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THE SMALL ASTRONOMY SATELLITE (SAS) PROGRAM

by Marjorie R. Townsend

Goddard Space Flight Center
Greenbelt, Md.

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ABSTRACT

The Small Astronomy Satellite (SAS), one of NASA's newest Explorer-class programs, can provide at relatively low cost much previously unavailable basic information on low and high energy radiation from sources both inside and outside the galaxy. This report gives sufficient information concerning the SAS volume and weight, and the power, command, telemetry, and control systems to permit the reader to ascertain its suitability for specific astronomy experiments. Project reliability requirements and environmental test conditions are included in the appendixes.

FOREWORD

Our current decade has seen many remarkable advancements in astrophysics. Some of the most outstanding are the discovery of over two dozen x-ray sources and the detection of gamma radiation from the galactic center. These findings represent a major step toward the goal of understanding the high energy phenomena in our galaxy and those which govern the principal physical processes of the universe.

It is now time for detailed studies of the phenomena already discovered, for much more thorough surveys of the sky to determine the numbers and types of objects emitting high energy photons, and for intensive studies of celestial objects of particular interest. Supernovae, for example, appear to be the most catastrophic events within galaxies and are also thought to be the origin of cosmic rays. They deserve examination in the ultraviolet, X-ray, and gamma-ray spectral regions to provide a fuller understanding of such processes as nuclear synthesis and possible mechanisms of cosmic ray acceleration. Further, high energy photon studies should provide some important pieces in the puzzle of dynamic balance among the various forces in our galaxy, including those of gravity, the cosmic ray gas, magnetic fields, and the kinetic motion of interstellar matter. The resolution of this problem, together with the determination of the composition of interstellar matter, is also fundamentally important in the study of star formation. Probing further into the universe for objects and phenomena still unknown to us is yet another very important facet of high energy astronomy; it should be remembered that none of the celestial phenomena known before the 1960's led astronomers even to predict the existence of celestial x-ray sources.

The principal aim of the Small Astronomy Satellite (SAS) program is to provide an Explorer-class satellite capable of supporting experiments in high energy astronomy, and thereby providing one means of accomplishing the goals just outlined. Designed for launching by the Scout vehicle, SAS is intended particularly to fill the gap between very small or short-duration experiments and the very large ones to be flown on observatories and large spacecraft. This document describes the SAS satellite and its general capabilities so that the potential user may determine whether it is compatible with his needs.

Carl Fichtel
Project Scientist

THE SMALL ASTRONOMY SATELLITE (SAS) PROGRAM

by

Marjorie R. Townsend

SAS Project Manager

Goddard Space Flight Center

INTRODUCTION

The objective of the 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, visible, and infrared regions of the spectrum both inside and outside of our galaxy. Surveys performed from such a small, Scout-launched satellite (Figure 1) will provide valuable data on the position, strength, and

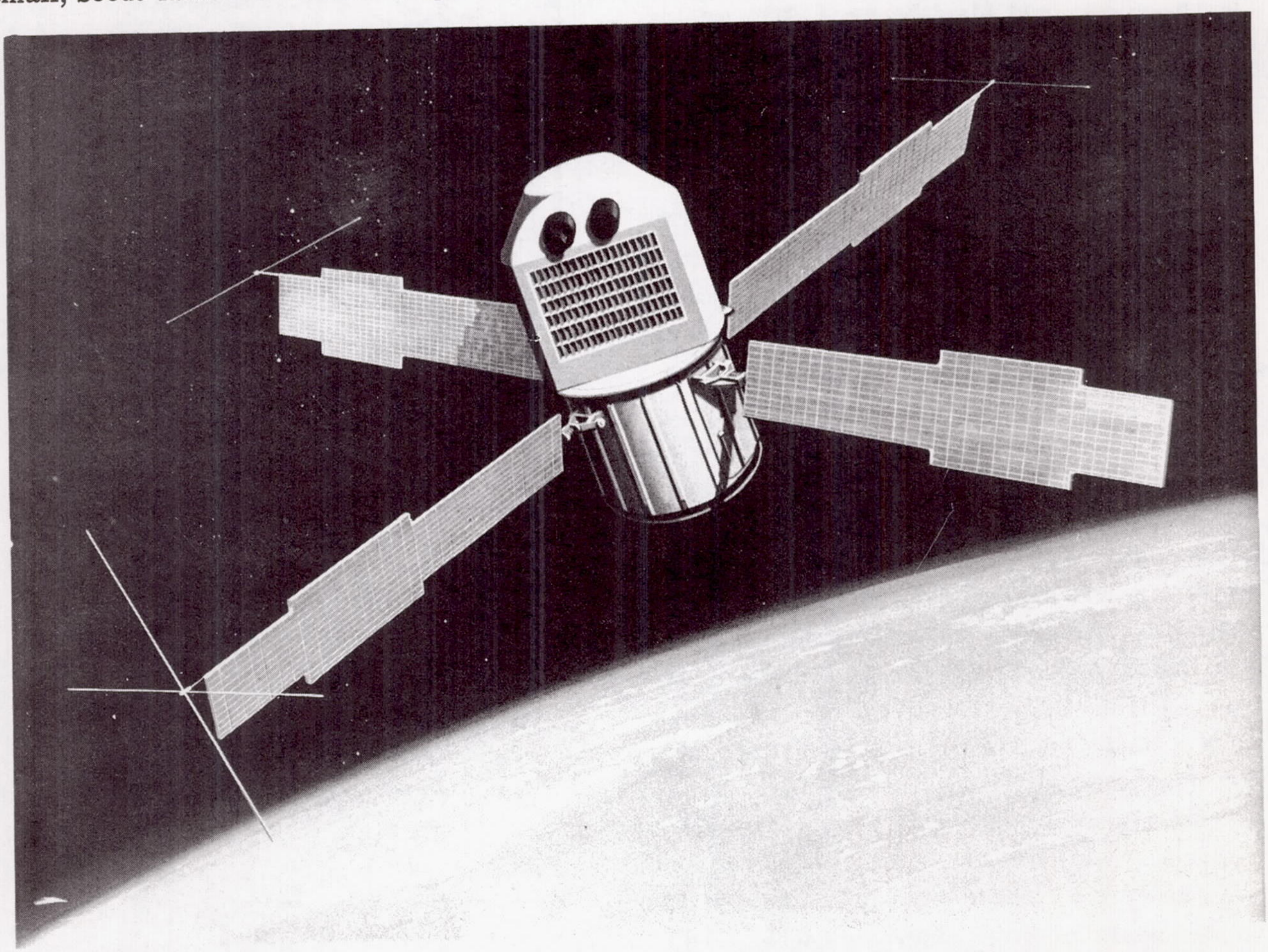

Figure 1–Artist's conception of the Small Astronomy Satellite (SAS): experiment, above; spacecraft, below.

time variation of sources radiating in these regions of the spectrum and lead to the selection of the more interesting ones for detailed study by later missions on this or more sophisticated spacecraft. It is anticipated that, beginning in 1970, one or two of these satellites will be launched each year (with an estimated lifetime of a year, or as limited by the experiment sensors).

A nominally circular 300 nautical mile (555 kilometer) equatorial orbit is planned for the SAS series. The equatorial orbit is highly desirable for an astronomy mission, because the satellite will miss the South Atlantic anomaly where the radiation belts come far down into the earth's atmosphere (Figure 2). This will avoid degradation of the spacecraft operation and minimize the background count which can affect the data from several different types of sensors applicable to satellite astronomy. This orbit, with an inclination of 2.9 degrees, can be achieved by launching from the San Marco platform located three miles east of Kenya in the Indian Ocean. The maximum payload capability under these conditions is 333 pounds.

The prime contractor for SAS is the Applied Physics Laboratory of the Johns Hopkins University, which is responsible for developing the spacecraft and for integrating and testing each spacecraft and experiment.

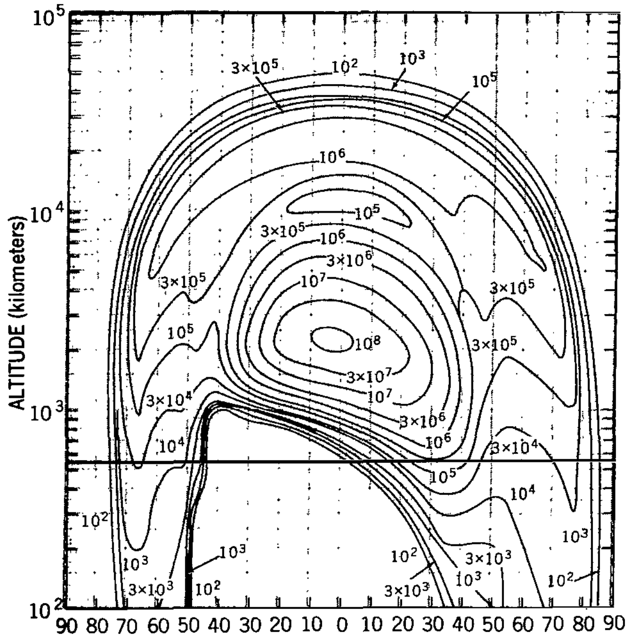

Figure 2–Longitudinally averaged map of the earth's radiation belts. The isolines refer to omnidirectional fluxes (cm^{-2}-sec^{-1}) of electrons with energies > 0.5 MeV.

BASIC SPACECRAFT DESIGN

The satellite is designed to have a clean interface between the basic spacecraft structure and the experiment payload (Figure 3). This is intended to minimize costs for follow-on programs. The basic spacecraft is a cylinder approximately 22 inches in diameter and 20 inches high, weighing about 180 pounds. The lightweight cylindrical shell is the primary support for the experiment. Four solar paddles, hinged to the outer shell, provide raw power to the spacecraft and experiment. At the tips of the paddles are the command and telemetry antennas. Inside the shell (Figure 4), and thermally isolated from it, is a honeycomb deck which is itself a good thermal conductor. On it are the systems to provide the basic spacecraft functions: a rechargeable battery with its charge control and regulator systems; command receivers and decoders; a 1000 bit per second telemetry system with one-orbit storage capability on tape; and a magnetically torqued commandable control system which can point the spacecraft stably to any point in the sky. The electronic circuitry is packaged in "books" mounted, with the nutation damper, on the underside of the honeycomb deck. The battery, tape recorder, transmitter, command receivers, and rotor are mounted on the top side.

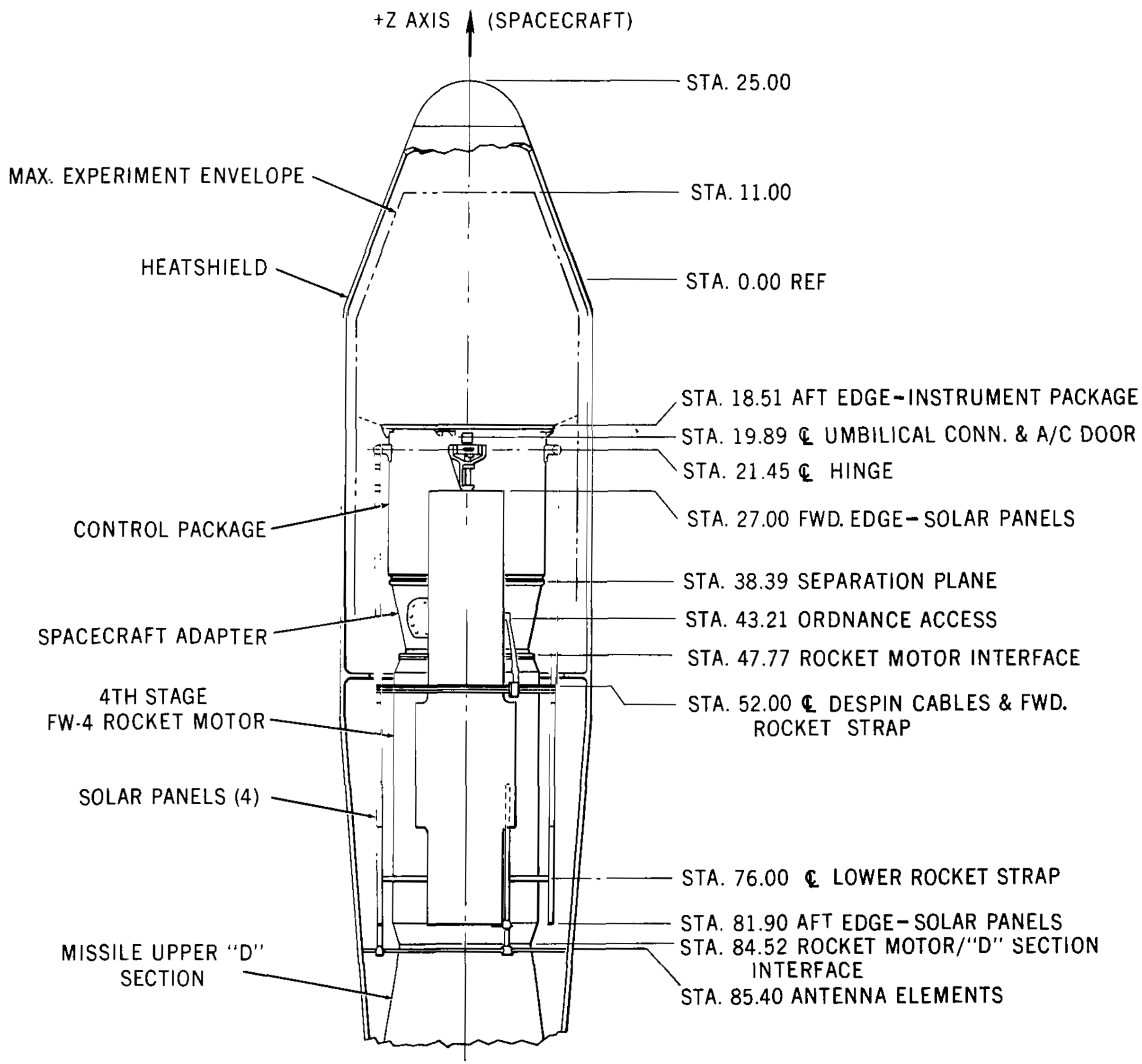

Figure 3—SAS in launch configuration on Scout with standard shroud.

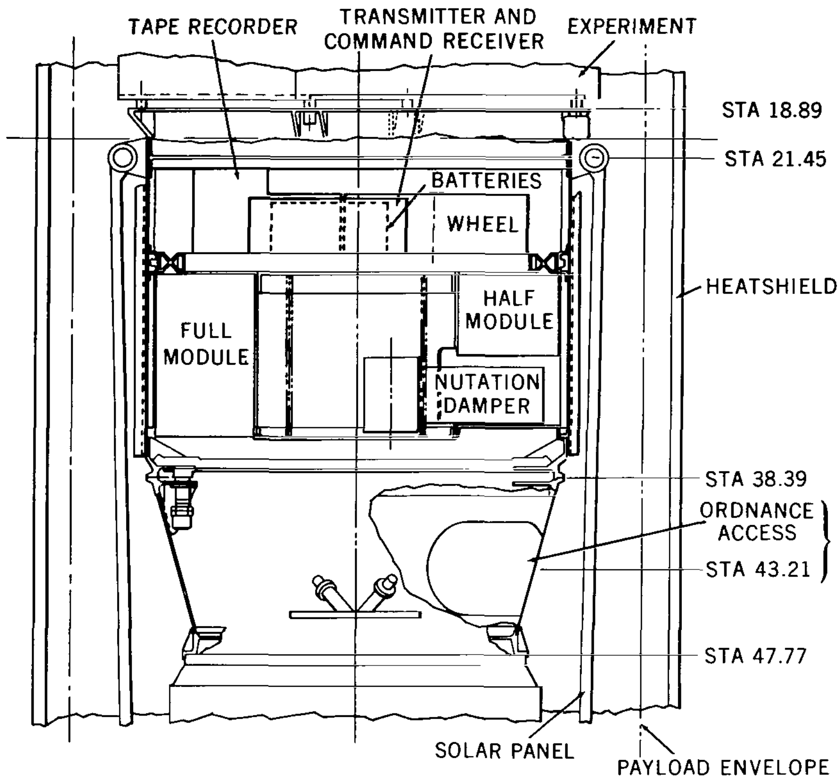

Figure 4—Cross-section of basic spacecraft structure.

One of the SAS design requirements was the ability to operate in any orientation, thus complicating the thermal design as well as that of the power system. The surface of the cylinder is the only part of the spacecraft portion which radiates the internal heatloads to space. Since the experiment is isolated from the spacecraft section with a high thermal resistance, we will consider the spacecraft section independently of the experiment section. The most critical items are the tape recorder and the battery, which are co-located for efficient thermal design. The judicious use of passive thermal coatings and multilayer insulation, with a small amount of active thermal control in the form of heaters, permits operation in any random orientation. While most of the spacecraft components can be operated satisfactorily from -20° C to +70° C, the battery prefers a temperature between 0° and +30° C. A separate array of solar cells on the base of the spacecraft provides heater power for the experiment in the worst-case cold condition.

POWER SYSTEM

The power system consists of: four paddles with solar cells on both sides; a 6-ampere-hour 8-cell nickel-cadmium battery; and a shunt regulator. Still assuming a random orientation, the power system must provide 27 watts average power over the entire orbit, both night and day. This is particularly important in astronomy experiments, which frequently get their most useful data at night, when scattered sunlight cannot interfere with the sensors. The basic spacecraft functions require 17 watts average power, allowing 10 watts for the experiment. Figure 5 shows the variation in available power as a function of sun angle. The total power includes operational needs plus enough to charge the battery to provide the same power through the dark part of the orbit.

A block diagram of the power subsystem is shown in Figure 6. The solar cell array is connected in two separate sections, the main array and an auxiliary array. The auxiliary array provides redundant power for the command subsystem, and trickle-charges the battery if it has been disconnected from the main bus by the low voltage sensing switch.

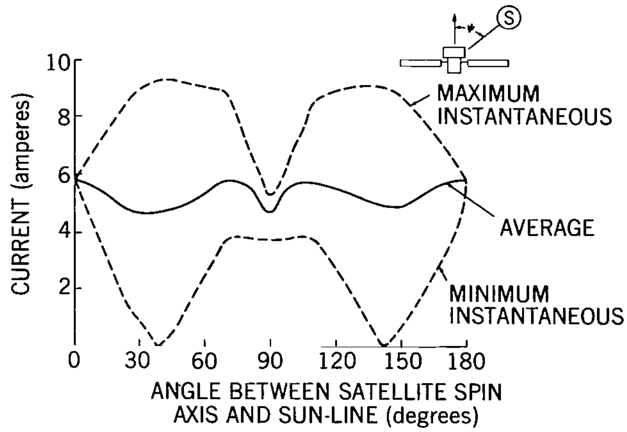

Figure 5—SAS solar cell array current.

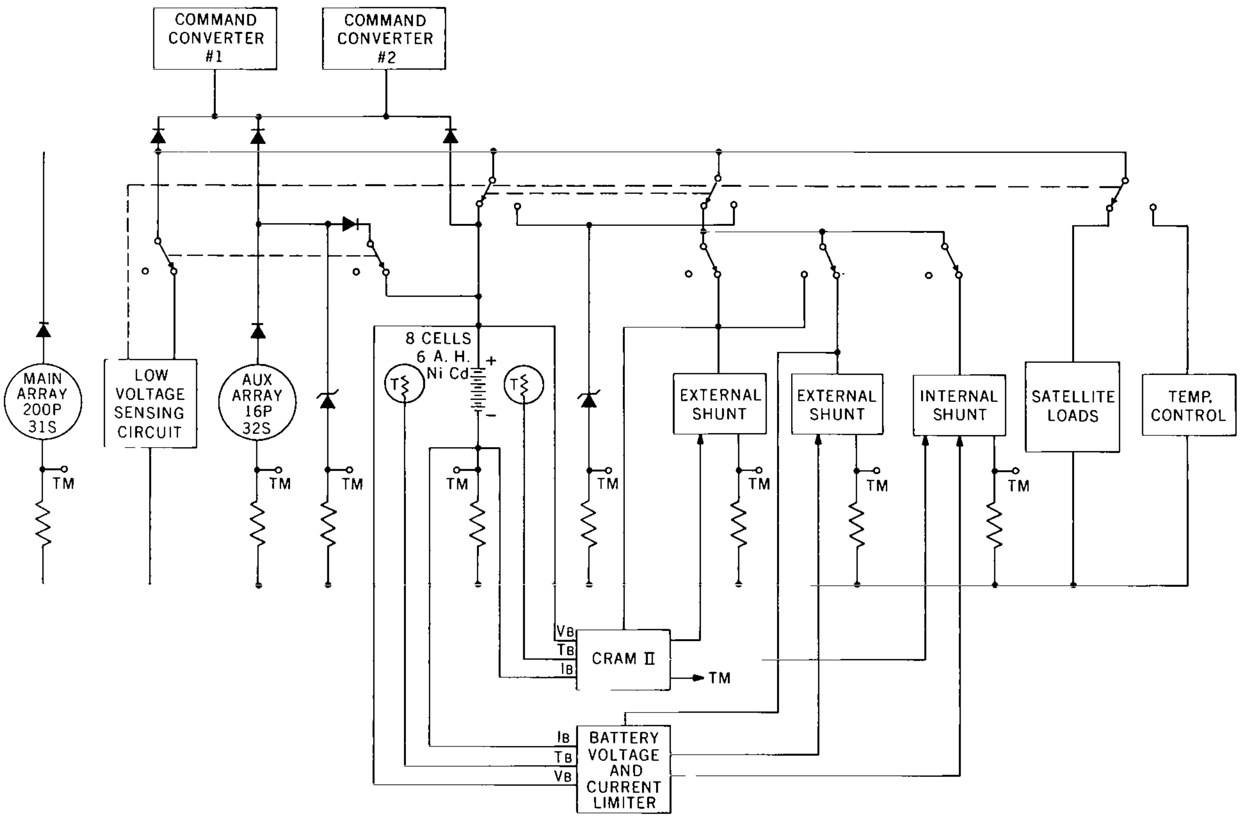

Figure 6—SAS power system.

The 6-ampere-hour nickel-cadmium battery is protected by two redundant charge-control devices, a coulometer and an enhanced zener.

The coulometer senses the total energy taken from and returned to the battery, by integrating the voltage developed across the battery telemetry resistor. When the energy taken from the battery on discharge has been returned, the coulometer reduces the battery charge current to a "safe", preset trickle-charge level by shunting the remainder. (This unwanted current is shunted internally if the battery is cold, or externally if it is hot.)

The enhanced zener is a voltage limiter. It actuates a shunt, decreasing the battery charge current, when the battery voltage reaches a preset level. Thus the charge current decreases to a level which makes the battery voltage equal to the enhanced zener limit voltage. As with the coulometer the shunted current may be dissipated internally or externally, depending on battery temperature. The power subsystem normally operates with both the coulometer and the enhanced zener. However, it can operate with either one alone.

The low voltage sensing switch continuously monitors the voltage of the main bus. If it drops below 8.8 volts (normal voltage is 10.7 volts), power is switched from the main converters and the battery to prevent damage due to possible shorted loads. In this "solar only" mode, the battery will be trickle-charged by the auxiliary array, while the only loads on the main array are the dual command systems and the heater circuits. Each circuit can be switched out by command if it becomes defective.

COMMAND SYSTEM

SAS employs the NASA standard PCM command system and carries redundant receivers and decoders. The redundancy continues through the relay coils themselves, but a single set of contacts is used. Figure 7 shows the command system. Independent dipole antennas are mounted on the tips of two adjacent solar paddles. The command signal, a pulse code modulated, frequency-shift keyed, amplitude modulated (PCM/FSK-AM/AM) signal, is demodulated by either or both command receivers and appears at the video output. Bit synchronization is obtained from a signal that has been amplitude-modulated onto the data subcarrier. The video signal is sent to a command decoder, which tests the tones for authenticity by examining the timing and frequencies. If the tones are proper, the decoder converts them to digital levels and sends the levels and appropriately timed clocking pulses to the logic circuitry. The decoder also applies power to the standby portions of the logic, beginning with the first correctly received tone burst and remaining on long enough to complete the command. The logic circuitry accumulates the decoded levels in a shift register and, if the satellite address bits are proper, it activates the proper row and column of the relay switching matrix to switch the desired relay or relay cluster.

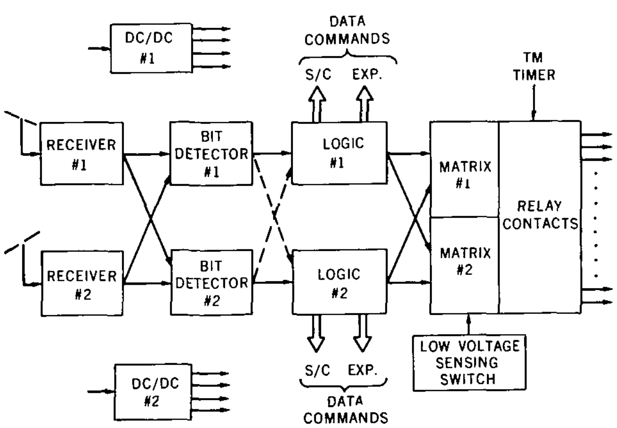

Figure 7–SAS command system.

From the redundant interconnections it is seen that only one receiver, one command logic, and one relay switching matrix need be operating to send commands. One bit in the command word (Figure 8) addresses the particular logic circuit and one bit the particular relay switching circuit to be used, so that only one logic and one matrix are activated during any given command. Each half-system has its own power converter. The system's magnetic latching relays each have two coils, one for either system half, and consume power only during a command. The direction of the current in any given coil determines the state in which the relay will latch. Because of these dual coils, actually located in the relay switching matrix, the system's redundancy continues up to the magnetic flux circuit in the relay pole piece, and no single relay coil open or short can disable the system.

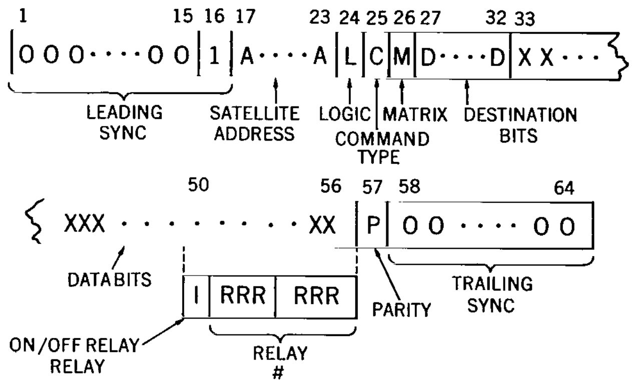

Figure 8—SAS command word.

The command system provides SAS with 35 on-off relay commands: 25 for the spacecraft and 10 for the experiment. In addition, if bit 25 in the command word specifies a "data command" instead of a "relay command", then bits 33 through 56 are routed, as specified by bits 27 through 32, to a location other than the relay matrix for separate decoding. This permits an experimenter to increase his allocation of 10 on-off commands by decoding up to 2^{24} within the experiment.

TELEMETRY SYSTEM

Because of the low equatorial orbit chosen for SAS, the spacecraft can be seen regularly by only one of the NASA Satellite Tracking and Data Acquisition Network (STADAN) stations. For this reason, the spacecraft telemetry subsystem (Figure 9) must include on-board storage of the data collected during the remaining 90 percent of the orbit. Therefore, the pulse code modulated/ phase modulated (PCM/PM) telemetry system has two basic modes of operation—record and playback. In the normal record mode, digital data from various sources are multiplexed and sent to be stored serially in Manchester Code on an endless loop tape recorder and to phase-modulate a VHF transmitter. This continuous output, radiated through a turnstile antenna at the tip of a solar paddle, is used as a tracking beacon and to collect real-time data. As the satellite passes over its data acquisition station, however, the tape recorder is commanded into playback mode and the data from the full 96-minute orbit phase-modulate the same transmitter with a 30 kbps signal. Simultaneously, the transmitter is switched into its high power (2-watt) mode. The recorder and transmitter return to the record mode automatically after completion of the 3.4 minute playback cycle, or upon ground command. In a special operating mode, real-time data can be transmitted at 2 watts. This system meets the requirements of the NASA/GSFC Aerospace Data Systems Standards for PCM telemetry systems. The maximum phase deviation is ±64°, with a real-time bandwidth allocation of ±15 kHz and a tape recorder playback allocation of ±45 kHz. Linear phase filters, having their 3-db points at 1 kHz and 30 kHz respectively for the record and playback modes, are used for premodulation filtering.

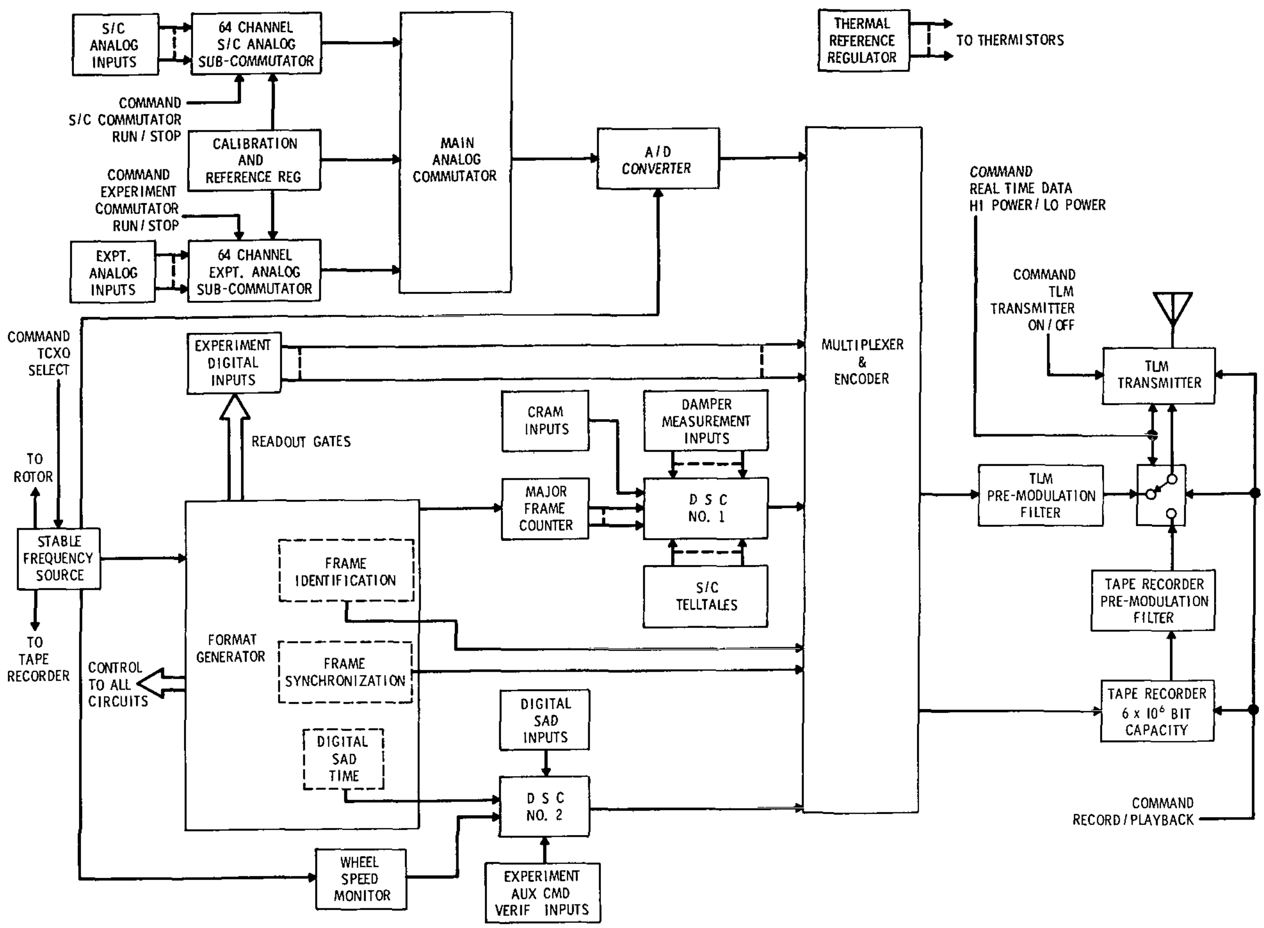

Figure 9—SAS telemetry system.

The real-time data rate of 1000 bps was determined by optimizing experimenter and housekeeping data requirements with a reasonable length of tape (300 feet) and a reasonably conservative bit packing density for single track recording (1667 bits per inch). The tape recorder, of flight-proven endless loop design, weighs only 7 pounds and requires 1.5 watts in record mode and 3.0 watts in playback mode. Wow and flutter will be kept below 2 percent peak-to-peak, with jitter below 3 percent peak-to-peak. Commands will bypass the output reclocking circuitry and energize the erase head, as well as provide the obvious record/playback and on/off control. Digital data recording was selected for its better signal-to-noise properties and for the ease of providing digital data to the telemetry system.

The basic format is shown in Figure 10. The minor frame is composed of 32 words, each with three 8-bit syllables, an arrangement that permits easy handling of 8 bit, 16 bit, or 24 bit data words. Figure 11 shows the breakdown of a minor frame. Of these 96 syllables, the first three are allocated to frame synchronization. The frame synchronization word used is the optimum 24-bit code word determined by Maury and Styles[1], 111110101111001100100000. One syllable

[1]Jesse L. Maury, Jr., and Frederick J. Styles, "Development of Optimum Frame Synchronization for Goddard Space Flight Center PCM Telemetry Standards", *Proceedings National Telemetry Conference*, June 1964, Los Angeles, California.

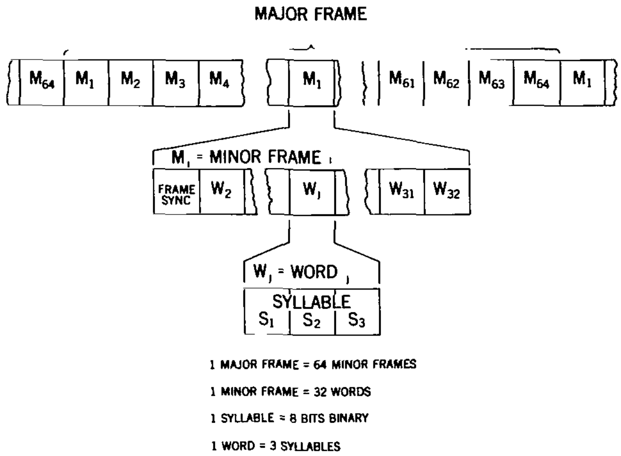

Figure 10—SAS PCM telemetry system frame format.

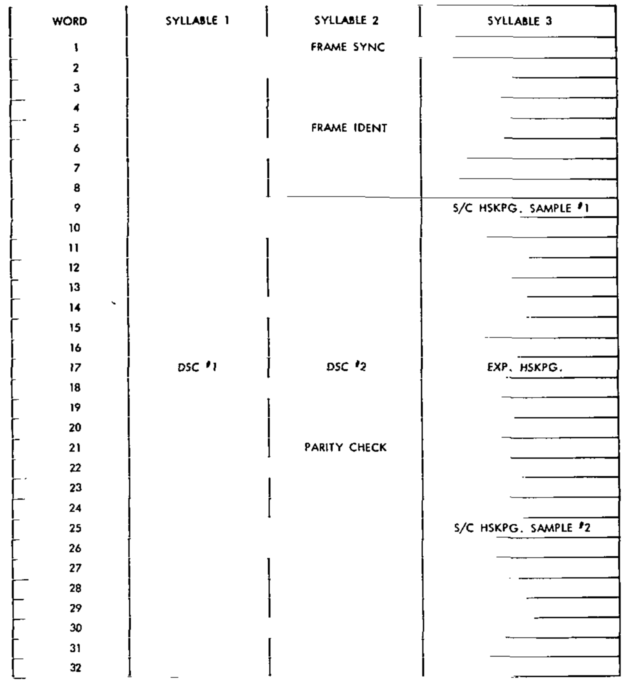

Figure 11—Telemetry minor frame format.

is allocated to frame identification, three to 64-position analog subcommutators, two to digital subcommutators, one for a parity word, and the remaining 86 for experiment data. The period of the minor frame is 0.768 second, and of the major frame is 64 times that, or 49.152 seconds. The basic timing source for the system is a temperature-compensated 1.024 MHz crystal oscillator whose frequency variation is less than one part in 10^6 per orbit. It serves as the source for all readout gates, clock signals to control the multiplexing of spacecraft and experiment data, motor drive frequencies for the tape recorder and rotor, and pulses for the 20-bit accumulator, which is incremented by the minor frame identifier every major frame, and sampled every eight minor frames. Unambiguous time correlation for about 20 months is provided.

While most of the data originate in digital form, many of the housekeeping functions are analog and must be converted. A dual-slope integrating analog-to-digital converter, accurate to 8 bits, is used for this purpose. It is clocked at 16 kHz and has an aperture of 8 msec. While its normal inputs are 0 to 250 mv, the telemetry system can provide a variety of attenuators so that signal levels of ±0.5, ±1.25, ±2.5, ±5.0, or ±7.5 volts can be accepted. An additional feature of the analog subcommutators is that it is possible to command either of them to stop and sample continuously—at the minor frame rate of 0.768 second—any one of its 64 positions. After receiving the RUN command again, it will automatically resynchronize itself at the beginning of the next major frame.

One of the digital subcommutators has 16 channels. Twelve of these accept 8-bit parallel data, while the other four accept 8-bit serial data. The second digital subcommutator has 8 channels, each of which accepts 8-bit serial data. These digital subcommutators provide command verification, the unambiguous time code, a monitor of the rotor speed, and time correlation to 24 msec for the digital solar aspect sensor data.

This telemetry system occupies less than 0.2 cubic foot, weighs 14 pounds, and consumes 4.5 watts. It is built in modular form for easy expansion, and the design is such that the present hard-wired format could be replaced by an in-flight programmable unit.

CONTROL SYSTEM

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

The basic control system uses the earth's magnetic field for torquing the satellite. An obvious advantage of this is the elimination of the need for expendables and their inherent life limitations.

Experiments can look along the Z-axis of the SAS spacecraft or perpendicular to it. If an experiment looks out perpendicular to the Z-axis, and the spacecraft is rotated slowly about the Z-axis, then a swath is swept out of the celestial sphere. The vector made by the Z-axis in space can be moved, and another swath swept. Thus, ultimately, the entire celestial sphere can be scanned.

This sweeping technique is valid even if the Z-axis wobbles, but data reduction is much easier if the Z-axis is stable. One way of achieving stability is by spinning the spacecraft. This is quite satisfactory for many experiments looking along the Z-axis. However, a high spin rate will reduce the effective time-on-target for perpendicular-looking experiments and, consequently, the signal-to-noise ratio of the celestial sources. An alternative method of adding the required angular momentum along the Z-axis is to include a momentum wheel whose axis is parallel to the Z-axis.

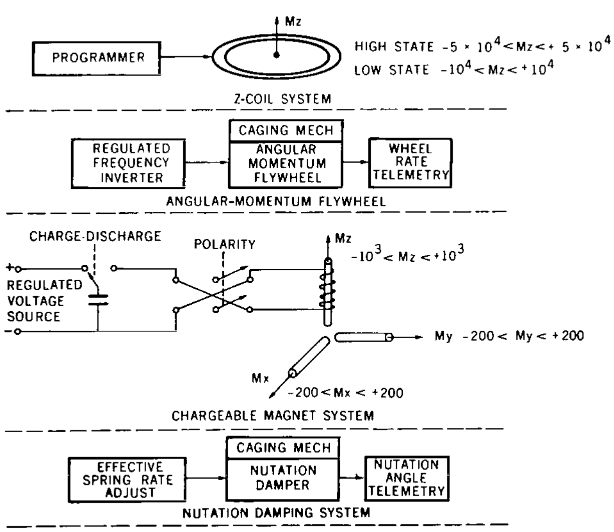

Figure 12—SAS-A spin-axis orientation control system.

Figure 12 shows the various components of the control system. Z-axis orientation is controlled 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 5×10^4 pole-centimeters, the Z-axis of the spacecraft can be moved 1.72 degrees per minute.

Figure 13 shows how a maneuver might be performed to point the Z-axis to a new direction as requested by an experimenter. This particular sequence is designed to result in no change in right ascension, but only in declination. In normal operation, a combined motion in both right ascension and declination will be

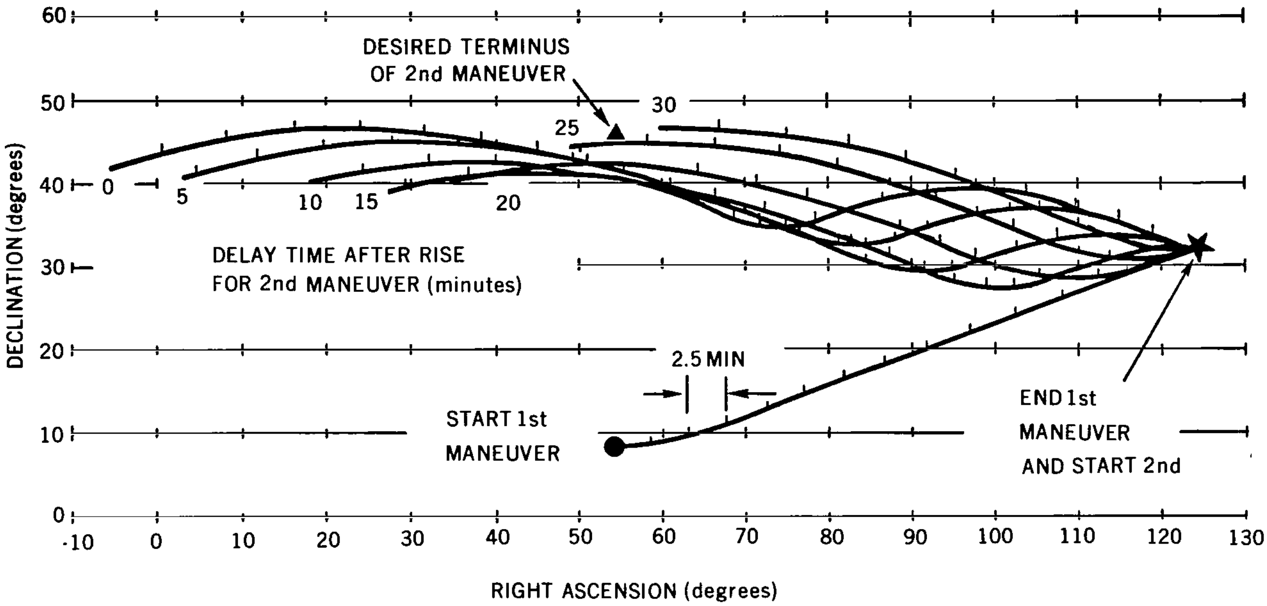

Figure 13–SAS-A declination maneuver sequence.

accomplished with the aid of a computer. The computer must store the values of the earth's magnetic field in each predicted satellite position. With this set of values, the dipole moment of the satellite, and the desired change of position, the computer will determine at what point in the orbit the torquing coil is to be turned on and for how long. Time delays of up to an hour are available in the satellite, so the maneuver can begin at the optimum time in the orbit, not just over the command station. Time increments for this delayed maneuver are 2.5 minutes, both for the delay and for the maneuvering period. Finer control can be obtained by turning the torquing command on and off while the satellite is within sight of the station. Once the position is achieved, it is held primarily by the momentum of the rotor (Figure 14), which is 2.12×10^7 gm-cm^2/sec. Drifting from this position is caused by gravity gradient, aerodynamic, and magnetic torques, the latter resulting from uncompensated dipoles. The total of these drift rates should not exceed 5 degrees per day on the average. To minimize its residual magnetism, the spacecraft is provided with a degaussing coil and a chargeable trim magnet system to compensate in orbit for residual magnetism along each axis. It is possible to use this system to produce a specific dipole moment for controlled spin-axis drift.

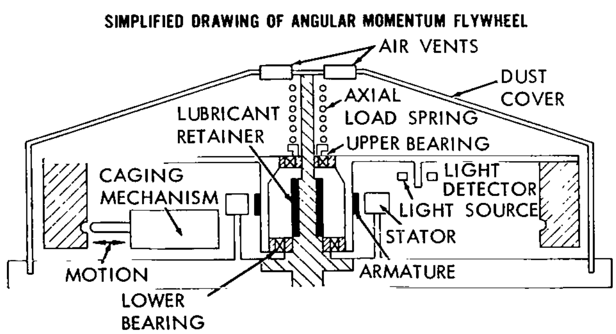

Figure 14–Simplified drawing of angular-momentum fly wheel.

The nutation damper is used to dissipate the lateral components of satellite angular velocity created by the magnetic torquing. It consists (Figure 15) of a torsion-wire suspended

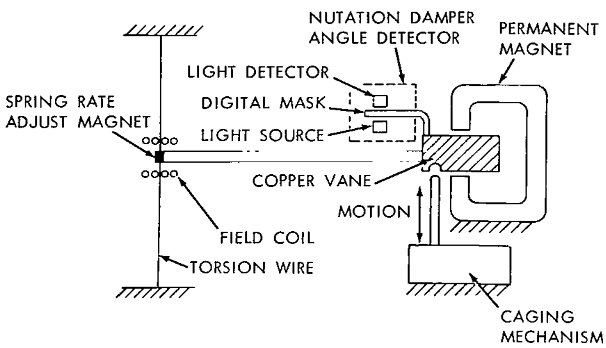

Figure 15—Simplified drawing of SAS-A nutation damper.

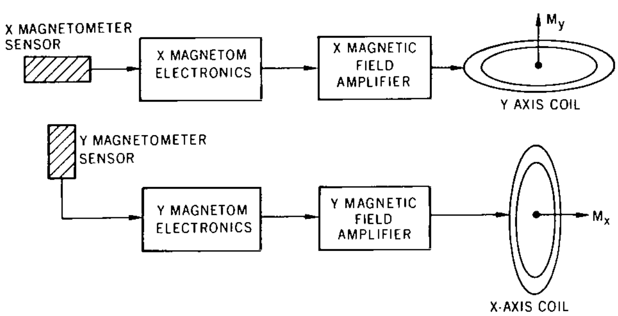

Figure 16—Block diagram of spin rate control system.

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. With the rotor on, the residual nutation should be less than 0.2 degree; with the rotor off, it may be a few degrees. The motion of the nutation damper, detected optically by a coded digital mask mounted on the movable arm is telemetered every 8 seconds via a 7 bit binary word.

The spin rate control system is shown in Figure 16. Signals sensed by the X- and Y-axis magnetometers are amplified and applied in quadrature to the Y- and X-axis coils. 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. The maximum rate of change of spin is 10 rpm per day.

Many astronomy experiments can be performed well with this basic SAS control system. Used in conjunction with a good attitude-determination system, it is ideally suited for sky surveys. Data telemetered from the magnetometers and a digital sun sensor on board the spacecraft can be used to determine attitude to ±5° or better.

PLANNED EXPERIMENTS AND FUTURE DEVELOPMENT

On SAS-A, the principal investigator, Dr. Riccardo Giacconi of American Science and Engineering, Inc., is flying star and sun sensors accurately aligned with the experiment X-ray collimators so that attitude may be determined to a minute of arc. This experiment will provide an accurate map of X-ray sources throughout the celestial sphere.

After such a survey, which will determine source strength, position, and spectral composition, later X-ray experiments will require more accurate pointing for detailed study of individual sources. The SAS design permits building on this basic control system, by adding a star tracker or star camera system, to provide three-axis stabilization and thus maintain the satellite attitude within one minute of arc or better. This would permit useful experiments in the ultraviolet and infrared regions as well.

Rough positioning of the spacecraft would be accomplished by the basic control system, and the exact attitude would be determined by ground computer. Then, for example, a program could be transmitted for storage in the spacecraft defining a pattern that the star camera should see, and fine torquing would be used to find and maintain the proper position. Two sets of optics with about 10° fields of view perpendicular to each other could permit three-axis attitude detection to a resolution of 10 seconds of arc, and control to 1 minute of arc. Error signals would control the speed of the basic rotor, whose excess momentum would be dumped into the earth's magnetic field (again avoiding expendables). Small gyros or a worm-gear tilting mechanism would be suitable for fine control of the X- and Y-axes. A system of this general type is planned for SAS-C and later flights.

SAS-B will perform a sky survey of gamma-ray sources, with Dr. Carl Fichtel of Goddard Space Flight Center as principal investigator. The basic control system is adequate for this purpose, as the Gamma Ray Spark Chamber has a 45° field of view and does not require very fine pointing.

Goddard Space Flight Center
National Aeronautics and Space Administration
Greenbelt, Maryland, October 4, 1968
878-11-75-01-51

Appendix A

Summary of SAS Characteristics

Launch Vehicle

Scout with FW-4 fourth stage

Orbit

Altitude: 300 n. mi. (555 km) ±40 n. mi. (74 km), or 1σ

Inclination: 2.9° ±0.15° (1σ)

Period: 95.8 min.

Eccentricity: 0.008

Size

Spacecraft: Shape - cylinder
Diameter - 22"
Height - approx. 20"
Volume - approx. 4.4 cu. ft.

Experiment: Shape - cylinder topped by truncated cone (Figure A1)
Base diameter - 30"
Height of cylindrical portion - approx. 14" or 29"
Height of truncated cone - 15.2"
Top diameter - 17.7"
Volume - approx. 6.4 cu. ft. or 12.5 cu. ft. (for the "pointed SAS" about 1 cu. ft. must be made available from the experiment volume)

Weight

Spacecraft: 180 pounds

Experiment: 150 pounds - must have low c.g. (see Figure A2)
(130 pounds on "pointed SAS")

Continuous Average Power

Spacecraft: 17 watts

Experiment: 10 watts (9 watts on "pointed SAS")

Voltage: +10.7 vdc +10% -15%

Ground: Three-wire system—separate chassis, signal, and power grounds

Telemetry

Type: PCM split phase

Real Time and Record Rate: 1000 bps for 96 min. (includes experiment and housekeeping data)
Note: 16,000 bits at 500 kbps or less could be accepted instantaneously into a buffer for readout onto tape at 1000 bps

Storage Method: Endless loop tape recorder

Capacity: 5.4×10^6 bits

Playback Rate: 30 kbps for 3.4 minutes

Transmitter Frequency: 136 MHz (90 kHz bandwidth)

Encoder: 8-bit analog-to-digital converter will provide better than 1.0% accuracy for signals up to 1 cps.

Input Signal Levels: Analog: ±7.5 v, ±5.0v, ±2.5v, ±1.25v, or ±0.5v, with source impedance of 5 k ohms
Digital: +4v for a "1" and +0.3v for a "0" into a standard LPDT micrologic

Tracking

Positional Accuracy from Minitrack System: 10 km

Commands

NASA STADAN PCM System

Spacecraft: 50 commands (25 "on" and 25 "off" commands)

Experiment: 20 commands (10 "on" and 10 "off" commands—can be extended up to 2^{24} by using logic internal to the experiment)

Control System

Basic System: Positioning of Z-Axis - <1° relative to known position
Average Drift of Z-Axis - 0.2 arc-min per minute (5 degrees/day)
Spin Rate - 0-60 rpm
Attitude Measurement - ±3° (if greater accuracy is needed, additional sensors may be provided by the spacecraft or as part of the experiment)

Pointed SAS: Three-axis stabilization to 1 arc-minute

Environment

Temperature Range: -20° to +50°C

Humidity: Must survive conditions of 100% humidity.

Vibration: Table A1 gives the qualification-level vibration specifications at the spacecraft-experiment interface. These are derived from data from the Scout vehicle and are intended to meet the requirements of GSFC Document S-320-S-1, "General Environmental Test Specification for Spacecraft and Components Using Launch Environments Dictated by Scout FW-4 and Scout X-258 Launch Vehicles" dated May 20, 1966.

Magnetism: Residual magnetic moment of experiment should be kept below 500 pole-cm.

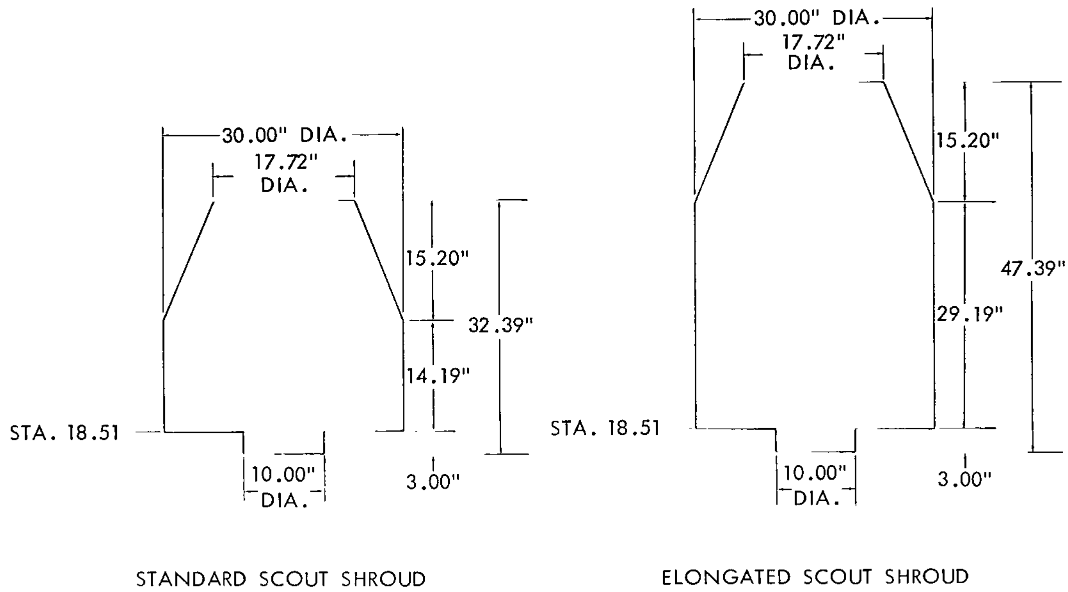

Figure A1—Outline drawing of SAS experiment package.

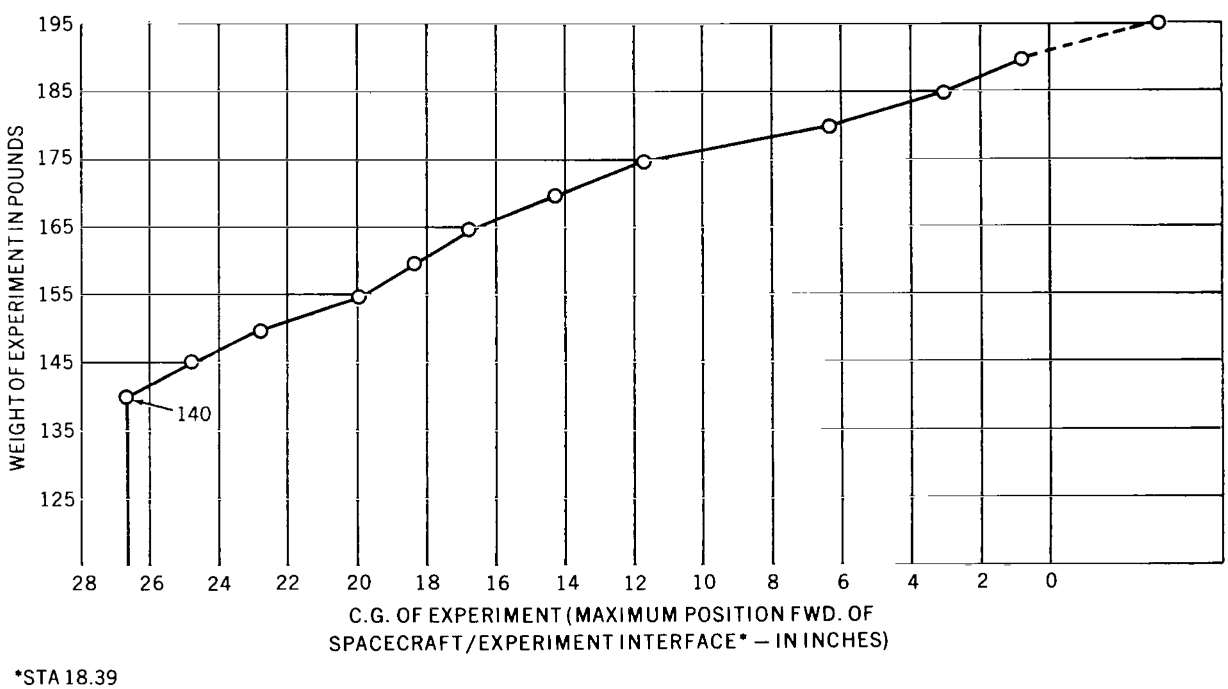

Figure A2—SAS experiment weight vs c.g.

Table A1

Qualification-Level Vibration Specifications for SAS Experiments (based on Scout launch vehicle data).

Random - 4 Minutes Each Axis		
Direction	Frequency (Hz)	Power Spectral Density (g^2/Hz)
Thrust Axis	20-50 50-100 100-150 150-370 370-2000	0.1 Increases at 10 db/octave 1.0 Decreases at 10 db/octave 0.07
Lateral Axes	20-40 40-150 150-370 370-2000	Increases at 10 db/octave 1.0 Decreases at 10 db/octave 0.07
Sinusoidal - 2 Octaves per Minute		
Direction	Frequency (Hz)	Acceleration
Thrust Axis	20-95 95-175 175-250 250-400 400-2000	0.187 inches double amplitude 40 30 15 7.5
Lateral Axes	15-36 36-200 200-400 400-2000	0.13 inch double amplitude 8 5 7.5

Appendix B

Special SAS Requirements

In designing experiments to be flown on the Small Astronomy Satellite (SAS), in general the requirements of NPC 200-2, "Quality Program Provisions for Space System Contractors," dated April 1962, must be met. In addition, parts and materials must conform to the current GSFC Preferred Parts List. While specific details must be worked out for each experiment, this list is provided for guidance.

Parts Screening

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

a. Visual inspection before sealing
b. Temperature cycling -65° C to maximum rated storage temperature
c. Centrifuge test
d. Electrical test with variables data recorded
e. 336 (±36) hours burn-in at 100°C and 80% of part rated power
f. Electrical test with variables data recorded (parts will be rejected if outside acceptable variables limits)
g. Fine and gross leak tests
h. Final inspection

All parts must be approved by GSFC before they can be used. Lists must contain the following information as a minimum:

a. Type of component
b. Value and rating
c. Manufacturer
d. Manufacturer's type and model
e. Manufacturer's screening process specifications to which part is being bought
f. Maximum anticipated electrical stress level

Reviews

A design review, a prototype readiness review, and a flight unit readiness review will be held on each experiment and the spacecraft. Malfunction reporting and configuration control will be implemented after the design review and prior to the beginning of prototype fabrication.

Reliability Assessment

A reliability assessment of the experiment will be performed by the experiment contractor.

Noise and Transient Protection

If the experiment requires the use of any high voltages, the experimenter must provide protection for the experiment and the spacecraft systems against high voltage surges or radiated pulses. The experiment must be designed to be insensitive to noise, e.g., that generated by clock pulses or dc/dc converter transients, and the experimenter is responsible for eliminating unnecessary transients and noise from his lines by means of suitable filters, shielding, etc.

If opening the 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 protective circuits (zener diode). Isolation shall be provided from the subsystem monitoring circuits in case of telemetry shorting.

Current transients in the main power line to the experiment must not exceed the following values:

Current	Time
4 a	0-5μsec
1.75 a	5μsec - 0.1 sec
(Normal) 1.0 a	After 0.1 sec

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— NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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