

THE SMALL ASTRONOMY SATELLITE PROGRAM – AN OVERVIEW

H. B. Riblet

Introduction

THE SMALL ASTRONOMY SATELLITE (SAS) program was started in the latter part of 1966. Discussions with NASA at that time revealed the desirability of designing a spacecraft that could be launched on small rockets such as the Scout and provide a basic platform from which a series of astronomy experiments could be conducted. NASA was already considering a proposal for an X-ray detecting experiment. This experiment was being designed by the American Science and Engineering (AS&E) Company of Cambridge, Massachusetts. The principal investigator for this experiment was Dr. Riccardo Giacconi. NASA needed a spacecraft to provide the attitude control, power, and data systems for this astronomy experiment and came to the Applied Physics Laboratory for its design.

A preliminary design study was conducted until the end of March 1967. During this design phase, careful attention was given to the mechanical and electrical interfaces between the experiment instrumentation and the control section of the spacecraft. At this time it was decided to pursue a design approach that would provide an experi-

ment package easily separable from the rest of the spacecraft that would contain the power system, attitude control system, and telemetry and command systems. With this design concept, it was easy to visualize a series of astronomy experiments that could be conducted on a spacecraft control section of a basic design. Thus, it was not long after the SAS-A (X-ray experiment) was started that a second experiment in gamma-ray exploration was initiated to be flown on the second SAS control section already under construction. A third SAS is now under preliminary design. This unit, called SAS-C, will carry another and more sophisticated X-ray experiment being designed by the Massachusetts Institute of Technology. In sum, the design objectives of the SAS program can be stated simply: to provide a basic structural and electrical satellite design which can be easily adapted to various astronomy experiments without a major redesign effort, and one that can be launched on a small vehicle such as the Scout rocket. The control section, containing the basic structure, power system, attitude control system, and telemetry and command system, is designed and fabricated by the Applied Physics Laboratory. The Laboratory is also responsible for integration of the experiment and spacecraft and for the qualification testing and launching.

The Small Astronomy Satellite Program is sponsored by the National Aeronautics and Space Administration and is managed by the Project Office at the Goddard Space Flight Center. The funding for the work reported in this issue of the *Digest* was provided by NASA through APL's contract with the Navy.

The Small Astronomy Satellite Program sponsored by NASA provides for a series of satellites that can be used for various astronomy experiments. The SAS-A mission is to map the X-ray sources in the entire sky. In addition, discrete X-ray sources are being observed for intensity fluctuations in the range of 1 to 20 keV. The control section of the SAS satellite, which includes the power system, the telemetry and command systems, and the attitude control and detection systems, was designed and fabricated by APL. APL was responsible for the experiment integration tests and the flight qualification tests. The launch preparation of the spacecraft from the San Marco Equatorial Range in Kenya, East Africa, was supervised by an APL crew. The SAS-A was launched December 12, 1970, and has been very successful in its scientific mission.

The SAS-A mission is exploration and mapping of the celestial sphere for X-ray sources in the spectral energy range from 1 to 20 keV. The X-ray energy in this range is heavily absorbed by the earth's atmosphere, and, therefore, either rocket probes or satellites are required for meaningful observation. The primary objectives of the SAS-A exploration are as follows.

1. Detect X-ray sources of intensity greater than 5×10^{-4} of SCO X-1, the strongest known source of X-rays in the sky.
2. Locate X-ray sources with an accuracy of 1 arc-minute for strong sources and 5 arc-minutes for weaker sources.
3. Search for temporal variations of several percent in X-ray source intensity.
4. Make simultaneous X-ray and visible light measurements of known X-ray sources and of any new sources discovered by this spacecraft.

The X-ray instrumentation consists of two independent directional detecting systems, one with a collimator aperture $\frac{1}{2}^\circ \times 5^\circ$, and the other $5^\circ \times 5^\circ$ full angle half amplitude. These two apertures are aligned 180° with respect to each other. A schematic of the sky scan is shown in Fig. 1. A star sensor is associated with each X-ray detector to determine the attitude of the spacecraft with respect to the visible stars. There are,

of course, many auxiliary components to this basic instrumentation such as power supplies, data storage units, and pulse-height and pulse-shape discriminators. Figure 2 shows an exploded view of the SAS-A X-ray experiment section. The data collection and transmission system is a part of the spacecraft control section. The SAS-A spacecraft was successfully launched on December 12, 1970, from the Italian San Marco Equatorial Range in Kenya, East Africa. The desired orbit was obtained at approximately 550 km circular with an inclination of 2.9° .

The SAS-B experiment will explore the universe for gamma-ray radiation sources. The gamma-ray experiment is being designed by NASA's Goddard Space Flight Center. The gamma-ray detector will measure both the intensity and the direction of arrival of the gamma rays in spacecraft coordinates. A star sensor located in the control section will determine the spacecraft attitude with respect to the star background to $\pm 0.25^\circ$, thus correlating spacecraft coordinates to celestial coordinates.

The SAS-C spacecraft will conduct another X-ray exploration incorporating more sophisticated detecting systems than SAS-A. An improvement in the attitude control system will enable more precise attitude determination and more

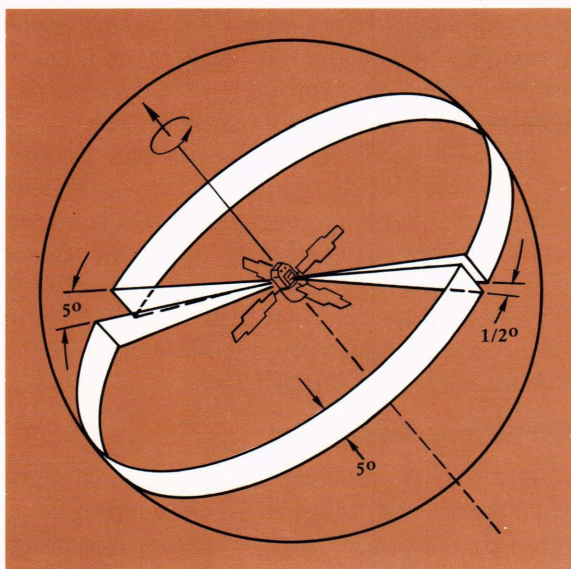


Fig. 1—Band of the sky swept by the two detectors during one revolution of the satellite.

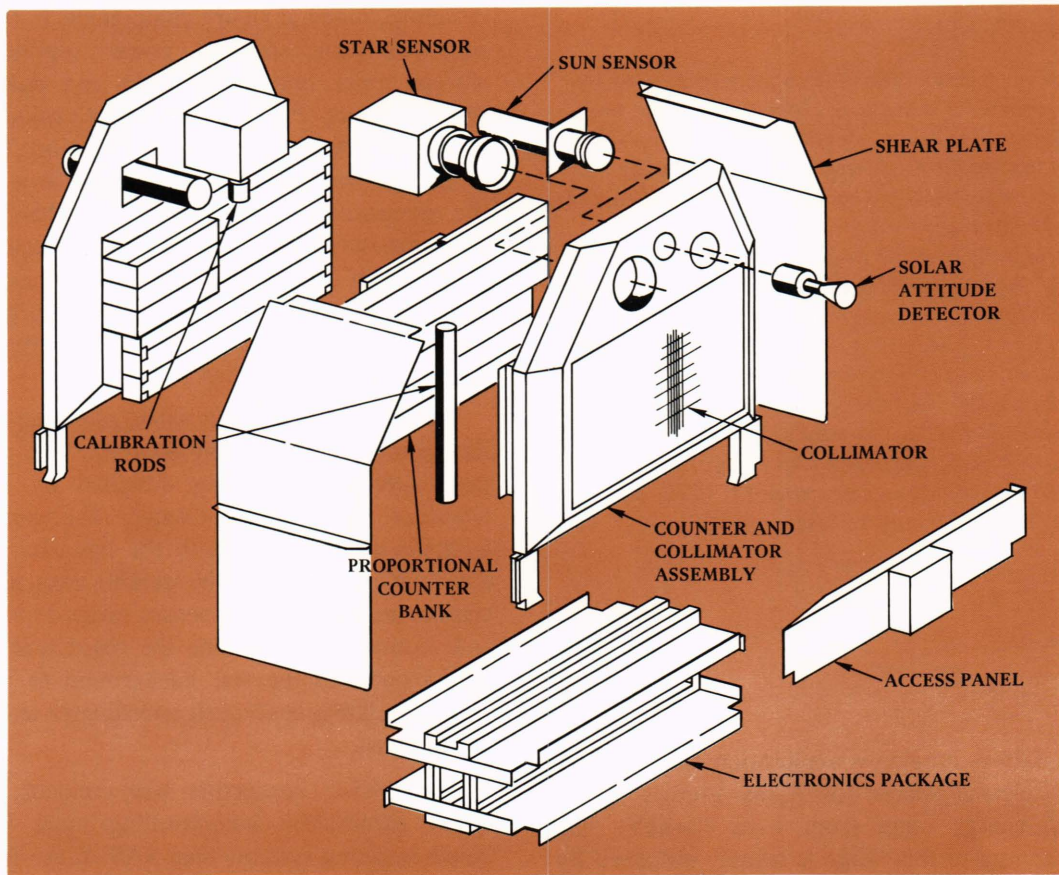


Fig. 2—Exploded view of X-ray experiment section.

flexible control over the measurement procedures. A closed-loop control system using a rate gyro for sensing the rate of spin will enable the spin rate to be controlled very precisely and to be programmed for specific measurement profiles.

The SAS-A control section provides the support structure for the experiment section and the subsystems for power generation, attitude control and detections, and telemetry and command. A drawing of the SAS-A control section is shown in Fig. 3. Each subsystem will be described briefly here with a more detailed discussion of the major systems in the following articles.

Structure Design

The SAS-A structure design has two basic requirements: (a) to provide the supporting structure between the 140-pound experiment package and the fourth-stage rocket, and (b) to provide a support for the internal instrumentation of the control section. A cylindrical sheet metal section

is strengthened with longitudinal riveted hat sections to carry the loads from the forward attach ring to the aft attach ring. The forward attach ring is a machined magnesium fixture riveted to the upper flange of the cylindrical structure. This forward ring provides the mechanical interface for the experiment package. There are attachment brackets that pick up the mounting bolts for the experiment. The aft ring provides for a marmon clamp fitting that attaches the spacecraft to an adapter section. This marmon clamp is the separation mechanism. The adapter section is a conical sheet metal construction that converts the outer diameter of the spacecraft to the attachment diameter of the fourth-stage rocket. This basic structural design allows the attachment of different experiment configurations by the simple expedient of installing a new forward attach ring.

The instrumentation of the control section is mounted on an internal structure that consists of a main deck of honeycomb material and a central

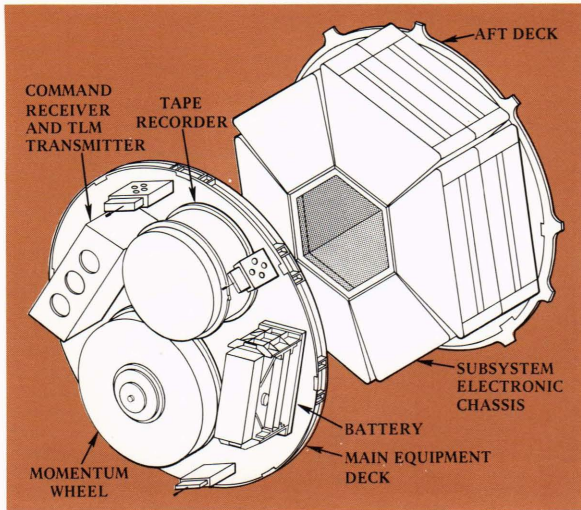


Fig. 3—Drawing of control section.

column. This assembly is attached to the outer cylindrical structure by eight attachment brackets near the top. Phenolic inserts at these eight points provide thermal isolation to control heat loss by conduction from the inner instrumentation to the outer shell. A number of instrument packages are mounted on the main deck. These are generally odd-shaped units such as the nickel-cadmium batteries, the momentum wheel, the magnetic tape recorder, the command receivers, telemetry transmitter, and recorder power supply. The electronic subsystem modules are housed in trapezoidal chassis and are mounted around the central column below the main deck. The wiring harness and interconnecting plugs are provided on the inside of this column. The inner structure with all the electrical subsystems installed can be supported by a handling fixture, giving access to the subsystems for electrical tests. The outer shell structure can be installed or removed as needed for vibration and other environmental tests.

Power System

The SAS-A power system¹ consists of a solar cell array with cells mounted on both sides of four-blade structures hinged to the control section structure and a nickel-cadmium battery whose charge is maintained by this solar cell array. In addition to this basic power system, shown in a simplified diagram (Fig. 4), a number of com-

ponents are included to provide the proper charge rate control for the battery and to properly dissipate the excess power generated at satellite altitude where solar input is a maximum. There are two independent charge control regulators of different design characteristics but which perform the same function. Either one of these charge control devices can be called into use by ground command. The primary charge control regulator is called CRAM, which stands for charge control regulator and monitor. This unit is a digital coulometer that continuously measures and integrates the current going into the battery from the solar array and the current being taken from the battery by the electrical load. The measurement made by this device then is the integral in ampere-minutes that has been either delivered to the battery from the solar cell array or taken from the battery by the electrical load. When the total energy delivered to the battery from the solar cell array equals 105% of that delivered to the load, the charge control regulator automatically decreases the charging rate to a trickle charge, thus preventing the battery from being overcharged. CRAM also monitors the temperature and voltage of the battery and modifies the value of the trickle charge to be consistent with the environment at that time, since the value of trickle charge required depends on the battery temperature and voltage. CRAM performs two other functions that are also very important. First, it limits the maximum current that can be delivered to the battery from the solar array to a value of 4 amperes. This limits the maximum charge rate to a value which, if exceeded, would damage the battery. Second, it provides the telemetry system with data that give the total number of ampere-minutes delivered to the load and returned to the battery by the charging circuitry. Thus the power profile during an entire orbit can be determined.

The second charge control regulator, which provides the same overcharge protection for the battery, is called a current and voltage limiter. It is a simple voltage limiter that uses a zener diode with amplified control characteristics. Its operation depends on the fact that a nickel-cadmium battery exhibits an increase in its terminal voltage when it reaches a fully charged state. The maximum value of the voltage during this period depends upon the temperature of the battery; therefore, the current and voltage limiter also monitors

¹ R. M. Sullivan, *Design and Test of the SAS-A Power System*, APL/JHU TG 1106, May 1970.

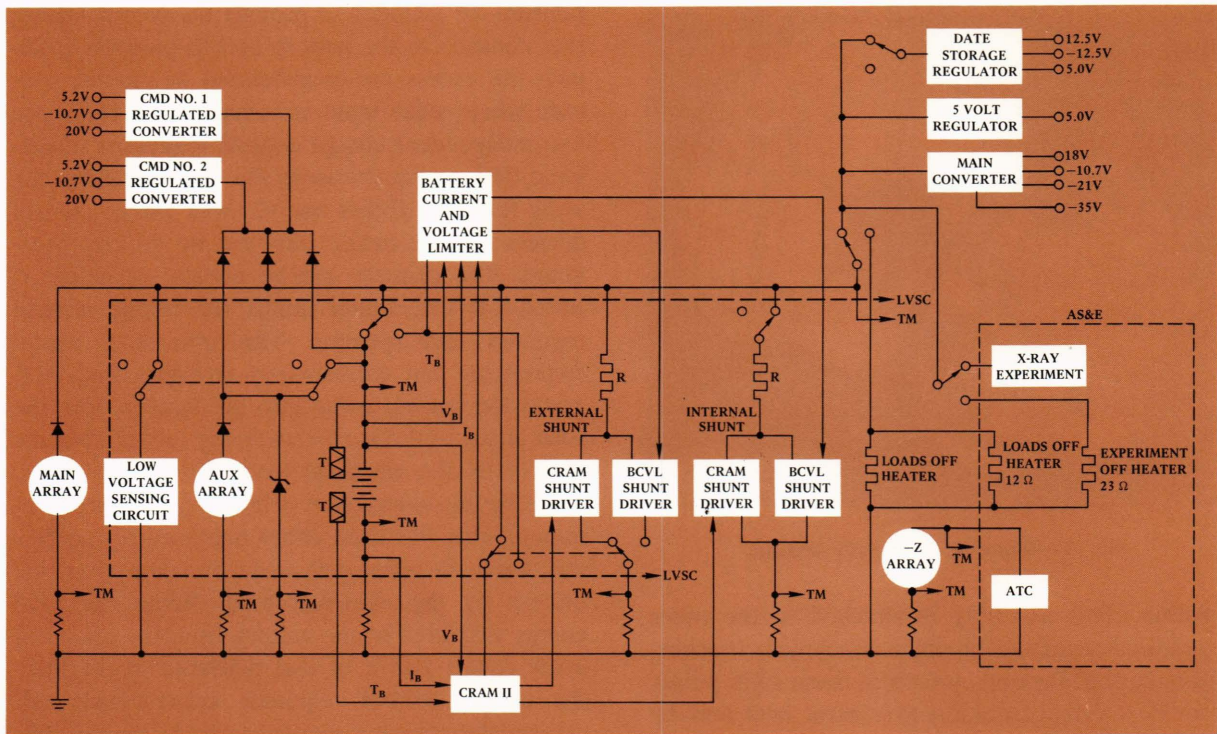


Fig. 4—SAS-A power system block diagram.

the temperature of the battery and modifies its control characteristics based upon this measurement. Thus, when the voltage at full charge rises to a specific value dependent upon the temperature, this charge control device reduces the charge rate to a trickle charge in a manner similar to that provided by CRAM. The current and voltage limiter also limits the maximum charge current to the battery to 4 amperes.

In addition to these two charge control devices, the power system uses power dump circuits to dissipate the excess power generated by the solar cell array when the batteries have reached full charge. There are two separate power dump circuits. One is located external to the spacecraft mounted on the solar cell blades, and the other is located internal to the spacecraft and uses a printed circuit resistor mounted on the main instrument deck. A temperature measurement on the nickel-cadmium battery determines whether this excess power will be dumped internally or externally. The temperature control is set to maintain the internal temperature at 55°F. If this excess power is dumped in the external circuits on the solar cell blades, it can be radiated to free

space without adding to the internal heat load. Both the internal and external dump resistors are controlled by transistor drivers; the control signals for these drivers are derived from whichever charge and control regulator is being used.

The power system includes a low voltage sensing switch that continuously monitors the battery bus voltage. It is designed to operate control relays that remove the battery from the charging circuit and electrical load when this voltage reaches a minimum value of 8.8 volts. The nickel-cadmium battery consists of eight individual cells to provide a nominal battery voltage of 10.7 volts. The value of 8.8 volts represents an individual cell voltage of 1.1 volts. When the cell voltage reaches a value close to 1 volt, there is danger of reversal in voltage polarity of a cell, which can cause permanent cell damage. Thus if for any reason the voltage approaches 1 volt, either due to a low charging capacity or an excessive load, the battery is automatically removed from the power bus until the potential problem is resolved. There is a small auxiliary solar cell circuit that remains connected to the battery and provides a small trickle charge even though it has been removed from the power bus.

In cases where the low voltage sensor switch has been activated, the procedure would be to return the battery to the power bus and determine its state of health. The command system provides a means of sequentially adding various subsystem loads to the power bus to aid in this diagnosis.

Attitude Control and Detection System

The attitude control system consists of two basic subsystems, one for attitude stabilization and the other for attitude maneuvering and spin rate control. An attitude detection system is used to monitor the spacecraft attitude, providing the information required to determine the experiment look angles and to compute the proper maneuver to reorient the spacecraft to a new position. The subsystem that maintains the spacecraft attitude stability uses a gyroscopic wheel that provides adequate angular momentum to stabilize the spin axis against external torques, such as those caused by aerodynamic pressure and the earth's magnetic and gravity gradient fields. A damper is provided to damp out any nutation or other oscillatory motions that are initiated by these external torques or by a maneuvering force. The portion of the attitude control system that provides the capability for reorienting the spacecraft to a new position includes an electromagnetic coil which is energized by a stabilized DC current. The magnetic dipole moment produced by this coil reacts against the earth's magnetic field to provide a precessing torque. This precesses the spin axis to a new position. A computer program used at the control station gives the information required to initiate and time the energizing of this coil. The other portion of this control system is operated only when it is required to add or subtract spin torque to maintain the desired spin rate. This system consists of two orthogonally mounted coils. The current produced in these coils is proportional to the magnetic field measured by a magnetometer which is itself orthogonal to the coil being energized. The magnetic dipole moments produced by these coils reacting against the earth's magnetic field produce a torque that either adds to or subtracts from the initial spin rate, depending upon the sense of the current in one of the coils.

The attitude detection system consists of an orthogonal set of magnetometers in the X, Y, and Z planes and a digital sun sensor to measure the

azimuth and elevation of the sun line with respect to the spin axis of the satellite.

Command System

The command system for this satellite uses a pulse code modulation (PCM) and instruction system that is standard in the NASA Space Tracking and Data Acquisition Network (STADAN). The command word is a 64-bit digital word which is transmitted at a rate of 64 bits per second. This command word can be coded in many different combinations to use a number of different instrumentation techniques. The SAS-A command system is redundant in that there are two command receivers, each with independent antennas. These two command receivers feed two bit detectors and logic decoding networks. The power circuits and some signal circuits are switched by 36 magnetic latching relays that are redundantly controlled to an "on" or "off" position by the command logic. In addition, a 24-bit word is coded to provide special commands for other circuits in the satellite system.

Telemetry System

The SAS-A telemetry system is a digital PCM system controlled by redundant crystal oscillators. There are two digital subcommutators that select the digital data and two analog subcommutators that feed an analog-to-digital converter. The data collected from all sources are properly combined and clocked into a serial bit stream by a format generator. The data are then fed to an RF transmitter and to the magnetic tape recorder through the PCM encoder in a manchester split phase code. The real time data rate is 1000 bits per second. Data are transmitted continuously via the RF transmitter and simultaneously stored on a magnetic tape recorder for subsequent playback. The magnetic tape recorder has a storage capacity of approximately 6 million bits or 100 minutes of data. The data throughout one orbit are thus preserved on the magnetic tape recorder and can be dumped by command at a data rate 30 times the real time rate when the satellite is over the prime SAS telemetry receiving station. The playback time is then approximately 3.4 minutes.

Testing and Qualification Criteria

Each component of a subsystem is very thor-

oughly bench tested after fabrication and inspection. The bench testing includes voltage tolerance performance and hot and cold temperature cycling. These tests are designed to show up any manufacturing deficiencies or errors in circuit wiring. Once a component of the subsystem has been thoroughly checked out and is operating properly, it is assembled into a complete subsystem such as the command system or telemetry subsystem. At this point the tests are broadened to include a check of all the interface characteristics and the subsystem performance. These tests require the design of special equipment to simulate various input data characteristics of the overall spacecraft system. At the completion of bench testing, the subsystems are given an environmental qualification test which includes vibration and a test in the thermal vacuum chamber. The thermal vacuum testing includes several cycles at temperatures of approximately -10° to $+50^{\circ}\text{C}$. The components are maintained at these temperatures for a number of hours to insure the internal portions of the package have stabilized. Full electrical performance is required during these tests and any system failure requires that the test be terminated to determine the cause of the failure.

After subsystems have been thoroughly qualified according to their specific test procedures, they are assembled into a complete satellite system. At this level of testing, complete system checks are made to determine that all the subsystem interfaces and electrical characteristics have been properly designed and instrumented. The performance of each subsystem is again thoroughly checked, and tests are conducted to assure that the complete satellite system performs according to the requirements of the design specification and meets the mission objectives.

The final testing sequence involves environmental testing of the complete spacecraft, including the experiment subsystem. A complete system test is made while the spacecraft is in the thermal vacuum chamber to insure that all its performance characteristics are within specifications at all temperatures in a vacuum environment. The thermal design characteristics are tested in the thermal vacuum chamber to check the internal temperature characteristics in both hot and cold cases. During this test, thermal radiation blankets are adjusted to provide the proper heat losses from the internal components to the outer shell,

which acts as a thermal radiator. Vibration tests of the completed spacecraft are conducted to assure that the assembly has been satisfactory and that there are no mechanical resonances in the structure that can cause the overstressing of any components of the subsystem. If there are excessive magnetic dipole moments they must also be properly compensated. The changes in magnetic characteristics as various subsystems are energized are also measured. The mass properties of the spacecraft, such as the weight and moments of inertia, must be measured. The final measurement of the mass properties must be done after all the modifications have been made and the spacecraft is in its flight configuration.

During the system level tests, data are gathered by the telemetry system and fed to a computer for data processing and display. The digital telemetry system requires complex data handling procedures that can be most efficiently handled by computer. The computer provides bit synchronization and clocking, decommutation, and data grouping. Programs are written to operate and collect specific groups of data and display the results on either a remote cathode ray tube (CRT) for numeric character display or by a line printer for detailed examination. The data are also stored on digital tape for later reruns. Two types of computer are used in the SAS program: the CDC 3200 located in the Satellite Injection Station at APL and the XDS Sigma 5 computer located in a portable van. There are two Sigma 5 computers; one was dispatched to Africa for prelaunch tests and one remains at NASA for another program, and will be shared with the future SAS testing schedules.

This computer data-processing system provides a very flexible way to handle a variety of test procedures. The remote CRT display provides the test conductor with the real-time data he needs for a specific test. A remote typewriter control to the computer allows the test conductor to alter the data display according to his needs. A program is available so that the computer can send commands automatically when asked to by operator control. These commands can be used to set up special operating modes in the satellite and then to collect the data for a special test. Thus a series of preplanned tests can be programmed and controlled by the computer. This scheme was used extensively in testing the X-ray experiment using

various calibration sources and determining proper operation and response.

The Sigma 5 van computer was used at the launch site with a radio frequency link for telemetering and command to perform the same types of test that were performed before shipment, thus verifying the flight readiness at the launch site.

Launch Operations

As mentioned in the Introduction, the SAS-A satellite was launched from San Marco Equatorial Range located off the coast of Kenya, East Africa. The launching platform and the range control platform are located about 3 miles offshore in Formosa Bay, which is approximately 18 miles north of the ancient town of Malindi. The launching platform is known as the San Marco platform and the range control tower as the Santa Rita platform. The San Marco Equatorial Range is operated by the Italian Government through the Italian Space Agency known as Centro Ricerche Aerospaziali (CRA). The CRA is administered by the University of Rome. The Scout launching vehicle was transported to Kenya via commercial shipping. Since the ships are unloaded at the launch platform, they do not enter a commercial harbor. The spacecraft and associated test equipment were flown to Kenya by commercial air cargo and trucked from Nairobi to the launching site. The spacecraft field crew consisted of about 20 personnel from the Applied Physics Laboratory, a number of NASA project representatives, and four representatives of AS&E. The field preparations prior to launch day occupied one month. A large portion of this time was involved in vehicle preparation, and a number of complete system tests were conducted on the spacecraft prior to mating with the rocket. The spacecraft was attached to the rocket fourth stage about 7 days prior to launch. During this final period, arming preparations and other final integration tests were performed to assure that all was in readiness for the launch. A dress rehearsal countdown was held approximately 3 days prior to launch, providing some recovery time for any deficiency that had occurred during the practice countdown. The launch (Fig. 5) occurred on December 12, which marks the independence day for the Kenya Republic, and our first U.S. satellite to be launched from this facility was named "Uhuru," which is the Swahili word for "Freedom."

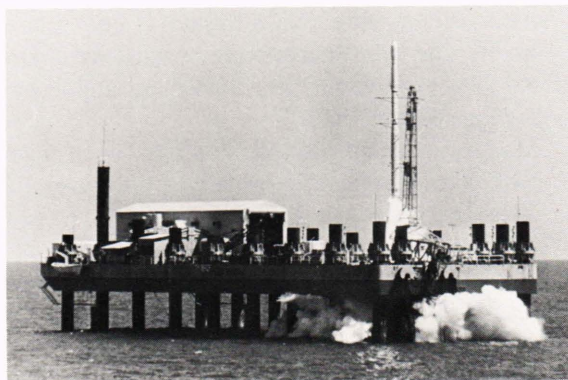


Fig. 5—SAS-A lift-off.

In-Orbit Operations

The operational sequence immediately after launch is automatic. The heat shield ejection occurs at about 350,000 feet altitude and the fourth stage is ignited at 569 seconds after launch. The fourth stage and the spacecraft are spin stabilized at about 180 rpm during the fourth-stage coast and burn time. About 1½ minutes after the fourth-stage burnout, the solar blades are deployed by the release of a "yo-yo" cable. This deployment decreases the spin rate to about 5 rpm. After a delay of another minute or so the separation is initiated. Springs installed at the separation plane insure a parting rate of about 3 feet per second relative to the spent fourth-stage casing.

During the first orbit over the prime NASA Control Station at Quito, Ecuador, the momentum wheel and nutation damper were uncaged. The momentum wheel came up to speed quickly, and by the next orbit the spacecraft had stabilized to a very low nutation angle. During the second orbit the spin-despin system was energized to begin a spin rate reduction to the desired ½ rpm. Reducing the spin rate to this low value nominally requires a day or so. It was soon discovered that the ambient temperature of the satellite at launch and the long thermal time constant prevented the satellite from cooling at a sufficiently rapid rate to make up for the added thermal energy dissipated by the spin-despin system. Therefore, in order to avoid overheating the internal instruments, particularly the battery and the tape recorder, it was necessary to duty cycle the spin-despin system. Thus, this operation took somewhat longer than was planned.

After the satellite was adjusted to the proper spin rate and all systems were verified as operational, the experiment was turned on and began collecting scientific data.

Many interesting observations have been made which have significantly added to the scientific knowledge related to X-ray astronomy. Among the first important discoveries is the large amplitude pulsations in X-ray energy from Cygnus X-1.² These pulsations occur several times a second with a duration of a fraction of a second. These pulsations exceed 25% of the average source intensity. The conclusions were that a pulsating X-ray star had been discovered.

The data have also provided a refinement of the locations of the X-ray sources of Cygnus X-1, Cygnus X-2, Scorpius X-1, and Cassiopeia A.³ These sources have been verified as to previous locations but the new data have significantly smaller error limits.

New sources of X-ray radiation have been observed from the Seyfert galaxies NGC 1275 and NGC 4151.⁴ Significant fluxes of X-rays in the range 2 to 6 keV have been observed from these sources. Their positions have been verified to a positional error of 0.01 square degree for NGC 1275 and of 0.56 square degree for NGC 4151. X-ray emission has been observed from the extragalactic objects M87, NGC 5128, and 3C 273;⁵ these sources are in the Virgo Cluster. These objects have been reported previously as X-ray sources; however, the emission from the direction of M87 extends over a region of more than 1° and may represent the emission from two sources, one being M87, the other M84.

Pulsating X-rays have been observed from Centaurus X-3.⁶ These pulsations are of large

amplitude comprising at least 70% of the total flux from the source. The period of oscillation is somewhat longer than that from Cygnus X-1, being about 5 seconds. In addition, there are intensity changes of a factor of 10 which occur in a period of about 1 hour. There also appears to be a longer period of about 3 months.

All subsystems have operated as planned and have given reliable performance with two exceptions. There has been an intermittent failure of the telemetry data modulation on the RF transmitter. About two weeks after launch, the modulation index decreased to a very low value, but not zero. Data could still be recovered with difficulty in the increased noise. At the time of this failure the internal temperature of the spacecraft was about 80°F. This temperature was lowered by maneuvering the satellite to a cooler attitude with respect to the sun. When the satellite cooled to about 75°F, the modulation returned to normal. It remained normal for many weeks, when it again dropped to about one-half the normal level. Late in June the modulation again dropped to a lower level, but not zero, and at this time the internal temperature was about 56°. At present it has not been possible to deduce the actual failure mechanism with the available data.

The other malfunction involved the tape recorder for data storage. It failed to operate in the playback mode about two months after launch. The failure has been diagnosed as a seizure of the bearings in the playback motor. The power drain by the tape recorder in this failure mode is equal to the normal current plus a stalled motor current. To compensate for this failure, real-time receiving stations have been activated at various locations around the equator to supply about 50% of the total data available in one orbit. When the tape recorder was operating normally, data from the entire orbit were recoverable. A second tape recorder has been installed in SAS-B to add some reliability in this area. In addition, modifications are being made to the tape recorder to improve its reliability.

The attitude stability and control has been excellent and has met all of the requirements of the scientific mission. The operation from the NASA Control Center and the data collection at Quito, Ecuador, are continuing satisfactorily.

² M. Oda, P. Gorenstein, H. Gursky, E. Kellogg, E. Schreier, H. Tananbaum, and R. Giacconi, "X-Ray Pulsations from Cygnus X-1 Observed from *Uhuru*," *Astrophys. J. (Letters)* **166**, May 15, 1971, L1-L7.

³ H. Tananbaum, E. Kellogg, H. Gursky, S. Murray, E. Schreier, and R. Giacconi, "Measurement of the Location of the X-Ray Sources Cygnus X-1 and Cygnus X-2 from *Uhuru*," *Astrophys. J. (Letters)* **165**, Apr. 15, 1971, L37-L41.

⁴ H. Gursky, E. M. Kellogg, C. Leong, H. Tananbaum, and R. Giacconi, "Detection of X-Rays from the Seyfert Galaxies NGC 1275 and NGC 4151 by the *Uhuru* Satellite," *Astrophys. J. (Letters)* **165**, Apr. 15, 1971, L43-L48.

⁵ E. Kellogg, H. Gursky, C. Leong, E. Schreier, H. Tananbaum, and R. Giacconi, "X-Ray Observations of the Virgo Cluster, NGC 5128, and 3C 273 from the *Uhuru* Satellite," *Astrophys. J. (Letters)* **165**, Apr. 15, 1971, L49-L54.

⁶ R. Giacconi, H. Gursky, E. Kellogg, E. Schreier, and H. Tananbaum, "Discovery of Periodic X-Ray Pulsations in Centaurus X-3 from *Uhuru*," *Astrophys. J. (Letters)* **167**, July 15, 1971, L67-L73.