SAS functional support section makes it feasible to manufacture several SAS spacecraft at a time, making them available quickly and at a minimum cost for a variety of experiment payloads.

Goddard Space Flight Center
National Aeronautics and Space Administration
Greenbelt, Maryland February 2, 1974
039-23-01-01-51

# APPENDIX

# GENERAL DESIGN CONSIDERATIONS FOR SMALL ASTRONOMY SATELLITE (SAS) EXPERIMENTS

## SCOPE

This general design considerations appendix describes special requirements imposed by the Small Astronomy Satellite (SAS) Project on flight hardware. It includes parts selection and screening plus environmental test requirements. As ground support equipment, the SAS Project has available a Xerox Data Systems Sigma 5 computer for use during the integrated testing of the experiment and the spacecraft functional support section. This equipment is also used at the launch site for prelaunch 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 protoflight model, and a flight unit of the experiment; and provide ground support equipment for pre-integration checkout of his experiment.

## 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 X-560-63-e, "Aerospace Data Systems Standards"
- GSFC X-325-67-70, "Magnetic Field Restraints for Spacecraft Systems and Subsystems"
- NASA TN D-7362, "A Compilation of Outgassing Data for Spacecraft Materials," September 1973
- NASA Handbook NHB 5300.4 (1C), "Inspection System Provisions for Aeronautical and Space System Materials, Parts, Components, and Services"
- NASA SP 208, "The Prevention of Electrical Breakdown in Spacecraft"
- MIL-STD-SS9, "Dissimilar Metals"

- 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-826, "Electromagnetic Interference Test Requirements and Test Method"
- GSFC-W3, "Welded Electronic Modules"
- NASA NHB 5300.4 (3A), "Requirements for Soldered Electrical Connections," May 1968, as applicable
- 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"

## PARTS AND MATERIALS

All parts, wire, cable, connectors, and material used in the SAS spacecraft shall be approved by the GSFC SAS Project Manager. Preference shall be given to those in the 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 functional support section and experiments. Specifically, for all semiconductors, that is, 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 semiconductor,
- Temperature cycling from 208 K (-65° C) to maximum rated storage temperature,
- Centrifuge test,
- Electrical test with variables data recorded,
- 336 ±36 hours burn-in at 373 K (100° C) (or 264 ±24 hours at 398 K (125° C)) and 80 percent 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, and
- 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 screening process specifications to which part is being bought, and
- 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, and
- Outgassing characteristics, if known.

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

## DESIGN CONSIDERATIONS

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

### Power System

In the design of the experiment power system and its experiment loads, consideration must be given to protection features for the spacecraft functional support 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 functional support section is 16.1 ±15% volts. It must tolerate ripple up to 250 mV at any frequency up to 20 kHz.

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 level 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 functional support section and the experiment section.

### Noise Protection

The experiment must protect itself against radiated pulses, for example, from the spacecraft clock. It must also assure that it is not radiating pulses that might interfere with the operation of the spacecraft functional support section, for example, 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 unnecessary transients and noise from his lines by means of suitable filters, shielding, and so forth.

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.

### Command System

Ten relay commands plus a main power on-off command are provided to the experiment by the spacecraft functional support 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.

### Telemetry System

From 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 Noise Protection.

### 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 a few arc-minutes. 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 functional support section.

### 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 (1) susceptibility to damage from ultraviolet radiation and (2) outgassing properties.

## ENVIRONMENTAL TEST REQUIREMENTS

### 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, the table provides a good estimate of qualification vibration levels which might be expected by an experiment having a resonant frequency greater than 150 Hz, which is bolted to the SAS functional support section.

### 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 ionosphere, 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 K (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."

### Humidity

Each part of the spacecraft must be designed to survive, operating, conditions of relative humidity 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

Table
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 (Hz) | APSD ($g^2$/Hz) |
|---|---|---|
| Thrust | 20-150 | +6 db/oct. up to 0.045 |
| | 150-500 | +6 db/oct. 0.045 to 0.12 |
| | 500-2000 | -3 db/oct. from 0.12 |
| Lateral | 20-300<br>300-2000 | +3 db/oct. up to 0.045 |

Duration: 2 minutes per axis

can be taken to protect those items in a high humidity environment, the waiver will be granted.

### Magnetic Testing

Permanent magnetic dipole moments should be less than 1000 pole-centimeters as a design goal. Variable magnetic dipole moments should be less than 5 pole-centimeters. 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; for example, nonmagnetic fasteners should be used. A test will be performed at the GSFC Magnetic Test Site to assure that these requirements are met.

### Shipping and Handling

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

## 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.

### Support During Integration, Testing and Launch

The integration contractor is responsible for integration of the experiment with the spacecraft functional support 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.

### 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.

## 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 functional support 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.

### 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.

### Safety Plan

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

### 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.

## 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

NASA-Langley, 1974