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ACHIEVING ARIEL II DESIGN COMPATIBILITY

by Allen L. Franta and Arthur C. Davidson

Goddard Space Flight Center
Greenbelt, Md.

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ABSTRACT

The Ariel II spacecraft mechanical and electrical design, fabrication, integration, environmental qualification and launch operations were accomplished by the Goddard Space Flight Center and United Kingdom experimenters. The experiments for this satellite were provided by British scientists. The solution of the major problem of achieving compatibility between the spacecraft subsystems and experiments was a joint effort involving Goddard Space Flight Center engineers and the British scientists. The principal items involved were the orbital requirements versus launch vehicle capability, location of appendages, experiment view angles, interface compatibility of subsystems, data rate versus telemetry bandwidth, spacecraft power requirements, ground loop interference, magnetic susceptibility, experiment programming, pulse current induction, and thermal control.

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INTRODUCTION

The Ariel II satellite (1964 15A), shown in Figure 1, continued the joint United States and United Kingdom research program in and above the ionosphere. Its objectives were to supplement the ion, electron and radiation studies of Ariel I (ο2) by investigating certain phenomena in the atmosphere, the ionosphere, and beyond. Successfully launched from Wallops Station, Virginia on March 27, 1964, the satellite placed three experiments in an elliptical orbit of 730 nautical miles apogee and 157 nautical miles perigee at an inclination of 51.66 degrees latitude with a period of 101.37 minutes. The three experiments on board supplied data on the vertical distribution of ozone in the atmosphere, the Galactic Noise Spectrum from 0.75 to 3.0 megacycles and the number and size of micrometeoroids encountered in the orbit.

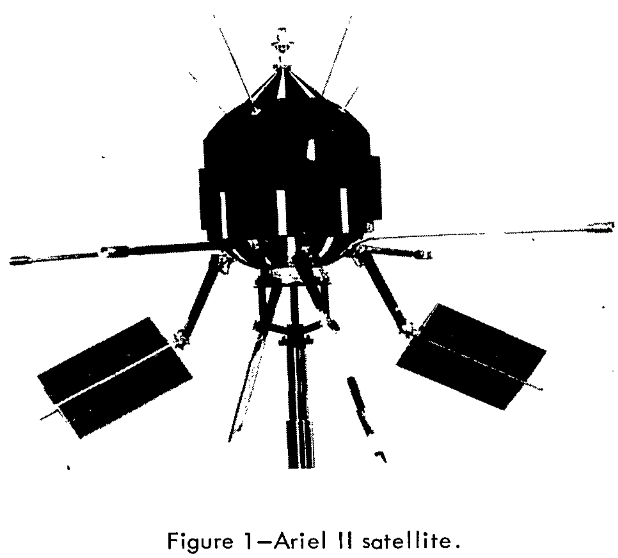

Figure 1—Ariel II satellite.

The operational success of the Ariel II satellite is a result of the combined efforts of many scientists, engineers and technicians of both the United Kingdom and the United States (Appendix A). The United Kingdom was responsible for all flight instrumentation pertaining to the experiments and for data reduction and analysis; the United States was responsible for the design, fabrication, integration and environmental qualification of prototype and flight spacecraft, except for the scientific experiments. Tracking and data acquisition was a joint responsibility. The satellite was launched by a United States Scout rocket in March 1964 and has been performing well for more than seven months. It is hoped that the design goal of a full year of operation will be achieved. A third satellite in this joint international program is being completely designed and produced in the United Kingdom. It is scheduled to be launched by a Scout rocket in 1967.

DESIGN COMPATIBILITY

During the course of the development of Ariel II, one of the "must" objectives was the attainment of functional compatibility of all the spacecraft subsystems and experiments. Individual subsystems engineers and experimenters are generally concerned first, with obtaining a reliable operational design and second, with compatibility of their design in the spacecraft environment. Being human, designers tend to be very concerned when their package receives interference from other subsystems but not quite as concerned should their package be the source of interference.

Design compatibility is a prime responsibility of the Project Manager. Either he or members of his staff must monitor all design concepts and completed configurations to assure that the systems and subsystems employed in the completed assembly are compatible from both a mechanical and electrical point of view. Compatibility problems will require either modification or compromise for their solutions. One modification may beget another. This occurs more frequently with mechanical modifications than with electrical. A mechanical or structural modification requiring a package or appendage location change often dictates minor changes in the location of other assemblies. To secure compatibility of all the subsystems and mechanical components, their functional interrelationships must be considered. The principal items involved are location of appendages, experiment view angles, interface compatibility of subsystems, data rate versus telemetry bandwidth, spacecraft power requirements, ground loop interference, magnetic susceptibility, experiment programming, pulse current induction and thermal control. The magnitude of the task involved in achieving design compatibility can be understood by noting the 44 subsystems listed in Table 1.

Design compatibility, being a major problem in any satellite program, must be considered from conception to completion. It follows the over-all program planning such as the major events in the Ariel II schedule, which were:

Feasibility Study and Project Approval
System Design Specifications
Subsystem Design Specifications
Structures and Subsystems Fabrication and Test
System Integration
Environmental Qualification
Launch Operations

A complete understanding of the dynamic nature of achieving design compatibility can be obtained by examining these events in detail.

Table 1

Ariel II Subsystems.

+ Power Supply	Micrometeoroid Drod A
Inverter Power Supply	Micrometeoroid Drod B
Batt. Switch Network	Micrometeoroid Irod A
- Power Supply	Micrometeoroid Irod B
Encoder No. 1	Micrometeoroid Electronics
Encoder No. 2	Galactic Noise Reel
Encoder No. 3	Galactic Noise Electronics
Battery A & Battery B	Galactic Noise Ferrite Loops
Solar Paddles (4)	Ozone Broad Band
Programmer No. 1	Ozone Spectrometer A
Programmer No. 2	Ozone Spectrometer B
Command Receiver	Ozone Electronics
Decoder	Antenna Booms (2)
Tape Recorder Converter	Inertia Booms (2)
Transmitter	Yo-Yo Despin Assembly
Tape Recorder	Mechanical Structure
Data Storage Control	Solar Paddle Arms (4)
Undervoltage Det. & Recycle	Interconnecting Assembly
Staticizer	Telemetry Antennas (4)
Tie Down Assembly	Top Dome Cover
Separation Assembly	Lower Dome Cover
Heat Shield Assembly	Mid Skin Covers (2)

FEASIBILITY STUDY AND PROJECT APPROVAL

The demand for space research in a specific area of investigation can originate in any of the space research-oriented universities and organizations in any of several countries throughout the world. At the time of project origin, the major portion of the activity is centered around a feasibility study generated by a small technical staff (Reference 1). This staff is generally headed by a project manager who requests the assistance of qualified technical personnel to assist him in the preparation of this study. Figure 2 addresses itself to the elements involved in the project origin. The mission requirements must be defined in order that the scientific experiments required to accomplish it can be properly chosen. In planning the experiments capable of fulfilling the mission, considerable emphasis must be placed on a practical spacecraft design. If details can be adopted from tried and flight-proven spacecraft designs, the mission feasibility is enhanced (the structural and component subsystems developed for Ariel I were used wherever possible to minimize the engineering and design work required for Ariel II). The mission requirements are defined with full consideration given to the choice of a launch vehicle capable of achieving the necessary orbital parameters. Information concerning satellite tracking and data acquisition requirements is factored in with the orbital requirements. If the established NASA tracking network, commonly known as STADAN (Space *T*racking *a*nd *D*ata *A*cquisition *N*etwork), can be used for tracking a spacecraft in the planned orbit, the problem is simplified. Twelve STADAN stations were used on Ariel II with United Kingdom personnel operating the Winkfield facility (Appendix B). Last but not least are the resources requirements. What will be the cost in manpower, money and

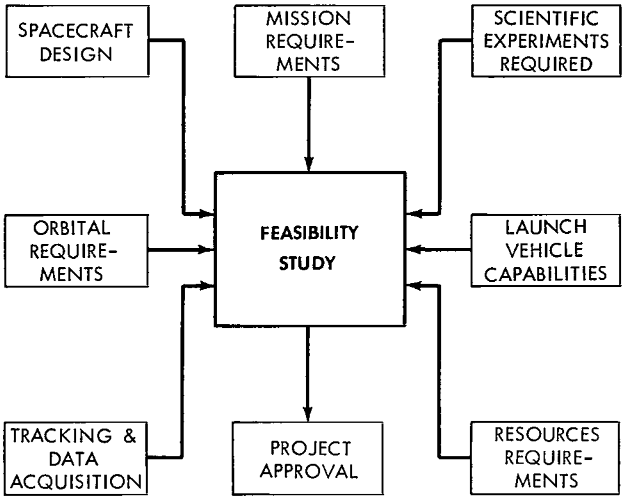

Figure 2—Elements of spacecraft project origin.

facilities? Are the resources available? Where? What will be the impact on other projects already in progress? What would be the most efficient manner of solving the scientific and technical problems? These and many other questions must be answered in order for Center management to determine whether the project is feasible. After project feasibility has been demonstrated satisfactorily to Center management, the results of the study are submitted for consideration to the NASA Headquarters officials concerned with the technical approval and funding of the proposed mission. The written study is generally followed by oral presentations and discussions to clarify all details and points in question.

During the feasibility study, the detailed compatibility problem is not paramount; however, any major problems concerned with the experiments are examined closely. The major areas investigated for Ariel II during this phase were:

Probable Behavior of the Galactic Noise Experiment Flexible Dipole (130 ft.) — Studies of the orbital behavior of the long flexible dipole were undertaken by UK and GSFC theoreticians. The investigations did not result in conclusive agreement concerning the half-life spin; however, there was general agreement that the antenna would behave essentially the same as the rigid antennas on Alouette.

Solar Paddle and Boom Shadowing of the Micrometeoroid and Ozone Spectrometer Sensors — A compromise was made on compatibility of the boom and paddle locations versus experiment sun viewing. The four booms were located for minimum shadow on the Ozone spectrometers and micrometeoroid sensors. The paddles were drooped from the body of the spacecraft so that shadowing of the experiment apertures would not occur until the sun was more than 40 degrees below the spacecraft equator.

Galactic Noise Radiometer Interference from Inverter or IF. Carrier Oscillators — Harmonics generated by the inverter or carrier oscillators could cause interference in the 0.75 Mc to 3.0 Mc band of the Galactic Noise experiment. Low frequency oscillators (i.e., 1 to 10 kc) could also pose a problem. The carrier oscillators of the Ozone spectrometers amplifiers were at 50 kc. Simulated power supplies were furnished to the experimenters during the early planning periods and an interference check on the bench was made. Since interference was not observed during these checks with the simulated power inverters, some assurance was established that compatibility in the spacecraft could be achieved.

Orbital Requirements vs. Launch Vehicle Capabilities — The project orbital parameters desired and the launch vehicle capabilities were reviewed and compared at this time for compatibility. The 4-stage Scout vehicle was found to be capable of injecting a payload of 165 pounds or less in the desired orbit (Appendix B). The nominal orbital parameters desired versus the computed parameters observed after launch are as shown in Table 2.

Data Rate Versus Telemetry Bandwidth — Pulse frequency modulation was chosen for the Ariel II telemetry. This proven type of telemetry was selected for its simplicity and high efficiency, having been employed in a number of small GSFC scientific satellites and probes (References 2 and 3). The Ozone and Galactic Noise experiments had data rates too high for compatibility with the data encoding equipment, thus requiring the design of staticizers to stabilize the rate during the data sampling time.

Table 2

Nominal Orbital Parameters Versus Computed Parameters.

	Predicted	Observed
Apogee	810 NM	730 NM
Perigee	150 NM	157 NM
Inclination	52.028 deg	51.662 deg
Period	102.5 min	101.37 min
Eccentricity	0.08413	0.07417
Sun Aspect Angle	87.0 deg	87.0 deg
Spin Rate	5.4 rpm	5.6 rpm

Spacecraft Power Requirements — A close estimate of the spacecraft power requirements was made during the feasibility study. All subsystem voltages and power requirement estimates were obtained from the designers. These were totaled for the over-all system power estimate. The Ariel II power requirement was estimated to be 12.5 watts in sunlight and 5.3 watts in shadow for a supply voltage of 14.5. Metered power measurement made during the integration phase proved this estimate to be very close. The measured power for sunlight operation was 12.65 watts and shadow, 5.36 watts with a 14.5 volt supply. The solar cell array for the spacecraft power supply was located on paddles attached to extended hinged arms. Solar array installation on the body of the satellite was considered for a time; however, this configuration would have required a larger diameter spacecraft to provide a greater surface area. When the original spacecraft design was laid down, it was determined that only the 25.7 inch Scout nose fairing would be available.

Ozone Experiment Programming — The Ozone experiment on Ariel II acquires its useful data during the sunrise and sunset periods of the orbit. During the feasibility study, a plan to turn on the Ozone experiment at sunrise via the solar current was initiated. This method later proved to be incompatible with the experiment since solar current was not available during the spacecraft early dawn (1% sunlight). The sunrise sensing was changed to solar paddle voltage sensing. This provided an Ozone turn-on signal at the desired early dawn light level. In orbit, the sunrise turn-on occurs 15 seconds prior to the spacecraft passing through the region of Ozone data readout. The sunset turn-on time was obtained by counting a fixed time interval from the sunrise event and turning the Ozone experiment on a few minutes before the time of the earliest sunset. This time interval for the Ariel II orbit was 60 minutes.

With regard to the Center resources for Ariel II, the amount to be accomplished in-house versus contract had to be determined. Original designs of scientific satellites are usually accomplished in-house to develop competence and establish capability. It is Center policy to contract, where possible, for tasks not considered basic innovations. Since Ariel II was in many respects a design modification or extension of Ariel I, a portion of the subsystem fabrication effort and all of the integration effort was contracted to the Aerospace Division of the Westinghouse Electric Corporation, Baltimore, Maryland. The subsystems supplied by GSFC were produced on other contracts or at the Center.

SYSTEM DESIGN SPECIFICATIONS

After project approval, all elements considered in the feasibility study were defined in a detailed specification of the complete system. The final selection of experiments was approved by the Space Sciences Steering Committee of NASA Headquarters (Figure 3). Of the five experiments originally proposed by the British National Committee on Space Research, the three previously described were mutually agreed upon as comprising the most desirable experiment package (Reference 4).

Spacecraft-Vehicle Interface

The spacecraft-vehicle interface problems were examined in detail to insure functional and dimensional compatibility. The mechanical designers finalized the structural design, conducted stress analyses and produced design specifications of all their subsystems (basic structure, despin mechanism, separation system, reeling devices, experiment and inertia booms and release mechanisms). The size and approximate weights of all subsystems and experiments were determined to arrive at the spacecraft's physical configuration, total weight, and moments of inertia. This information enabled the vehicle personnel to determine the adequacy of the proposed launch vehicle. The spin-up and despin numbers were determined jointly by the vehicle and mechanical design personnel.

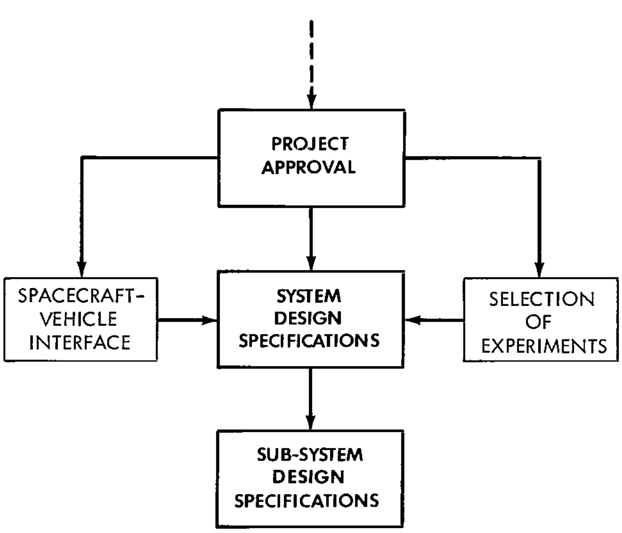

Figure 3—Generation of system specifications.

Though spin-stabilized satellites have no active attitude control system as contained in large spacecraft systems such as the Orbiting Geophysical Observatory, the mechanical group was concerned with attitude changes due to perturbations of the spacecraft while in orbit caused by solar radiation pressure, magnetic torques, and aerodynamic drag. To maintain a favorable orientation of the spin axis, the spacecraft system must be designed so that the ratio of the spin axis moment of inertia to the pitch or roll axis moment of inertia is always greater than unity. This is another reason for the constant monitoring of the spacecraft weight which always seems to grow.

The vehicle was also examined to determine the location of the spacecraft umbilical plug, the separation umbilical connector, the turn on plug, the instrumentation plug, and the battery charging requirements. The over-all spacecraft dimensions in a folded configuration were compared for physical compatibility with the inside dimensions of the Scout nose fairing (or heat shield) selected for this mission. The location of all necessary access doors in the nose fairing for the payload plugs was established at this time.

System Design

The mechanical and electronic designers work very closely in determining the optimum location of all subsystems within the spacecraft structure. The preliminary layouts are made with spin stability as the only concern. Power dissipation, noise, RF interference and induced magnetism are all considered next in the detailed system design. The final design is a compromise, with no individual subsystem designer being completely satisfied (e.g., if the mechanical designers located all of the appendages with no consideration given to the antenna pattern, the antenna designer would have a hopeless task. Also, experiments with large angles of view cannot be completely free from seeing an appendage even though they are afforded the most ideal location). Many potential trouble areas are avoided through this mechanical and electrical integration interplay.

The Project Management Staff, assisted by electronic integration personnel, contributed to the over-all design of the spacecraft by virtue of their advisory responsibilities to the subsystem designers. They exercised an influence on the selection of housekeeping functions and on the methods of telemetering them. They served a liaison function between the power system designer and subsystem designers, exercising some influence on the design of all subsystems. They established the guidelines with regard to system electrical design and fabrication and, based on previous experience, they assisted in the system design by identifying and suggesting solutions to potential noise, interference between subsystems, and magnetic problems.

Generation of System Design Specifications

While the Ariel II system design was in progress, the Project Manager was required to prepare the Project Development Plan which defined the major areas of responsibility for all project personnel and established a detailed project master schedule (Reference 4). Following completion of the Project Development Plan, the System Design Specifications Document was generated through

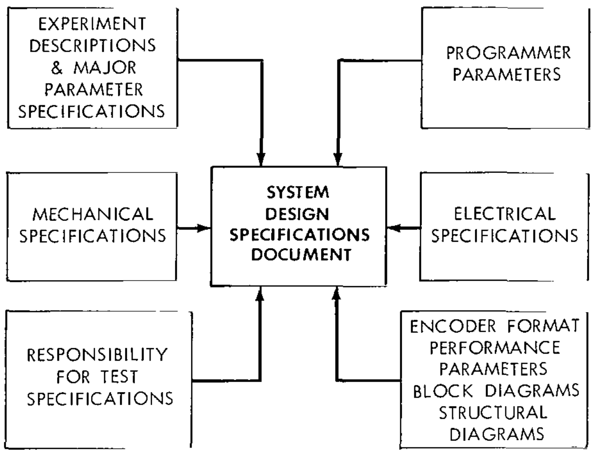

Figure 4—System design specification document contents.

joint consultation between the project management staff and all the subsystem designers (Figure 4). Included in it were detailed values and limits of the major parameters of all the spacecraft subsystems and experiments. The chief parameters were: voltages, voltage regulation, power requirements, impedance matching, weight and size, recording speed, playback rate, encoder format, appendage final location, performance parameters and programming sequence (Reference 5). This document was used as a reference for the major parameter values throughout the entire project. A detailed description of each experiment was included. It explained the measurement methods employed by the experimenters and also the data rate required for the desired measurement accuracy.

Responsibilities for the formulation and approval of all subsystem specifications and test procedures were firmly stated in the Systems Design Specifications Document. Each project engineer (or engineer-in-charge) of each subassembly or subsystem was directed to formulate design and test specifications. All specifications, whether generated by GSFC or contractor personnel, were submitted to the Project Manager for approval. All test procedures were approved by the Project Test Manager. By directly defining the areas of responsibility, the Project Manager maintained control and uniformity, which brought about the desired compatibility between the numerous and varied events occurring throughout the project.

The programming requirements of the spacecraft were also listed in precise detail in the subject document. Since the programmer subsystems of Ariel II had either a direct or indirect interface with all the electronic subsystems, maintaining compatible programming sequences was a continuous problem throughout the design, fabrication, integration and environmental qualification phases. These requirements served as a ready reference for all design, integration, and test personnel.

SUBSYSTEM DESIGN SPECIFICATIONS

As shown in Figure 5, the major mechanical subsystems are the structure, thermal control, separation device, attitude control, despin mechanism and parts of experiments. The major electronic subsystems are the RF portion (modulators, transmitters, antennas and command receivers); spacecraft instrumentation (performance parameters, housekeeping functions and attitude control); encoding, data handling, power (solar paddles, batteries, converters, regulators and power supplies) and the experiments. Except for the Ariel II experiments, furnished by the

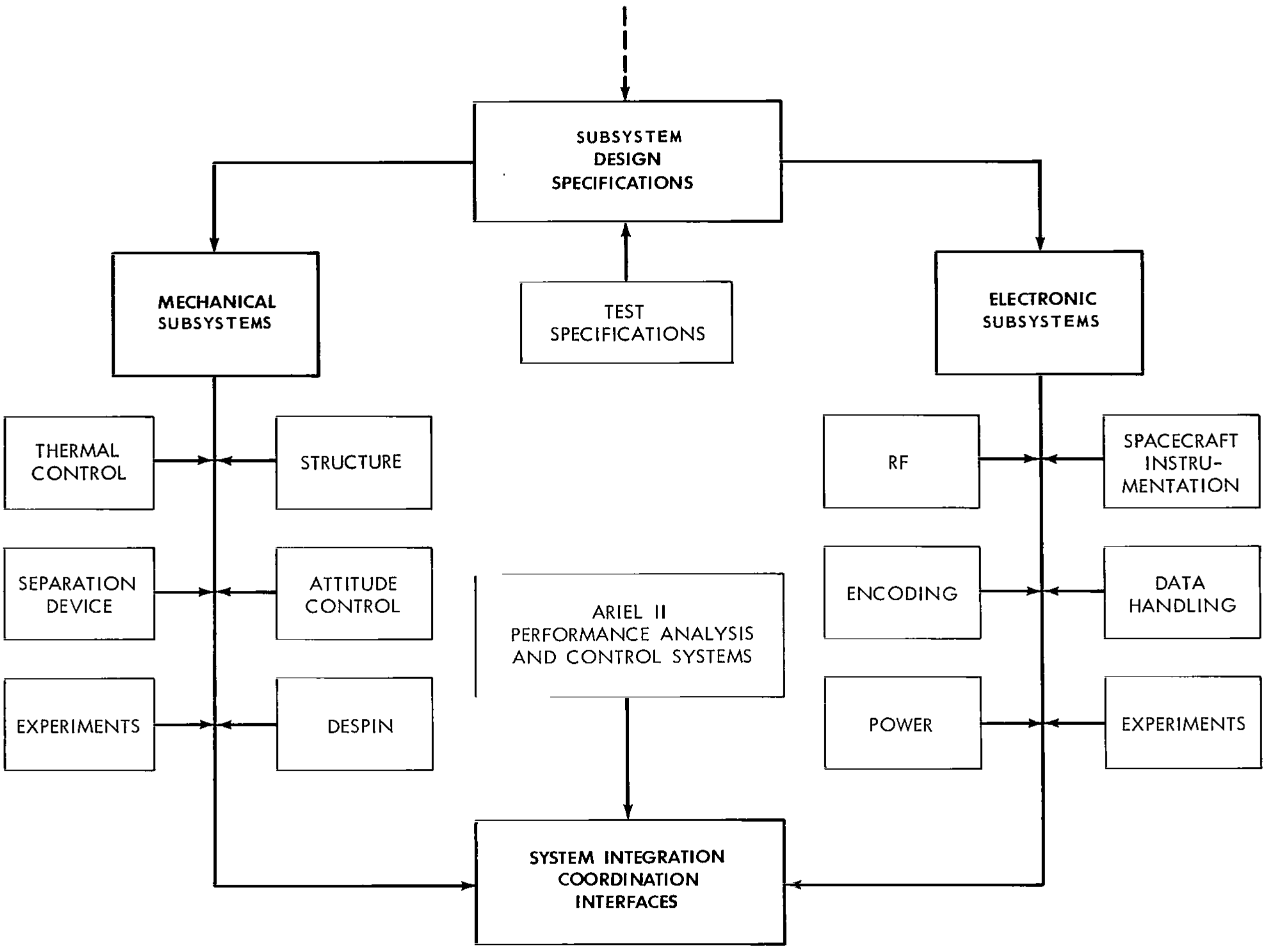

Figure 5—Generation of subsystem specifications.

scientific experimenters, all of these electronic subsystems were supplied by a spacecraft electronics branch, a flight data systems branch and a space power technology branch of GSFC, which also assisted in the detailed technical integration as required.

An integral part of the structural design was the consideration of thermal control. This subsystem was the responsibility of a thermal systems branch. Since the thermal control of Ariel II was of a passive nature, the spacecraft outer surface was designed to accept the desired paints or coatings. The mechanical designers assisted the experimenters and subsystem suppliers with any mechanical design problems they encountered. They designed special mechanisms and booms for the experimenters and pressurized containers for flight tape recorders. They assisted the power group by designing special cases and containers, heat sinks for converters and solar paddle structures. The electrical designers contributed by designing circuits for supplying power to all the electromechanical areas, such as release mechanisms and erection systems.

The electronic group comprised of the Project Management Staff and integration support personnel functioned as the electronic systems group for the project. Information contained in the system design specifications were furnished to all subsystem designers to assist them in the completion of their specifications. After the system design specifications were completed, the electronic systems group informed the experimenters with regard to the system philosophy and physical design, alerting them to possible system problems. The electronic group acquired detailed information with regard to experimenters' needs and desires. This helped to resolve individual problems at an early date. They assisted the experimenters and subsystem suppliers by furnishing them with design guidelines covering materials to be used, electronic fabrication practices, corona, noise, RF interference, induced magnetism problems, and preparation of electronic ground support equipment.

The subsystem design specifications for Ariel II were generated by the system engineers (Project Management Staff with contractor support) and subsystem design engineers (Appendices C and D). They were followed by detailed dimensional drawings, block diagrams and schematics. These specifications and drawings were reviewed on an as-produced basis by the Project Management Staff. At this point in project development, weekly meetings were held by project management to promote schedule compatibility of the subsystems. These meetings, attended by experimenters, subsystem designers, test engineers, contractor personnel and the staff, served as a clearing house for all compatibility problems for the duration of the project. Schedules for all subsystems of Ariel II were projected using the subsystem design specifications as a basis for the necessary development time. The schedules were reviewed and revised as necessary each week during these project status meetings. Schedule compatibility was aided through the assistance of a project support branch, GSFC, using a PERT chart. This chart was up-dated during the course of the project to reflect any anticipated slip-time due to late procurement of critical subsystems.

Interconnection Cabling Design Criteria

The interconnection system problems are best resolved as early as possible by management and system integration personnel interviewing each subsystem designer and experimenter, advising them of the spacecraft wiring reliability objectives and assisting them in the assignment of specific redundant leads and pin connections, complete with pin numbers. The desired redundancy of interconnections between subsystems must be decided during subsystem specification generation. Subsystem designers of the Ariel II designated the redundance desired at their interconnection points. Circuit redundancy within a subsystem was provided by the design engineer. A review of the subsystems for adequate parallel or back-up circuitry was conducted by the Project Management Staff.

The electrical interconnecting harness (or loom) of the Ariel II was designed with redundant wiring to provide a greater reliability factor. The general philosophy for the optimum redundancy throughout the cabling assembly was to provide redundant wiring for all critical signal and power leads. This resulted in approximately 70% of the conductors in the Ariel II being redundant. Also, at the connector interfaces where the probability of an "open" is greater since the path is through two solder joints and a pin-sleeve, spare pins were used when available to provide double redundancy. Test access to interfaces was incorporated by designating a spare pin in existing connectors, thereby eliminating external signal terminal boards from the wiring. Determination of which leads should be redundant and which should not was made by consultation between the subsystem design engineers, system integration engineers and the Project Management Staff. The first designation is usually made by the subsystem designer with respect to the leads he desires to be redundant. Subsequent review by project management is made to maintain uniformity and efficiency in the over-all system cabling for the desired redundancy that is practical in the spacecraft. Considerable weight may be added by redundant wiring. To reduce the weight and size of the Ariel II cabling, a Polyolefin insulation was used instead of Teflon. This resulted in a 60% weight saving and a 20% volume saving.

The umbilical plug circuitry and turn-on plug circuitry were finalized following completion of the subsystem design specifications. The Ariel II umbilical functions and monitoring points were held to nine. This provided adequate power and turn-on control of the spacecraft, a command function, and cyclic control of two timer functions. The Ariel II turn-on plug connected the main battery supply and solar power supply to the spacecraft load. Power to the Galactic Noise reel motor was connected through the turn-on plug. All of the leads through the turn-on, being critical, were made fully redundant leading to and through the turn-on connector.

Data Storage in Ariel II

A tape recorder was included in Ariel II to store data on a time-sharing basis from two experiments, the Galactic Noise and the Ozone experiments. The reason for the recorder inclusion was to relieve the real-time format of some of the data transmission and to obtain more complete coverage of the data throughout complete orbits. The recording time of the tape recorder is slightly greater than one orbital period. A playback command was initiated once per orbit for data retrieval. Playback time required 138 seconds nominally.

Test Specifications

Test specifications were generated by members of a test and evaluation division as soon as the subsystem specifications were completed (References 6, 7 and 8). The interface test parameters and limits were arrived at by conferences between the subsystem designers, integration engineers, test engineers, and Project Management Staff. Each interface of all subsystems was investigated for source and terminal impedance values to obtain compatibility with the test equipment complex. The number of test points requiring monitoring was determined on a subsystem basis first; then the test points were screened as to their priority to reduce the number of test leads in the spacecraft interconnecting harness to provide the desired over-all system monitoring. From the compatibility point of view, the fewer test leads in a spacecraft system the better, since any one could introduce detrimental crosstalk. From a test philosophy point of view, the greater number of monitoring points obtained the better. A compromise between these two viewpoints was made to acquire the least complex test instrumentation wiring and still provide adequate test monitoring for the spacecraft. Also, from the standpoint of reliability and weight, the instrumentation test leads were minimized.

Ariel II instrumentation consisted of 47 monitoring points. The critical waveforms of the encoder and experiment outputs were given top priority. The balance of the monitoring points were selected on the basis of necessity of observation during thermal environment tests. Many additional points throughout the spacecraft could have been monitored to obtain more complete "fingerprinting" of the subsystems, but the added wiring complexity necessary to provide them was considered a crosstalk risk to be avoided. Too many test leads in the spacecraft harness could be troublesome during the first thermal run, when crosstalk often appears, necessitating a test shutdown and lead modifications in the interconnection system.

Design and Fabrication of Support Equipment

While all of the subsystem suppliers were engaged in the detailed design and fabrication of their prototype and flight hardware, the integration groups designed and produced all of the necessary ground support equipment. The mechanical group designed all the jigs, fixtures, spacecraft handling dollies and shipping containers simultaneously with their design of the over-all structure. The electronic group, having acquired detailed information with regard to all the electronic subsystems, designed and produced the spacecraft control and spacecraft performance analysis systems prior to integration (Appendix E).

The spacecraft electronic control system furnished external power to the spacecraft during integration in the absence of solar paddles. It also furnished power, control, and monitoring to individual experiments being used to stimulate experiments, to change the mode of operation and to supply test pulses for experiment calibration. Certain special experiment test devices were contained in this system. The system was capable of decommutating the spacecraft data handling system without employing the RF system. The performance analysis system of the Ariel II Test Stand decommutated the spacecraft telemetry, processed the received data, and presented it in a readily usable form.

FABRICATION AND TEST OF STRUCTURES AND SUBSYSTEMS

The fabrication of the Ariel II spacecraft was started as soon as the subsystem specifications and drawings were completed. There were changes and modifications to both the specifications and drawings as the fabrications proceeded. For control, schedule, and compatibility of interfaces, the Project Management Staff reviewed and approved all modifications or changes proposed prior to their inclusion.

For structural test purposes, two units were fabricated. The first was a dynamic test unit which simulated the weight, size, and center of gravity of the spacecraft and had all attaching appendages necessary for the dynamic separation and despin tests. The second unit constructed was an engineering test unit which closely simulated the detailed structure of the completed spacecraft. Dummy subsystems and experiments inside the spacecraft were used for weight and center of gravity simulation. The skins, frame appendages, and separation system were exact duplications of those to be used in the actual spacecraft. This unit was employed for vibration testing. Several minor modifications to the structure were necessary to correct defects which appeared during tests of the engineering test unit. One major modification was necessary to provide additional support for the Ozone Broadband experiment on the top of the spacecraft. Final vibration tests were made to prove the structural adequacy of the modified engineering test unit for the qualification levels of the Scout fourth stage (X248). Fabrication and test of these two units assured compatibility of the mechanical design for the launch environment on a Scout vehicle.

The prototype structural assembly and the interconnection harness fabrication was started at the contractor's plant following successful testing of the DTU. Fabrication of the interconnection cabling system for the prototype was almost completed when reports were received of a tendency of cadmium surfaces to provide fine whisker growths under vacuum exposure. As a result, it was necessary to delay fabrication of flight model interconnection systems for some weeks pending receipt of gold-plated connectors. The prototype was completed using cadmium connectors. No difficulties during tests could be attributed to their use.

The prototype experiments and subsystems packages of Ariel II were fabricated simultaneously with the prototype structure and interconnection system. Since these packages were derived from three sources (the contractor, United Kingdom, and GSFC), considerable scheduling difficulties arose when attempting to dovetail their *ready date* with that of the prototype structure for integration. Subsystems which were to some extent repetitive in design (such as Ariel II's transmitter, receiver, converters and flight tape recorder) did not pose a difficult scheduling problem. The experiments, however, being all of original design, required a greater effort to maintain the same time schedule during fabrication and test. The two programmers and three encoder packages also required additional effort for schedule harmony. The schedule status of the subsystems was reviewed each week by project management and projected dates for their completion were determined by consultation with the designers. The current and projected status of the subsystems was documented by the project management weekly report and distributed to all cognizant project personnel. This served to inform them of the need for expediting the design, fabrication, or test of any subsystems lagging behind the over-all schedule.

Subsystems tests for Ariel II consisted of two types, design qualification or prototype level tests and flight acceptance level tests. Subsystems fabricated for the prototype spacecraft or engineering test units fabricated for design approval were subjected to higher test levels than those subsystems fabricated for the flight spacecraft. Design qualification vibration levels were 1.5 times the measured launch levels; flight acceptance levels were 1.10. Design qualification temperature levels were established at +60°C upper and -15°C lower limits. The flight acceptance levels were +50°C and -5°C. Due to changes in the predicted temperatures, the final lower limit of the flight acceptance levels was reduced to -15°C.

Whenever delivery schedules permitted, interdependent subsystems were interconnected as a partial system during tests. This provided a close check on the compatibility of the assemblies during test conditions which more nearly duplicated the interconnecting complex of the completed spacecraft. For example, the following subsystems of the Flight I spacecraft were interconnected as a partial system during certain environmental flight acceptance tests (vibration, thermal and thermal vacuum):

Transmitter	Command Receiver
Decoder	Data Storage Card
Programmer #1	Undervoltage Detector
Tape Recorder	Tape Recorder Converter

During the development of the Ariel II subsystems, numerous modifications were necessary to secure the desired circuitry performance during environmental testing. Without adequate control via project management, these modifications would have been a source of severe incompatibility. Uniformity of circuitry in the Prototype, Flight I and Flight II subsystems had to be rigidly maintained to avoid this compatibility problem. Whenever a modification was necessary in a subsystem, all of the completed assemblies of that type had to be modified similarly as soon as the work schedule permitted. Modification of a subsystem after qualification resulted in decreased confidence of the subsystem due to the necessary rework and retest. Any failure during subsystem development is cause for alarm. Immediate investigation must be made to discover whether or not it is due to human error or failure of a component during normal test conditions. Careful analysis of all component failures were made and documented throughout the project development. The current GSFC philosophy holds that the words "random failure" are meaningless and misleading and that every failure must be analyzed and explained.

SYSTEM INTEGRATION

After all the subsystem and experiment hardware had been produced and qualified, it was furnished to the project office for the detailed technical integration. The over-all system integration, coordination, and solution of all interface problems was accomplished through the joint effort of the Project Management Staff and the mechanical and electronic integration groups. The output of the physical integration effort was the spacecraft logs, spacecraft data, and the system status documents (Figure 6). With the completion of spacecraft integration, telemetry test tapes were furnished to the tracking, data acquisition, and data reduction technical personnel for compatibility checks.

The integration of Ariel II was accomplished at the contractor under the direct supervision of the GSFC project management. A compatibility test layout board was constructed at the contractor (Figure 7). This test fixture contained terminals for measurement of all interfaces within the interconnection system. Complete spacecraft systems or subsystems were inserted in this test complex prior to physical integration in the actual spacecraft. A detailed integration plan was

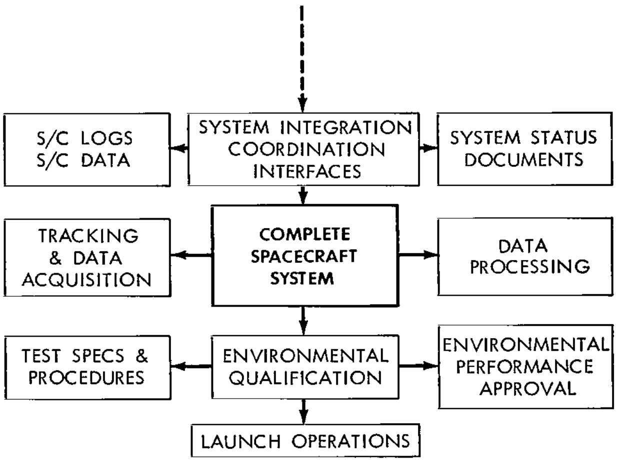

Figure 6—Integration, environmental qualification and launch operations.

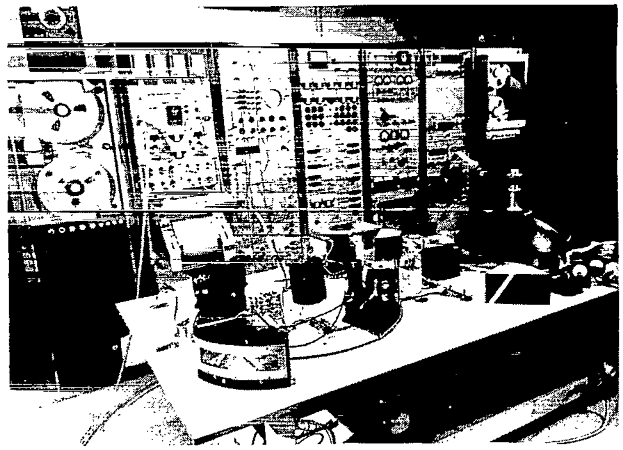

Figure 7—Compatibility test layout board.

drafted by the contractor. This was reviewed and approved by GSFC project management. The successive phases of the integration plan were:

Subsystem Functional Test
Limited Compatibility Test
Complete System Compatibility Test
Prototype Integration
Flight I Integration
Flight II Integration
Sunlight Tests of Prototype and Flight Units

The Ariel II test stand shown in Figure 8 was assembled by the contractor under supervision by GSFC project management. The test stand included the necessary controls, power supplies and connectors to test all of the electronic subsystems individually. It also contained a telemetry receiver, data reduction system and a single-channel printout for the over-all checkout of the Ariel II spacecraft. All of the electronic subsystems and the experiments were individually checked by the test stand prior to further integration steps. The values and tolerances of the parameters listed for the checks were obtained from the subsystem specifications. Any subsystem whose parameters were outside the test specification level was removed from integration and returned to the designer for retest and analysis of the tolerance error. All parameters measured were documented for each subsystem. This provided a test history which was used as a reference for any slight tolerance variations that occurred as a result of aging during subsequent tests.

Limited compatibility tests were made using related subsystems to establish partial system compatibility prior to complete systems tests. The compatibility test layout board was employed for the first over-all system test. All subsystems and experiments were required to be fully

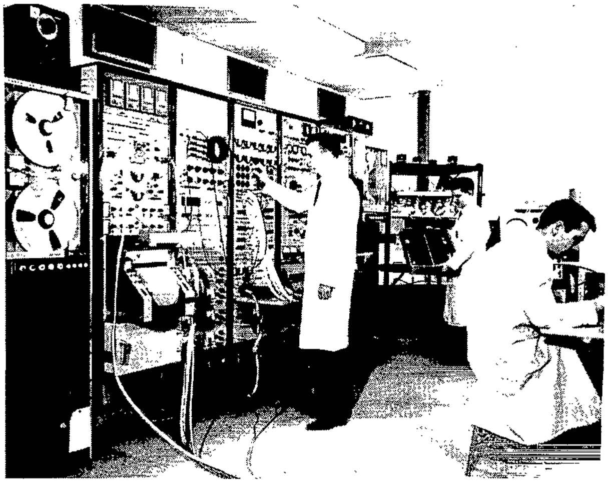

Figure 8—Ariel II test stand.

compatible on the layout board before integration into the spacecraft. All modifications, additions, or subsystem component changes were accomplished to provide compatibility between the systems. Ariel II Prototype required eight major modifications and six minor modifications. The eight major modifications were:

1. Staticizer subsystems were added to stabilize the data rate during the data sampling time. This was necessary to prevent the high data rate of the Ozone and Galactic noise experiments from causing dropouts during data reduction.

2. Pre-amps were designed and added to the IROD assembly of the Micrometeoroid experiment. This was necessary to correct the instability and low gain of the amplifiers supplied with the experiment.

3. The encoder oscillators exhibited frequency shifts with changes in orientation in the earth's magnetic field. A low-level degausser was designed to reduce the residual moment of the nickel hookup leads in the encoders. After the degaussing treatment, the oscillator shifts due to random orientation in the earth's magnetic field were not significant.

4. Ozone light baffles were designed and installed as light separators between the Ozone spectrometers. This was necessary to prevent the observed crosstalk between spectrometers caused by reflections. Four baffles per spacecraft were required.

5. Crosstalk between the Galactic Noise experiment and the Micrometeoroid experiment via the common +12 volt supply was detected. As a remedy, voltage stabilizers were designed and added to stabilize the +12 volt input to the Micrometeoroid instantaneous readout detectors (IRODS).

6. The capacitor units supplying the foil advance pulse power for the ledexes were leaky, causing intermittent foil advance. An improved type of capacitor was procured by the experimenter as a remedy.

7. The trigger and selector units of the micrometeoroid experiment produced double triggers when a foil advance pulse occurred. A 47K resistor was installed to isolate the output of the two units.

8. During prototype environmental testing, several failures of a particular type of resistor used in the Ozone electronic packages occurred. A search of the test history of all packages from the time of manufacture disclosed additional failures of a similar nature. As a result, all existing Ozone electronic flight unit packages were reworked to include Mil spec resistors.

The six minor modifications were:

1. The IROD and DROD (delayed readout detector) units of the Micrometeoroid experiment had light leaks disabling their operation. Light-sealing techniques were developed by the UK experimenters at GSFC to solve this problem.
2. The Encoder No. 1 required modification to obtain symmetry of the divide by 48 low speed tape data storage envelope.
3. The aluminum foil of the IROD units in the micrometeoroid developed severe creases which caused non-advance or tearing during foil advance operation. The units were removed and mechanically reworked to obtain proper clearance between the foil and the light-tight baffle plate.
4. A GSFC-designed Programmer No. 2 replaced a contractor's Programmer No. 2 after several failures of the unit occurred. GSFC programmers were used in the final spacecraft configuration.
5. The Data Storage Control Unit was removed for replacement of the Sprague C-109 capacitors with Sprague C-150D capacitors. This change was made due to failure of C-109 capacitors during the separation timer vibration tests.
6. The playback time of the tape recorder was several seconds short of the desired minimum time. The data storage unit was removed and adjusted for the nominal playback time.

All of these changes were expedited to secure a minimum schedule slip during integration.

The over-all satellite integration was started as soon as system compatibility on the test layout board was achieved. The subsystems were inserted in the spacecraft as related partial systems. These systems were checked prior to the insertion of additional related partial systems. The final check was an over-all test with all of the spacecraft subsystems and experiments installed. The system test of the completed spacecraft was made using the Ariel II test stand. Complete documentation was maintained during the final integration. It consisted of daily logs of work accomplished, tests performed, and failures experienced. Data sheets were maintained on the parameters and tolerances of each subsystem. All data sheets relating to a particular spacecraft were maintained in a separate file.

As a compatibility test for operation in sunlight, the Prototype and Flight spacecrafts were placed on a rotator and rotated at 5 rpm, the nominal orbital spin rate (Figure 9). This test proved the capability of the solar power supply and the operation of the two experiments excited by sunlight—the Micrometeoroid and the Ozone. The compatibility of all subsystems with the solar power rotational perturbation was checked.

GSFC project personnel observed all integration tests at the contractor and reviewed all documentary data. A final series of systems tests performed on the spacecraft were witnessed by GSFC personnel and the experimenters to determine the readiness for environmental testing. After the final inspection, the satellite and ground-handling equipment were transported to GSFC for experiment calibration and environmental tests.

ENVIRONMENTAL QUALIFICATION

The environmental tests of the Ariel II satellites were conducted to evaluate their performance when exposed to launch and orbital environments (References 9 and 10). The prototype was subjected to test levels above the required levels to prove the desired overrate factor. Table 3 lists the types of tests and calibrations performed on the Ariel II prototype and flight spacecraft. Continuous monitoring of test points throughout the satellite is desirable during any of the above tests; however, too many hard-line connections to the spacecraft may introduce noise and short duration anomolies. For each type of test, the optimum instrumentation must be ascertained. The final test plans were derived by consultation between the Project Management Staff and the Project Test Manager.

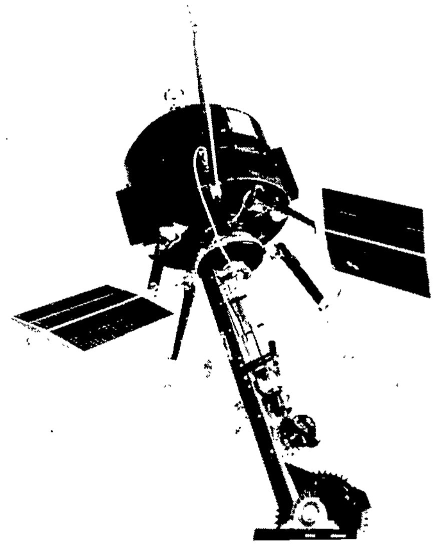

Figure 9—Sunlight test rotating mechanism.

Experiment Calibration

Experiment calibrations of the Ariel II spacecraft were conducted before and after any type of test which the experimenter had reason to believe would cause a calibration shift. Two subsystems of the Ozone experiment on Ariel II required these calibrations. Documentations of the calibrations were used by the experimenter for selection of the optimum performance subassemblies and reference compatibility of the experiment calibration stability.

Vibration Tests

The Ariel II test instrumentation was kept at a minimum during vibration. Only RF telemetry readout was used when vibration tests were under way. During these vibration runs some temporary dropouts occurred. These dropouts or anomolies were recorded and later analyzed to determine whether induced by faulty connections, faulty components or structural failure. A check was made to insure that immediate recovery was established at termination of the test. Some of the vibration tables that do not have

Table 3

Tests and Calibrations Performed on the Ariel II Prototype and Flight Spacecraft.

Prototype	Flight Units
Antenna Pattern Test	Antenna Pattern Test
Experiment Calibration	Experiment Calibration
Vibration	Vibration
Experiment Calibration	Experiment Calibration
Temperature and Humidity	Temperature
Acceleration	Thermal Vacuum
Thermal Vacuum	Experiment Calibration
Experiment Calibration	Antenna Pattern Test
Solar Simulation	
Experiment Calibration	
Spin-up and Separation	
Antenna Pattern Test	

degaussing coils for deperming may have permanent magnetic fields in the order of several gauss. This may affect relay operation, or, in the case of oscillators, a step frequency shift. Any subsystems containing components affected by perm fields of two gauss or less should be vibration-tested on tables equipped for deperming. Figure 10 shows the prototype under preparation for the vertical vibration test.

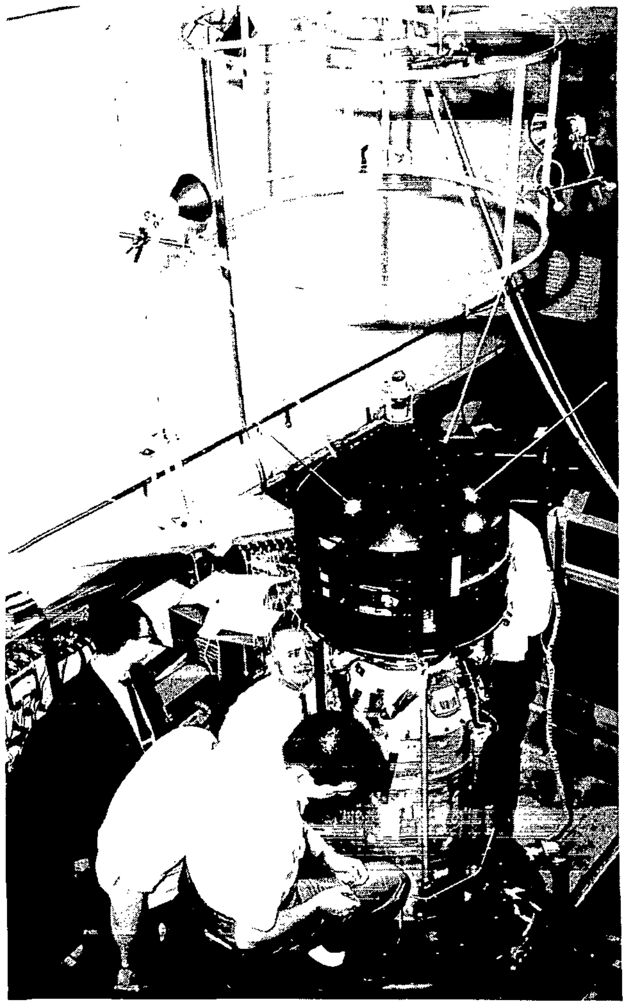

Figure 10—Ariel II vertical vibration test preparation.

Temperature Tests

A temperature test of Ariel II was employed to discover any system incompatibility problems that might arise due to temperature gradients or excursions prior to thermal vacuum testing. The prototype disclosed a number of malfunctions during its first temperature run of +60 degrees C to -15 degrees C. If these failures had occurred during the first thermal vacuum run, considerable time would have been consumed restoring chamber pressure to permit disassembly and investigation. A five-week delay in the prototype test schedule was incurred at this time during which mechanical redesigns and component changes were made to correct the malfunctions appearing in the first thermal run. A short second thermal run was made after the modifications to verify system performance. To prevent time consuming shutdowns during thermal vacuum testing, a thermal test of a completed spacecraft should always precede the thermal vacuum test.

Thermal Vacuum Tests

Thermal vacuum tests of satellites simulate orbital environments to some degree (Reference 11). For a true compatibility test of the spacecraft operation in orbit, a number of test conditions are missing. Among these are: solar radiation, earth albedo, the hard vacuum of space (10^{-12} mm), thermal gradients, and solar power. The Ariel II prototype was subjected to a short solar simulation test in a vacuum chamber. Some significant data was obtained for temperature predictions from this test; however, the environmental control during the test was considered too

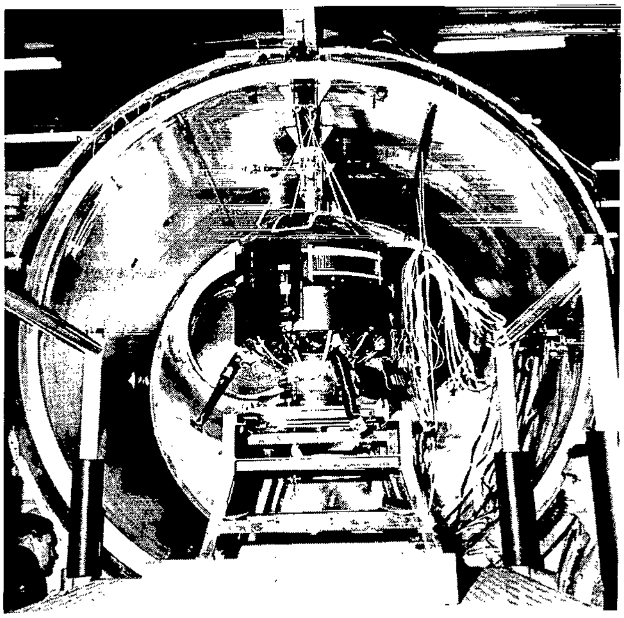

Figure 11—Insertion of the prototype into a thermal vacuum chamber.

hazardous at the time for flight units. Figure 11 shows the prototype being inserted into an 8' × 8' thermal vacuum chamber.

The duration of thermal vacuum tests was extended to obtain the greatest confidence level possible for satellite operation in an orbital environment. Ariel II Flight Unit No. 1 was subjected to three thermal vacuum test periods. The duration of the first was 336 hours (14 days), the second, 96 hours (4 days) and the third, 120 hours (5 days). This totaled 23 days of thermal vacuum testing. The second thermal vacuum test was a retest after replacement of the Micrometeoroid experiment; the third test was the final thermal vacuum run just prior to launch. It was considered necessary following a six month delay of launch due to vehicle difficulties and replacement of the Ozone experiment with recalibrated units. Extended duration thermal vacuum tests are desirable for achieving reliability confidence levels. The amount of time for test is often dictated by the over-all project schedule; in general, the longer the better.

Excitation of the experiments on Ariel II during thermal vacuum testing posed many problems. Two of the experiments, ozone and micrometeoroid measurements, are excited by sunlight which is scanned by rotation of the spacecraft in orbit. During thermal vacuum testing, the spacecraft was mounted on a pedestal and remained stationary throughout the test. An aluminum frame was fabricated, to which lamps were attached for a step-level excitation of these experiment sensors. The lamps employed for this presented some hazard in the event of their explosion during test and excessive heating of the skin of the spacecraft. One of the lamps exploded during the prototype thermal vacuum testing. The test was shut down following this incident and screen shields placed over the lamps. No additional explosions occurred during the balance of the prototype tests.

During the Flight I thermal vacuum tests all excitation lamps were kept exterior to the chamber. Only those sensors were excited which could be placed in front of the chamber portholes and periodically excited by manually held lamps. In addition, an ultra-violet lamp assembly was mounted in front of one porthole and scanned across one of the ozone spectrometer sensors. This did not provide a satisfactory spectrum but was employed as a relative excitation level test. Since calibrated excitation of the sensors during thermal vacuum testing was not practical, the amplifiers of the micrometeoroid and ozone experiments were checked via pulse inputs inserted at the front end of the amplifiers during Flight I and Flight II tests. This by-passed the actual sun sensors and although it provided good test data on the amplifiers' behavior during the thermal vacuum run, the test data on the experiments was only a "Go - No Go" type. The ozone experimenter had by this time enough test history from the prototype and subsystems tests to have confidence in the sensors' performance in the thermal vacuum environment.

Calibration curves for the 16 performance parameters were made during the thermal vacuum test periods. The temperature performance parameter curves were cross-checked with the original temperature sensor calibration and also compared to additional sensors mounted in the spacecraft near the onboard sensors. Calibration of the current and voltage parameters was maintained by the periodic standard calibration of all instruments in the test stand. The performance parameter curves which evolved during the Flight I thermal vacuum tests were used for in-flight performance data reduction. Documentation during the thermal vacuum testing was maintained on a round-the-clock basis. A log of all tests, conditions, failures, or temporary operational anomalies was maintained for each spacecraft. A data performance sheet was maintained on all of the test point parameters and calibration parameters of the spacecraft.

Whenever a temporary or continuing abnormality in performance occurred, an analysis of the cause was made as soon as possible by consultation between the electronic test team, Project Management Staff, and the appropriate subsystem designer. The chronological log and performance data sheets were used as a reference during this analysis. Diagnosis of the cause of every performance anomaly must be expedited in order to arrive at a decision as to whether the test should be altered, continued, or shut down. Infallible intuition would be a good tool for project managers when contemplating this decision, but since this is nonexistent, accurate test data documentation for reference is the next best thing to use.

During the vibration and thermal vacuum test periods, numerous complete or partial disassemblies were necessary to permit subsystem changes, modifications, repairs, or inspections. Every partial or complete disassembly compromises the confidence level of the spacecraft reliability. The number and depth of disassemblies of the Ariel II spacecraft were closely documented. There were 22 major disassemblies of the prototype. Major disassemblies of Flight I were held to five. All of the disassemblies and assemblies were accomplished by the GSFC and contractor mechanical integration personnel. The number of technicians performing the mechanical assemblies was kept to a minimum to reduce the human-error factor. GSFC mechanical engineers were in charge of all assembly, disassembly, erection, loading, or transporting activities.

FINAL TESTS AND MEASUREMENTS

Following the environmental tests, the prototype was utilized in a dynamic spin-up and separation test in vacuum to check the compatibility of the flight separation system with a completed spacecraft (Figures 12 and 13). Flight spacecraft were not used in this test due to the hazards presented in the extensive handling and transporting involved. A telemetry compatibility test was conducted at the Blossom Point, Maryland STADAN station. The Ariel II prototype was operated in all modes including playbacks. Real-time data reduction with the station receivers connected

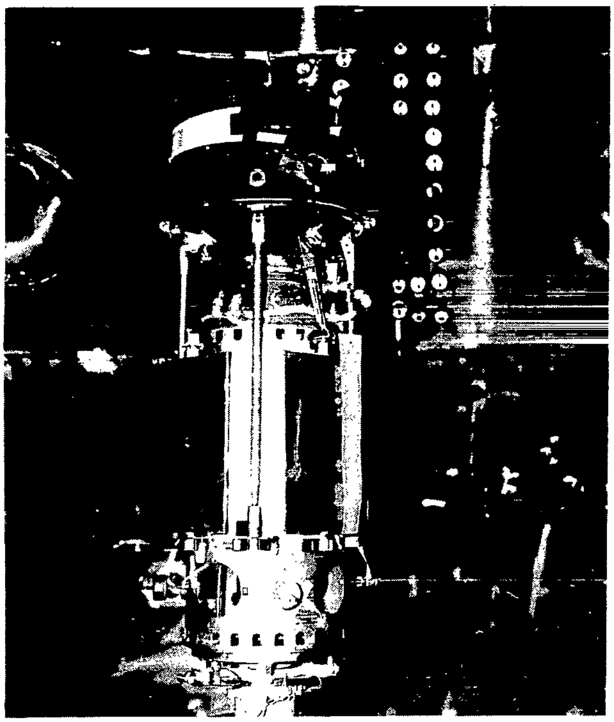

Figure 12—Ariel II prototype prior to dynamic spin-up and separation test in a 35-foot vacuum chamber.

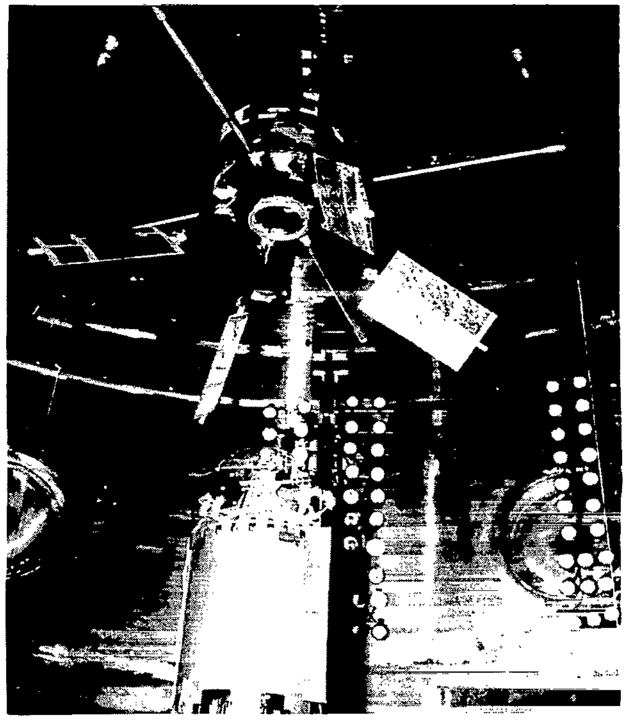

Figure 13—Ariel II prototype at termination of dynamic spin-up and separation test.

to the Ariel II test stand was performed and recordings of data were made to verify the compatibility of the Ariel II telemetry and Blossom Point data acquisition.

The prototype and Flight I spacecraft had the following measurements made after environmental tests:

Solar Paddle Alignment
Dynamic Balance
Moment of Inertia Measurements
Magnetic Configuration Survey
Magnetic Perm Compensation
Magnetic Induced Equalization
Antenna Pattern Check

Although the prototype was not intended for flight, it was used as a first practice run for these measurements, which concluded the tests and checks of the spacecraft prior to deployment for launch. During all handling, set-up and transporting of the spacecraft, a stripcoat paint was used as a protective covering for the passive temperature control finish. It was removed only during the thermal vacuum and solar simulation tests.

LAUNCH PREPARATIONS

Prior to the launch preparations, periodic conferences between the Scout vehicle project personnel and the Ariel II project personnel were held to insure compatibility of the spacecraft requirements with that of the vehicle requirements (Figure 14). During the conference visits, the facilities to be used for launch preparations were inspected by the Project Management Staff. All necessary alterations and modifications to the facilities or work areas were made well in advance of the launch schedule. Six months prior to the launch date, the Project Management Staff prepared a payload description document which contained all the launch operation requirements (Reference 12). It was prepared primarily for the Scout project group.

The compatibility of the spacecraft versus vehicle schedules during launch preparation and launch activities was dependent on the payload description document accuracy. This document contained a description of the spacecraft, its mission and technical data, project personnel roster and all special requirements for tracking during the launch phase. Any changes made in this document after its distribution required immediate notification to all the recipients to prevent schedule interface problems during launch preparations.

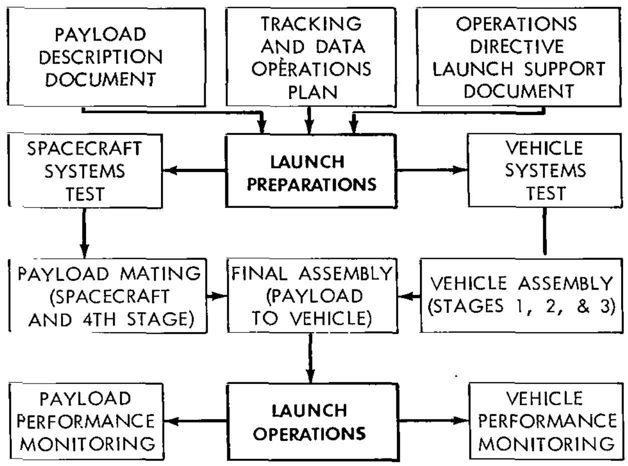

Figure 14—Launch preparations, final assembly and launch operations.

An operations directive for the launch of Ariel II at Wallops Station was published and distributed by the Scout project office of the Langley Research Center (Reference 13). This document directed the support of the launch tracking stations and stated the vehicle launch requirements for the desired orbital parameters. The communication net requirements, meteorological support, range time utilization, radar and telemetry tracking stations support and special material and services required for the launch were also included. Specific requirements and directives for the launch effort were forwarded to the Vehicle Test and Operations Group at Wallops Station. Range Control was notified by the operations directive as to the time and duration (window) a cleared range would be required. Directly prior to the spacecraft deployment for launch, the following pre-launch schedules were made as firm as inputs at the time would permit:

Vehicle Systems Tests Schedule—This schedule, prepared by LRC and Wallops Station, detailed the step-by-step procedure and test equipment to be used for inspection and calibration of the Scout vehicle mechanical and electrical systems and subsystems.

Payload Systems Test Schedule—This schedule, prepared by the Project Management Staff, provided for systems tests of the Ariel II spacecrafts at Wallops Station to be conducted during the time scheduled for vehicle systems tests. An over-all systems test, spacecraft-vehicle RF interference test, antenna pattern check, and a sunlight rotation test were performed. These were the last tests to prove compatibility prior to mounting the spacecraft on the 4th stage of the Scout vehicle.

Vehicle Assembly Schedule by LRC and Wallops Station—This schedule, prepared by LRC and Wallops Station, itemized the events and performance steps necessary to erect the first, second and third stages of the Scout vehicle on the launch pad. The schedule also included the time required for erection and inspection.

Payload Mating Schedule—This schedule, prepared jointly by the Ariel II Project Management Staff and the Wallops Station Scout Launch Team, detailed the procedures and equipment to be used for mating the spacecraft to the Scout 4th stage. This included physical mating of the spacecraft to the 4th stage mounting ring, dynamic balance of the mated assemblies, and installation of the nose fairing or heat shield. It dictated that the time for completion of this work should coincide with the completion of the vehicle assembly on the launch pad. The final assembly step in the schedule was the mating of the 4th stage and payload assembly to the 3rd stage of the erected Scout vehicle.

Countdown Schedule—This schedule was prepared jointly by the Scout Launch Director and the Ariel II Project Management Staff. The initial countdown was prepared by inserting the payload requirements in proper time sequence with the established Scout vehicle countdown. A number of changes were made in the initial countdown during the launch preparation events. Each change was reviewed and approved by both Ariel II Project Manager and the Scout Launch Director before being inserted in the countdown.

LAUNCH OPERATIONS

To provide a practice run of all launch operation events, the Ariel II prototype was transported to Wallops Station. The practice launch operations included vehicle-payload compatibility checks, payload mating and spin balance on a dummy 4th stage as in Figure 15, nose fairing installation around the mated payload and dummy 4th stage, transferring the assembly to the gantry, and simulating erection of the completed assembly atop the vehicle. A platform installed at the payload level was used to simulate the Scout vehicle 3rd stage interface during the erection of the Prototype assembly at the gantry. The practice run proved to be very informative. Deficiencies of procedures and facilities were discovered during this simulated launch operation. All of the personnel accomplishing the testing, mating, balance, and assembly of the prototype became sufficiently experienced in their procedures that during the launch operations of the Flight I spacecraft, all events went smoothly. The practice launch operation was accomplished three days prior to the launch operations of the Flight I spacecraft.

The schedule followed for the Scout vehicle and Ariel II Flight I is given in Table 4. After installation of the Ariel II payload on the Scout Vehicle, operational control was established in the blockhouse. Compatibility checks of the spacecraft, vehicle telemetry and the tracking radars were made prior to the final countdown. The final system test of Ariel II installed on the launch vehicle required removal of the nose fairing to permit final excitation of the experiments by simulated sunlight according to a systems test procedure prepared by the Project Management Staff. An action item list of all final connections and inspections was prepared to insure that the critical final steps were performed prior to reinstallation of the nose fairing. Each item on the list was initialed at the time it was performed or observed on the gantry.

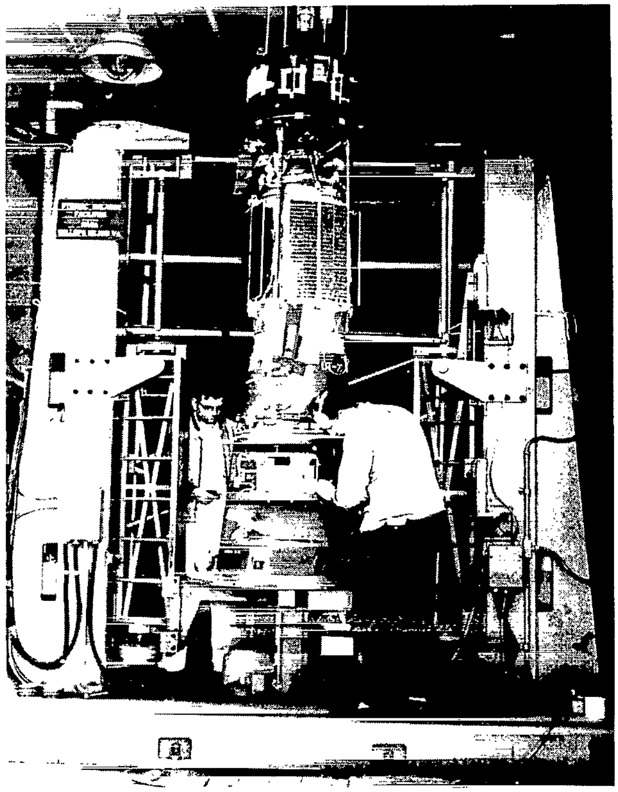

Figure 15—Dynamic balancing of the Ariel II Flight I spacecraft at Wallops Station.

The ground stations were used to monitor the Ariel II telemetry. One station was located on the main base in the hangar area. The other station was located in a van one mile south of the launch pad. These two stations provided redundancy in the event of a failure and permitted a continuous cross-check on data readouts whenever the spacecraft was operated. One communications channel was reserved for payload use. The Blockhouse Payload Control, Range Control Central, and both ground stations were on the payload net. A countdown rehearsal was conducted two days prior to launch. All stations were manned and all personnel were in attendance to fully simulate the true countdown conditions. The rehearsal revealed a number of minor procedural errors which were corrected during the post-rehearsal critique.

During the final countdown, which was completed in eight hours, one compatibility problem appeared. The galactic noise experiment had a hi-level interference signal appear during a turn-on test in the early part of the countdown. A check of the immediate area revealed an atmospheric sounder transmitter operating about one mile from the launch site. The hi-level interference disappeared from the experiment output as soon as the sounder was turned off. During the terminal countdown period (last 30 minutes), the spacecraft was operated at all times except for the time required for vehicle arming. Evaluation of the telemetry readout was continuous at both ground stations. The evaluation of the experiment data was made by experimenters located at the ground stations. Evaluation of the housekeeping functions (performance parameters) was made by the GSFC Ariel II electronics test teams at both stations. Comparisons of the readouts obtained were

Table 4

Launch Operations Schedule for the Scout Vehicle and Ariel II Flight I.

Days Required	Ariel II	Scout Vehicle
15	S/c Systems test	Vehicle systems tests
4	S/c mounted on 4th stage, assembly balanced, heat shield installed	First, second and third stages erected on launch pad, operations check and environmental control check
1	Completed assembly erected on Scout 3rd stage	Assembly inspection

made as requested by the Project Manager in Range Control Central or by the Payload Control Engineer in the blockhouse. At T-2 minutes, the spacecraft was switched from external to internal power via the blockhouse payload console. The telemetry evaluation at the main base ground station continued from lift-off to T + 15 minutes. Figure 16 depicts the Ariel II launch.

POST LAUNCH ACTIVITIES

After the spacecraft was successfully launched, it was tracked by 12 STADAN stations to establish the orbit and acquire telemetry data according to an operations plan generated by the Office of Tracking and Data Systems, GSFC. This plan included obtaining tracking data during the launch phase as well as during the orbital life of Ariel II. The data acquired was recorded on magnetic tape and sent to the data reduction facility for processing (Figure 17). The orbital data was reduced by GSFC. All telemetry data was shipped to the United Kingdom in raw form for data reduction and processing by the Radio Research Station, Slough, England prior to being sent to the individual experimenters for analysis. Orbital information was furnished the

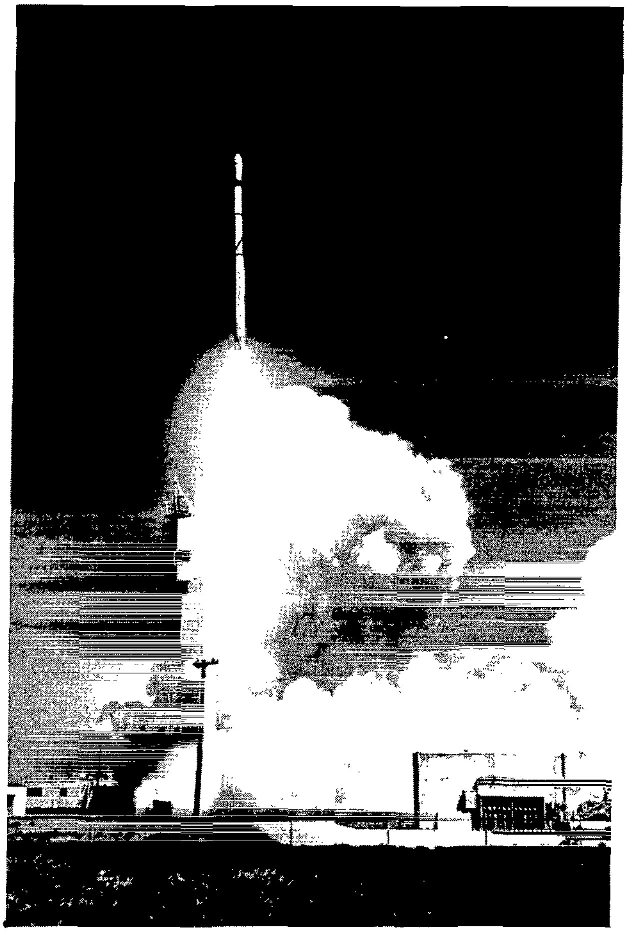

Figure 16—Launch of Ariel II at Wallops Station, Virginia.

experimenters by GSFC. Upon completion of the data analysis, it will be distributed to the scientific community through symposia and scientific journals. The telemetry data acquired by the Blossom Point, Maryland STADAN station was reduced at GSFC to enable the Project Management Staff to monitor the performance parameters in order to determine the status of the Ariel II on a daily basis.

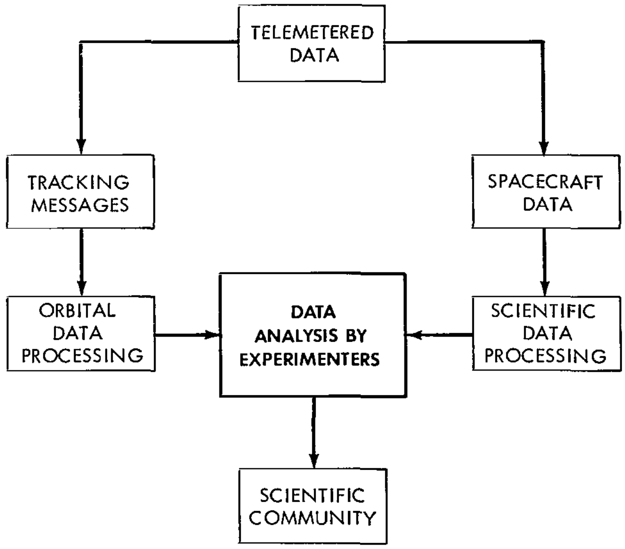

Figure 17—Data acquisition, reduction, and analysis.

CONCLUDING REMARKS

The responsibility for the system compatibility of a completed spacecraft rests with project management. Achievement of this compatibility is accomplished by the combined efforts of all personnel assigned to the project. Compatibility must be achieved on a continuous basis as each detailed design, fabrication, integration, and test event occurs during the course of the project.

The first major milestone for Ariel II was reached at the successful completion of prototype integration. The subsequent environmental tests and special measurements were accomplished to verify the completed spacecraft performance in the simulated orbital environment. Close surveillance and documentation of the spacecraft configuration and system parameters were maintained for design continuity and control throughout the project.

Since the final proof of over-all system compatibility is demonstrated by the satellite's performance in orbit, a continuous effort must be expended with regard to the design and development of the spacecraft from conception to completion. The successful performance of Ariel II during its first seven months in orbit is the most significant measure available regarding the achievement of design compatibility.

(Manuscript received June 4, 1965)

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APPENDICES

The following five appendices have been lifted out of context from the International Satellite UK-2/S-52 Project Development Plan and the UK-2/S-52 Handbook prepared by Westinghouse (Reference 4). They are included to provide more detailed information with regard to project background and content.

List of Appendices

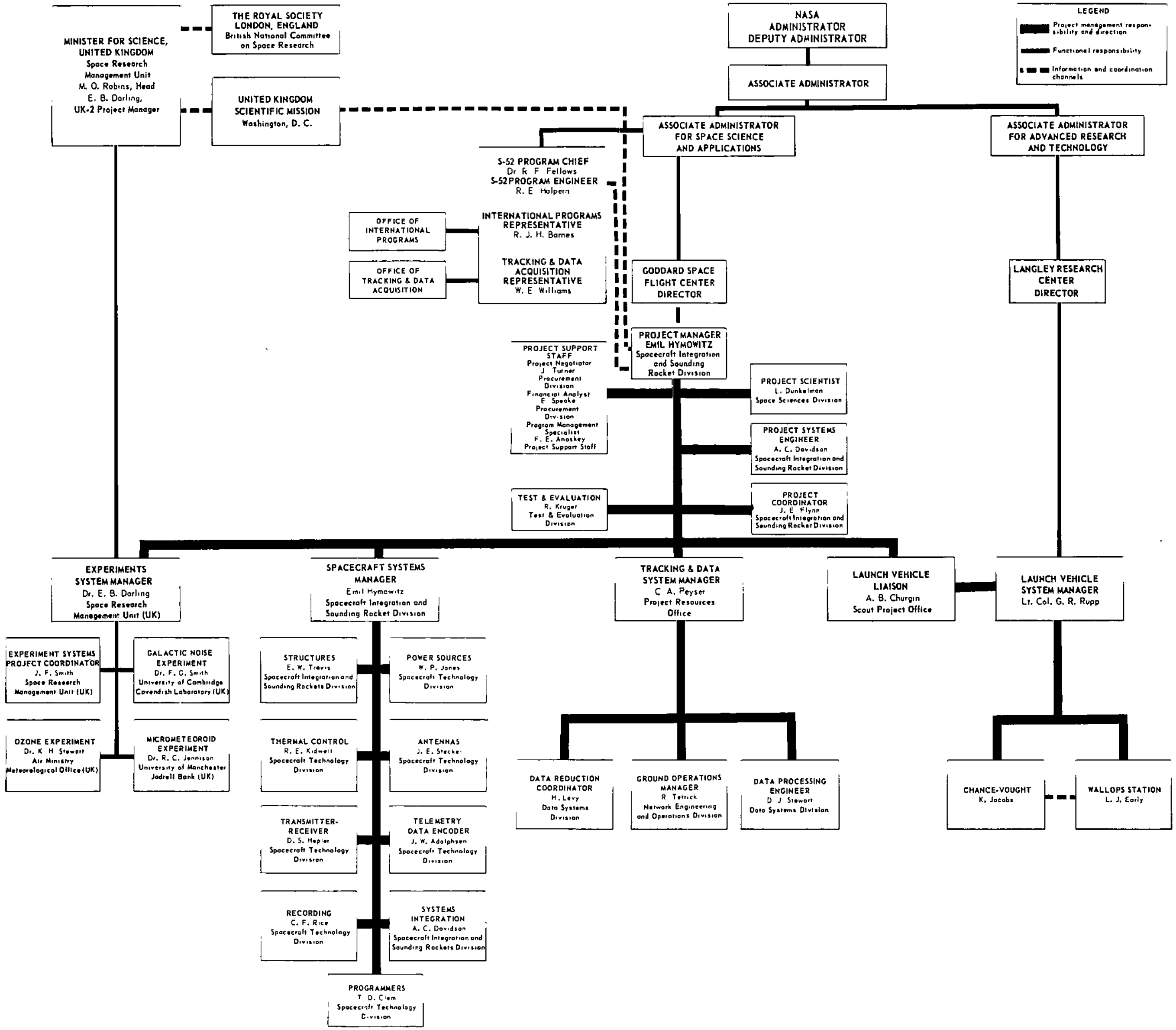


Figure A-1—UK-2/S-52 Project organization chart.

Appendix B

UK-2/2-52 Data Sheet

SPACECRAFT

Weight --	165 pounds, including 15 pounds for the separation mechanism
Size --	UK-2/S-52 body is 23 inches outside diameter at center, by approximately 35 inches long, not including solar paddles. After injection into orbit, a 130-foot dipole antenna is erected normal to the spin axis. Paddles extend 30 inches below S-52 base in tie-down condition.
Power Subsystems --	
Required:	Approximately 6.3 watts during shadow and 8.08 watts in sunlight.
Supply:	Rechargeable NiCad batteries, plus approximately 5400 N/P solar cells mounted in four fixed paddle arrays. Solar arrays furnish approximately 21.0 watts of power for a 60° sun angle at launch.
Telemetry and Tracking Subsystems --	
Data acquisition:	Continuous real-time transmission of galactic-noise experiment and micrometeoroid experiment in the Mode Number I or normal condition. The exceptions are: • A period of approximately 5 minutes at each satellite sunrise and sunset, when the telemetry switches to Mode Number 2 and data from the scanning ozone experiment is transmitted in real time while data from coarse ozone experiment is recorded. • Upon command, low-speed (real time/48) data from the ozone and galactic-noise experiments, stored in the tape recorder, are transmitted at 48 times the recorded rate, giving same bandwidth characteristics as real-time transmission.
Telemetry transmitter:	Operates in the 136-137-Mc band. PFM/PM emission is used. RF power output will be 0.25 watt.
TRACKING	The telemetry transmitter carrier signal is used for tracking purposes. Minitrack system.
COMMAND SYSTEM	The command receiver operates in the 120-Mc region. It receives the ground-station signal which initiates tape-recorder readout. An address-command decoder will be used.

TELEMETRY STATIONS	Woomera, Australia Mojave, California E. Grand Forks, Minnesota Blossom Point, Maryland Fort Myers, Florida St. Johns, Newfoundland	Quito, Ecuador Lima, Peru Antofagasta, Chile Santiago, Chile Johannesburg, South Africa Winkfield, England

LAUNCH PHASE

Launch facility:	Wallops Station, Virginia
Launch vehicle:	Scout B
Orbital plan:	51° inclination, 150-nautical-mile perigee, 810-nautical-mile apogee
Satellite life:	One year
Launch date:	Calendar year 1963

Appendix C

Description of Equipment

The S-52 satellite, shown in Figure C-1, is composed of five major subsystems. These are the Experiment Subsystem, Structural Subsystem, Power Supply Subsystem, RF Recorder-Programmer Subsystem, and Encoder and Performance Monitoring Subsystem. Each subsystem and the subunits they contain are described in the following paragraphs.

EXPERIMENT SUBSYSTEM

The Experiment Subsystem is composed of three subunits, which are the Galactic Noise Unit, the Atmospheric Ozone Unit, and the Micrometeorite Flux Unit (see Figures C-2, C-3 and C-4). These subunits are supplied by the British National Committee for the purpose of gaining scientific information from the ionosphere.

STRUCTURAL SUBSYSTEM

The Structural Subsystem of the satellite is composed of the Structure Assembly, the Solar Paddle Erection Arms, the Inertia and Galactic Noise Antenna Booms, and the De-spin Mechanism. The Structure Assembly provides the basic mounting surface and members for the other subsystems, as well as the required rigidity to withstand vibration, thermal stress, thrust forces of the booster, and other critical mechanical effects. The main structural member of the satellite is the cylindrical magnesium center post to which the satellite portion of the booster separation mechanism is attached. The lower portion of the center post contains eight equally spaced radial support ribs which absorb a portion of the acceleration loads during launch of the satellite. These support ribs provide mounting surfaces for the solar paddle erection arms and the inertia and galactic noise antenna booms. The center post houses the tape recorder and the galactic noise receiver and antenna release mechanism. It also serves as a cable passage between equipment decks; the interconnecting cables are part of the Structure Assembly.

The upper and lower equipment decks, in addition to mounting the experimental and satellite equipment, add stability to the structure. Both decks are honeycomb members containing mounting brackets and threaded inserts for equipment mounting and integral inner and outer mounting flanges. The outer flange of the lower deck contains the de-spin mechanism. Experimental equipment is mounted on the upper deck and satellite electronics equipment and battery packs are mounted on the lower deck.

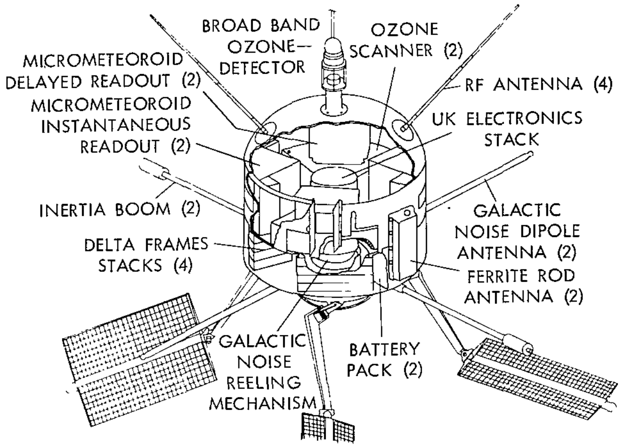


Figure C-1—UK-2/S-52 International Satellite No. 2.

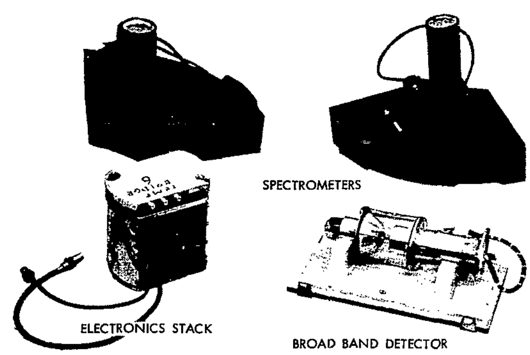


Figure C-2—Ozone experiment.

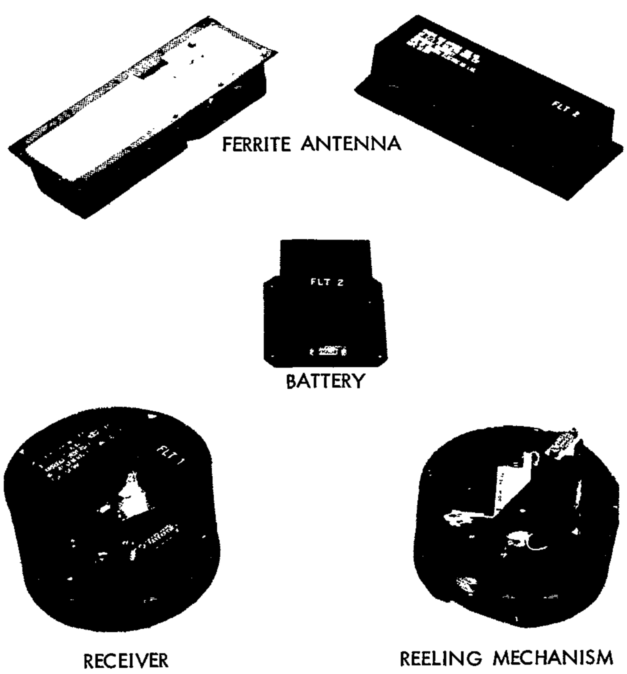


Figure C-3—Galactic noise experiment.

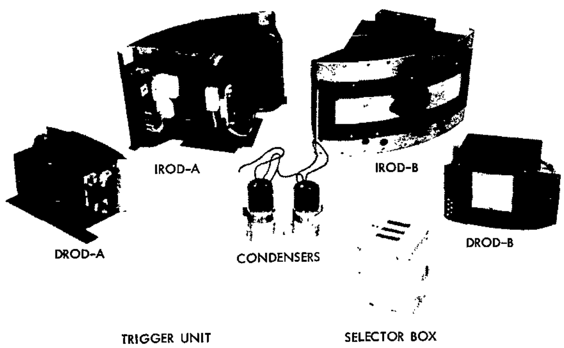


Figure C-4—Micrometeorite experiment.

The De-spin Mechanism is of the radial release type utilizing two wire cables and two lead weights. The weights are released by squib-actuated guillotines which are controlled by the separation mechanism timer. The purpose of the de-spin mechanism is to decrease the initial satellite spin rate of approximately 160 rpm to approximately 5 rpm for the duration of the satellite in orbit (one year minimum).

The outer skin of the structure assembly is filament-wound fiberglas which not only serves as an additional structural member, but furnishes protection for internal equipment from solar radiation and micrometeorite bombardment and acts as a thermal radiator to aid in controlling temperature within the satellite. The outer skin extends over the two cylindrical center sections (midskins) and over the upper and lower dome assemblies. The upper midskin contains openings for experimental equipment sensors and cutouts for access hatches which are used during assembly, test, and maintenance of the experimental equipment. The lower midskin is removable in two parts for access to the satellite equipment mounted on the lower deck. The skin on the lower dome assembly is detachable in four parts so that removal of the solar paddle erection arms, antenna booms and inertia booms is not required during normal test and maintenance routines.

The upper dome assembly contains an extension which houses the Atmospheric Ozone Unit. Four turnstile antennas, which function in the telemetry link, are attached to the upper dome reinforcement ribs. These antennas fold inward to facilitate installation of the nose fairing on the booster vehicle.

The Solar Paddle Erection Arms are attached to mounting surfaces on the lower dome assembly reinforcement ribs by means of hinge assemblies. Inertia and Antenna Booms are also hinged to the lower dome for stowage along the booster prior to launch. These booms are spring loaded to ensure erection when the satellite is in orbit. Smaller springs located at the ends of the antenna booms start unreeling the dipole at the instant of release.

POWER SUPPLY SUBSYSTEM

The Power Supply Subsystem of the satellite provides the necessary electrical power for operation of all satellite equipment. It includes batteries which permit equipment operation while the satellite is within the Earth's shadow. The following subunits are parts of the Power Supply Subsystem: Solar Paddle Assemblies, Battery Pack Assemblies, and Power Converters.

The four Solar Paddle Assemblies charge the battery packs and supply electrical power to the satellite when their cells are subjected to sunlight. Each solar paddle contains four modules each of which is made up of 42 solar cells, such that the complete solar paddle system consists of 16 modules (672 cells) connected in parallel to supply a total current output of 1.050 amperes at 12.8 watts. At a cell operating temperature of 28° C (82.4°F), the current output of each module is 370 milliamperes at an operating voltage of 11.9 to 14.6 volts.

Two battery packs, each containing 11 nickel-cadmium cells, supply power to the satellite when the solar paddles are not subjected to sunlight. The batteries are highly reliable and, in conjunction with the battery charging and protective unit, they are intended to support a minimum satellite life of one year. The battery packs are designated A and B. Normally, battery A is the power source, while battery B is the reserve unit. A differential voltage sensor, in the battery charging and protective unit, controls a latching relay which reverses the status of the batteries when an appreciable voltage drop occurs across the terminals of the operating battery. The reserve battery is trickle-charged through a transistor network during the daylight portion of the orbit. The battery packs are mounted on opposite sides of the lower equipment rack of the structure assembly.

A single inverter and a magnetic amplifier regulator produce the -18 volt and the +15 volt supplies from which all low voltage supplies are obtained. A 1700 cps supply is derived from the inverter feedback transformer. Through series regulation, +12 volts is obtained directly from the battery bus. All other positive low voltage supplies obtain their input power from the regulated +12 volt source. The -3 volt, the -4 volt, and the -6 volt supplies derive their input power from the regulated -18 volt source through series regulators. The nine voltages supplied by the power converters are as follows:

7.5v ± 1/4 percent	-6v ± 1/4 percent
3v ± 1/4 percent	-4v ± 5 percent
6.5v ± 5 percent	-18v ± 1 percent
12v ± 1 percent	15v ± 1 percent
-3v ± 1/4 percent	

RF RECORDER-PROGRAMMER SUBSYSTEM

This subsystem of the satellite provides the functions of programming the occurrence, in the proper sequence, of events aboard the satellite. Upon command from the programmer, the recorded data is played back for transmission to ground stations via the telemetry link. Subunits are as follows: Antenna and Hybrid, Command Receiver, Transmitter, Recorder, Main Programmer, Recorder Programmer, Receiver Decoder, Undervoltage Detector, and Tape Recorder Converter. The subsystem is an item of Government-furnished equipment.

ENCODER AND PERFORMANCE MONITORING SUBSYSTEMS

These satellite subsystems receive data from the experiments and the performance monitoring transducers and translate this data into a format which is compatible with the pulsed frequency modulation telemetry system. The subsystems are composed of High-speed Encoders, Low-speed encoders, and Performance Monitoring Transducers. The performance monitoring transducers are items of Government-furnished equipment.

Appendix D

Theory of Operations

GENERAL

The UK-2/S-52 is the second of three scientific satellites to be built under the cooperative international space exploration agreement entered into by the United States and the United Kingdom. The responsibility of the United Kingdom, as assigned to the Minister for Science, is to design and construct the instrumentation for the S-52 experiments as well as to reduce and analyze all data gathered from the spacecraft. Goddard Space Flight Center, for the National Aeronautics and Space Administration, has United States responsibility for the program, including the design, development, construction, and testing of the spacecraft, as well as launching, tracking, and data acquisition phases. GSFC has contracted with the Aerospace Division of the Westinghouse Electric Corporation to integrate the satellite, furnish certain subsystems, and assist in testing the satellite. The NASA Langley Research Center is responsible for the launch vehicle system.

DESCRIPTION OF EXPERIMENTS

Galactic Noise

This experiment is designed to measure galactic noise in the frequency range from 0.75 to 3.0 mc. The information gained will be of primary value to radio astronomers. Radio astronomy is the study of the Universe through the reception and analysis of radio-frequency signals from stars, galaxies, and interstellar space. Very little information exists as regards galactic noise in the frequency range of this experiment. The experimental equipment will be built by the Mullard Radio Astronomy Observatory, Cavendish Laboratory, University of Cambridge.

Measurements will be made by means of a 130-foot dipole antenna and two ferrite rod antennas coupled into a low-noise, frequency-swept receiver. Immediately after separation from the fourth stage motor, the 130-foot dipole antenna (65 feet on each side) will be deployed from a drum through two booms on either side of the satellite. Centrifugal force will pull the wire through the boom from the drum, which will be held at a constant speed by means of a motor drive mechanism.

Two ferrite loops, mounted on diametrically opposite sides of the satellite, will be tuned to approximately 2 megacycles. These antennas will continue to provide useful data even if the dipole antennas fail to deploy.

The galactic noise will be measured from both types of antennas as the receiver sweeps repeatedly over the range of 0.75 to 3 mc.

Atmospheric Ozone

This experiment will allow the measurement of vertical distribution of ozone in the upper atmosphere by two methods: a broadband method and spectrum scanning method. Although ozone distribution in the lower atmosphere is regularly probed by rocket soundings, continuous monitoring by a satellite should provide much useful data. The information gained is important to meteorologists because it is believed to affect the heat balance of the earth, and therefore the weather. The experiment is supplied by the Air Ministry Meteorological Office.

The broadband ozone method makes use of a simple photocell with a spectral response in the region of 1800 to 3500 A. U. In this wavelength region most of the atmospheric attenuation is due to ozone. Therefore, during satellite "sunrise" and "sunset," when the satellite is illuminated by sunlight passing through the atmosphere, a measure of the photocell output gives an indication of ozone present by virtue of the spectral energy absorbed.

Because some attenuation is caused by dust and air molecules, a second photocell is added in the broadband unit to compensate. In this case the photocell response is in the region of 3600-4000 A. U. The output of this photocell corrects the information obtained from the other photocell, so that a more accurate measure of ozone concentration is obtained.

The spectrometer or "scanning" method of measuring ozone involves the use of a simple form of prism spectrometer to scan through the solar spectrum in the 2650-4000 A. U. region. The scanning is accomplished by the rotation of the satellite. There are eight optical units; as the satellite rotates, each optical unit in turn causes the solar spectrum to scan across one of two photomultiplier units. The shape of each of the pulses of light received by the photomultipliers, corresponding to the shape of the solar spectrum as modified by ozone absorption, constitutes the information used in the experiment.

Micrometeorite Flux

This experiment will measure the size and number of micrometeorites in the ionosphere. Micrometeorites, in sizes ranging from that of a grain of sand down to millionths of an inch, have an eroding effect on spacecraft. The information to be gained is necessary for the design of future spacecraft and space stations. The experimental equipment will be supplied by the Nuffield Radio Astronomy Laboratories, University of Manchester.

There are two pairs of micrometeorite detectors: Instantaneous Read-Out Detectors (IROD) and Delayed Read-Out Detectors (DROD).

The instantaneous readout or IROD units determine the size and quantity of micrometeorites by measuring the light admitted through punctures in a 12-micron aluminum foil. The punctures are made by the micrometeorites on exposed foil, which is advanced across an aperture in the IROD unit. The foil is coiled on a spool and is advanced approximately 1/16 inch every other orbit. The light is sensed by means of solar cells inside the IROD units.

The delayed read-out detectors or DROD units are primarily designed to measure the erosion caused by very small micrometeorites. This measurement is made by determining the amount of light admitted through abrasions on a metalized mylar surface. The metallic coating on the mylar is only about 5 microns thick. The mylar is coiled on a spool and is advanced across an aperture in the DROD unit in a manner similar to that of the IROD.

STRUCTURE

The Spacecraft Structure is a lengthened and redesigned version of the UK-1/S-51, approximately 23 inches in diameter and weighing 160 pounds. The unit will have a spin rate in orbit of 5 rpm after a de-spin sequence from an initial spin rate of 160 rpm. The structure basically consists of two honeycomb decks coupled by a center tube to the separation mechanism. The lower deck and appendages are braced by several struts between the lower deck bottom and the center tube. The outer shell of the satellite, made of fiberglas, completes the structure support and serves also as the thermal coating surface and ground plane.

POWER SUPPLY SUBSYSTEM

This subsystem supplies power to the various electronic packages at specified voltage levels and within the required regulation limits. There are 10 levels of d-c voltages and one a-c voltage, in addition to the basic battery bus voltage. The solar paddles are the primary source of power and they must supply the load as well as charge the batteries during the daylight portion of the orbit. During the darkness period, batteries supply the load.

Four solar paddles are mounted on the ends of hinged arms attached to the lower deck struts. Each solar paddle contains four modules, each of which is made up of 336 solar cells in a series-parallel arrangement (48 x 7). The complete solar paddle system therefore consists of 16 modules (5376 cells) connected in parallel to supply power at an upper voltage limit of 16.5 volts. The cells were originally planned to be of the P-on-N type, but because of degradation expected by the artificial radiation belt, a change was made to utilize the more radiation-resistant N-on-P cells. Power available at most favorable aspect (assuming no radiation degradation) should approach 30 watts. Power available at the end of one year (assuming radiation degradation) and poor aspect (±30 degrees) is expected to be adequate to supply satellite minimum power requirements of approximately 14 watts.

Two battery packs contain 11 nickel-cadmium cells each. Depth of discharge during a single satellite night will not exceed 12 percent. The batteries are highly reliable and intended to support a minimum satellite life of one year. Only one battery is required; the second is supplied as a redundant or reserve unit.

There are three converter delta packs in the Power Supply Subsystem: Input and Inverter, Positive Voltage Regulators, and Negative Voltage Regulator. The delta pack is a standardized packaging module utilized in the satellite, consisting of a basic frame and card encapsulated in foam plastic. The inverter pack receives power from the unregulated bus and generates the 7.5v ± 1 percent, 1700 cps, 15 vdc ± 1 percent, and a negative unregulated voltage. The positive regulator generates the positive d-c 12v ± 1 percent, 7.5v ± 1/4 percent, 6.5v ± 5 percent, 6.0v ± 1 percent and 3v ± 1/4 percent supply voltages. The negative regulator pack generates the -18v ± 1 percent, -6v ± 1/4 percent, -4v ± 5 percent, and -3v ± 1/4 percent supply voltages.

One delta pack mounted on the lower deck houses the battery-charging and protective-circuit networks. This circuit, through a differential voltage sensor, can switch the batteries if a satellite undervoltage condition occurs and the voltage drop across the terminal of the operating battery is appreciable. The operating battery charge rate, standby battery trickle charge, and load dumping (to protect against overcharging) are also controlled from this circuit. Another delta pack mounted on the lower deck houses the undervoltage detector. This circuit, when activated by an undervoltage condition, trips the satellite load and allows for 18 hours of battery charge before normal satellite operation is resumed.

PROGRAMMER

This subsystem consists of two delta packs, identified as Programmer No. 1 and Programmer No. 2. Programmer No. 1 serves as the interface between the Encoder, Tape Recorder, Transmitter, and Command Receiver. It provides timing functions and generates identification frequency bursts (the "horn") associated with tape recorder playback commands. Upon receipt of a playback command from a ground station, a 320.83 cps signal is gated to the telemetry transmitter for real-time transmission, and at the same time is recorded for two seconds. After two seconds, Programmer No. 1 initiates tape-recorder playback, and the stored data is played back via the telemetry transmitter at 48 times the recorded rate. Because the data was recorded at real-time/48, it appears as real time to the ground station. Programmer No. 1 also provides the 320.83 cps fill frequency inserted between the bursts of low-speed data.

Programmer No. 2 supplies to the high-speed and low-speed encoder the control signals that determine their modes of operation. It also provides the trigger pulses to the micrometeoroid experiment for initiating the foil advance and selector switching functions. A sunset sensing circuit is activated by a decay in solar paddle voltage, and a sunrise circuit by a rise. This voltage change is used to determine the program start time. The program cannot be initiated unless the sensor has seen a decrease in solar voltage (shadow) followed by a rise (sunrise) in solar voltage.

The beginning of the program or T = 0 occurs at less than 1 percent sunlight, determined by sensing a solar paddle voltage (loaded by approximately 20k resistance) rise to 12.15 ± 0.25 volts.

At time T = 0 the high- and low-speed encoders are switched to mode 2 for six minutes of real-time telemetry and tape recorder storage devoted entirely to the ozone experiment (morning twilight period). The Programmer No. 2 clock is also reset and started at time T = 0 (sunrise).

The next key event in the program occurs at T + 6 minutes. At this time the programmer signals the high-speed and low-speed encoders to cease processing ozone information and switch to mode 1. The High-speed Encoder (real time) format is shared by Performance Parameter monitoring, the Galactic Noise Experiment and the Micrometeorite Experiment in Mode 1 operation. The Low-speed Encoder processes galactic noise information for storage in the Tape Recorder during Mode 1. Also, at time T + 6 minutes, a signal is sent to the Micrometeorite Experiment to initiate the micrometeorite foil advance and to select the proper micrometeorite IROD and DROD for the current orbit.

At T + 60 minutes Programmer No. 2 commands the high- and low-speed encoder to switch to Mode 2 for a second (evening twilight) period of acquiring ozone information. Any time after T + 6, the appearance of shadow will terminate the program. (Because of built-in circuit delays, a shadow response time of approximately 2 minutes is normal.) The clock would be stopped, both encoders would return to Mode 1, and Programmer No. 2 would go into a standby condition, waiting for the next sunrise to initiate a new program sequence. (Shadow can be expected any time after T + 65 minutes.)

If shadow has not been sensed by T + 78 minutes, Programmer No. 2 will command the high-speed encoder to cease real-time ozone processing and return to Mode 1. The low-speed encoder will continue to process ozone information (Mode 2) for storage in the tape recorder.

At time T + 110 minutes, if shadow still has not been sensed, the spacecraft has entered a 100 percent sunlight orbit, since orbit time is approximately 103 minutes; and it may not sense shadow for a period of as much as 10 calendar days. In this case the clock runs out at time T + 110 minutes and the Low-speed Encoder is commanded to join the High-speed Encoder in Mode 1. Programmer No. 2 goes to a static condition and awaits a sunset before a new program can be initiated by a sunrise.

Figure D-1, shows the event programming when: (a) sunset does not occur, as in 100 percent sunlight, (b) sunset occurs between T + 60 and T + 78, the normal mode of operation, and (c) sunset occurs between T + 78 and T + 110, the long twilight periods before and after 100 percent sunlight phase.

TELEMETRY SYSTEM

The telemetry system utilized in the S/52/UK-2 satellite is a Pulsed Frequency Modulation system (PFM). This is a particular form of time-division multiplexing in which the intelligence being telemetered is contained in the frequency of a sequential series of 9.09 millisecond pulses separated by 9.09 millisecond intervals. The pulse frequency is derived from a set of pulsed subcarrier oscillators operating in the frequency range from 4.5 kc to 15 kc.

The PFM telemetry system, particularly the ground station data-handling equipment, requires that the signal level contained in a given data pulse remain constant. Since the information from both the Ozone and the Galactic Noise Experiments may include signals with rapid rates of change, a sample hold or staticizer circuit is provided to convert samplings of these experiment outputs to constant-data-pulse values for the encoder inputs.

Encoders

Two encoders, termed the High-speed and Low-speed Encoders, are used in this satellite. Their purpose is to accept transducer outputs from the experiments, commutate them, and produce a pulsed-frequency output proportional to the value of the parameter being measured in each of the experiments. The output from the High-speed Encoder modulates the transmitter directly (real-time data), but the output from the Low-speed Encoder is recorded on a tape recorder for a complete orbit at 1/48 the information rate of the high-speed system. On command the tape output is played back at 48 times the recorded speed, so that the output from both systems when received at the ground station will have the same bandwidth.

Operating Modes

There are two modes of operation for the encoders. During Mode 1 the High-speed Encoder output consists of a synchronization pulse, digital frame identification, monitoring of performance parameters, high-bandpass data from the Galactic Noise Experiment and data from the Micrometeoroid Experiments. This output is transmitted directly in real time. The low-speed encoder output during Mode 1, consisting of Galactic Noise low bandpass data, is recorded by the tape recorder during this period.

Satellite Operational Modes

Experiment	Output Designation	Encoder	Mode
Galactic Noise	G1	High Speed	1
	G2	Low Speed	1
Atmospheric	0 3	High Speed	2
Ozone	0 1	Low Speed	2
	0 2	Low Speed	2
Micrometeorite	IROD A	High Speed	1
	IROD B	High Speed	1
	DROD A	High Speed	1
	DROD B	High Speed	1

High-Speed Encoder Output

The High-speed Encoder output consists of 256 data coordinates arranged in 16 frames, with each frame in turn consisting of 16 channels. Since the blank/burst interval for each coordinate takes 18.18 milliseconds, a complete high-speed telemetry sequence of 256 channels takes 4.654 seconds. The format for the High-speed Encoder is tabulated in Figure D-2.

Low-Speed Encoder Output

The Low-speed Encoder output into the tape recorder consists of a single frame of 16 channels. The time of recording of the low-speed blank/burst interval is 48 times that of the high-speed interval, or 0.87 second. Therefore, a complete low-speed telemetry sequence of 16 channels takes 13.96 seconds. The channel allocation for the Low-speed Encoder is tabulated in Figure D-2.

Encoder Synchronization

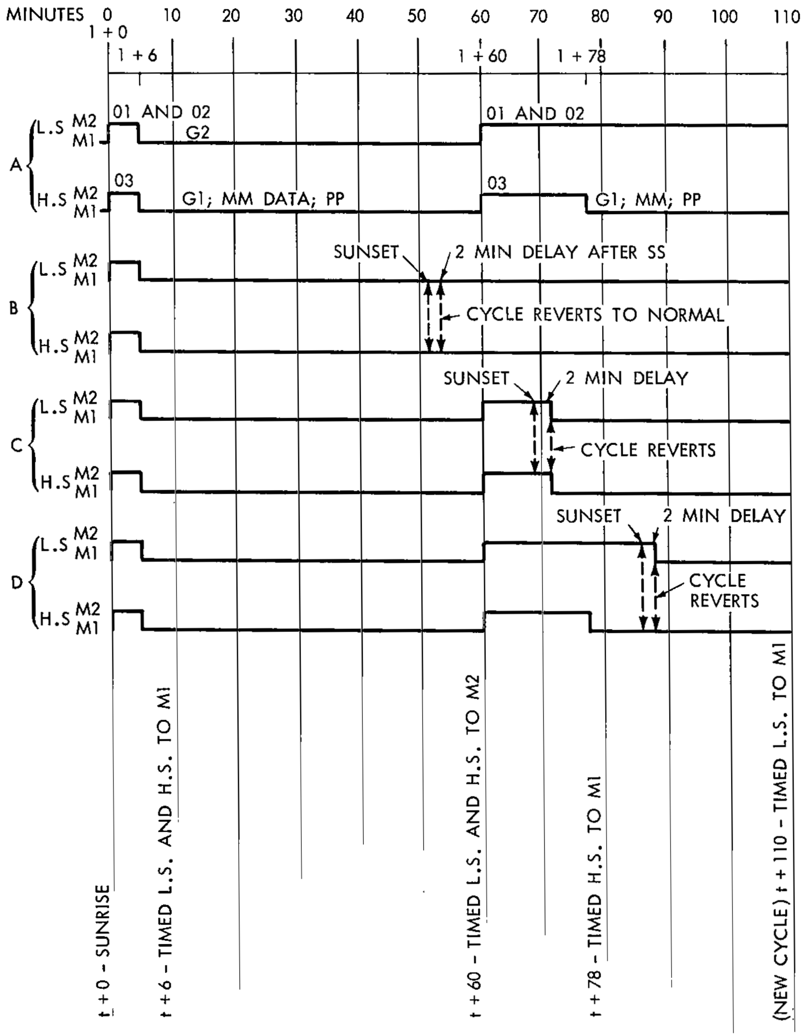
It will be noted that three high-speed telemetry sequences (48 frames) will occur for each low-speed telemetry sequence. The two encoders will be synchronized so that the initiation of channel 0 frame 0 of the low-speed encoder will correspond to channel 15 frame 15 of the high-speed encoder. This synchronization will automatically take place every 13.96 seconds.

High-Speed Encoder

Channel zero for all odd number frames is devoted to frame synchronization. This practice simplifies the data reduction problem in that frame synchronizer may come out of a special filter. Even-numbered frames are identified by using 8 levels of a 9-level digital oscillator. The 9th level is the frame synchronizer frequency.

The exact format for high-speed Mode 1 is as follows:

Channel — Frame	Designation	Frequency (kc)
0 — 0	000 digit frequency	5.1
0 — 1	Synchronizer frequency	4.5
0 — 2	001 digit frequency	6.3
0 — 3	Synchronizer frequency	4.5
0 — 4	010 digit frequency	7.5
0 — 5	Synchronizer frequency	4.5
0 — 6	011 digit frequency	8.7
0 — 7	Synchronizer frequency	4.5
0 — 8	100 digit frequency	9.9
0 — 9	Synchronizer frequency	4.5



LEGEND

M1 L.S. = G2 M2 L.S. = O1 AND O2
M1 H.S. = G1, MM AND PP M2 H.S. = O3

Figure D-1—Programmer No. 2 cycling.

ENCODER FORMAT

FRAME	CHANNEL 0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
0	0 5.1 0 0 196.1								PP OZ TEMP 1 O MON CELL							
1	SYNC	M	G	M	M	M	G	M	PP OZ TEMP 2 1 OZ CELL	M	G	M	M	M	G	M
2	0 6.3 0 1 158.7	M	N	M	M	M	N	M	PP OZ TEMP 3 2 SPECT A	M	N	M	M	M	N	M
3	SYNC								PP OZ EHT 3 MON							
4	0 7.5 1 0 133.3	I	H	I	D	I	H	I	PP +15V 4	I	H	I	D	I	H	I
5	SYNC	R	S	R	R	R	S	R	PP TAPE REG 5 TEMP	R	S	R	R	R	S	R
6	0 8.7 1 1 114.9	O		O	O	O		O	PP DUMPED 6 CURRENT	O		O	O	O		O
7	SYNC	D		D	D	D		D	PP UNREG 7 BUS	D		D	D	D		D
8	1 9.9 0 0 101.0								PP GN REEL 8 / + 12V							
9	SYNC	A		A	B	A		A	PP SOLAR 9 CURRENT	A		A	A	A		A
10	1 11.1 0 1 90.1	OR		OR		OR		OR	PP BATT 10 CURRENT	OR		OR		OR		OR
11	SYNC	B		B		B		B	PP BATT A 11 TEMP	B		B		B		B
12	1 12.3 1 0 81.3								PP PADDLE 4 12 TEMP							
13	SYNC								PP DOME 13 TEMP							
14	1 13.5 1 1 74.1								PP SKIN 14 TEMP							
15	SYNC	A		B		A		B	PP GN SW 15 MON	A		B		A		B

HIGH SPEED MODE ONE

SYNC	OZONE SPECTRUM

HIGHT SPEED MODE TWO
LOW SPEED MODE ONE

SYNC	GALACTIC NOISE	SYNC

LOW SPEED MODE TWO

SYNC	01	02	01	02	01	02	01	02	01	02	01	02	01	02	01

SYNC = 4.5 KC 222.2 NS   01 = OZ PHOTOCELL

LS B/B = 15.4 KC 64.9 NS   02 = MON PHOTOCELL

Figure D-2—Encoder format.

Channel — Frame	Designation	Frequency (kc)
0 — 10	101 digit frequency	11.1
0 — 11	Synchronizer frequency	4.5
0 — 12	110 digit frequency	12.3
0 — 13	Synchronizer frequency	4.5
0 — 14	111 digit frequency	13.5
0 — 15	Synchronizer frequency	4.5

In addition to the above, width modulation is employed. Channel 0 of each frame of the High-speed Encoder consists of a 4.54 millisecond blank followed by a 13.62 millisecond burst. Thus the total burst/blank period of 18.18 milliseconds is maintained throughout.

The high-speed Mode 2 sequence format is shown in Figure D-1. To facilitate data reduction it is planned to telemeter the synchronizer frequency (4.5 kc) during channel zero time.

Low-Speed Encoder

Channels 0 and 15 are used for frame synchronization of Mode 1, in which galactic noise is the sole experiment. Channel 0 is used for synchronization in Mode 2, during which two outputs of the ozone broadband experiment are telemetered. When recorded on tape, the encoder output frequency is first divided by 48. When the tape is played back at 48 times the recording speed, the exact synchronizer format for the low-speed encoder is as follows:

Channel	Telemetered Frequency
(Mode 1)	
0 & 15	4.5 kc
1 — 14 (GN2)	5 to 15 kc
(Mode 2)	
0	4.5 kc
1 — 15 (0_1 & 0_2)	5 to 15 kc

As in the High-speed Encoder, width modulation is also used in the low-speed synchronization channels. Thus, when received on the ground, the synchronization channels of the Low-speed Encoder will consist of a 4.54 millisecond blank followed by a 13.63 millisecond burst at 4.5 kc synchronizer frequency.

It should be noted the "blanks" for the Low-speed Encoder will not be blanks but will consists of a stable 15.4 kc reference signal on playback. The Low-speed Encoder provides a gating signal to the programmer which generates a 15.4 kc/48 signal to be recorded on tape.

Intelligence Bandwidth

The calibrating and frame synchronization techniques have extended the bandwidth from 4.5 kc to 15.4 kc, instead of the usual 5 kc to 15 kc bandwidth.

Commutation

Commutation in the two matrices is done by completely electronic means, utilizing silicon transistors throughout. The count-down units were made up of complementary binaries. The subcarrier oscillators are designed so that the most likely failure will be in the off condition; so that only the information contained in that one oscillator will be lost. (It is highly undesirable for an oscillator to fail in an "on" condition, inasmuch as that experiment would then be on all the time.) Thus, for most of the time its output data would be mixed with outputs of the other oscillators with the result that all data from the other experiments would be lost.

TAPE RECORDER

The Tape Recorder system, in addition to the tape recorder itself, includes three electronic subsystems: Tape Recorder Converter, Programmer No. 1, and Data Storage Control. The Tape Recorder Converter is powered from the satellite unregulated bus and provides regulated positive and negative voltages for the tape recorder system operation. Programmer No. 1 receives low-speed data from the encoder, conditions it, fills the blank time with a locally generated 320.8 cps signal, and gates this combination to the tape recorder record amplifier. In addition, Programmer No. 1 processes high-speed data from the encoder and delivers it to the transmitter modulator. Upon command from the receiver decoder, Programmer No. 1 switches gates, blocks high-speed data, and transmits tape recorder playback data to the Transmitter. The Data Storage Control receives positive and negative regulated voltage from the tape recorder converter and produces the 100 cps tape recorder motor drive power. This module also contains the relays which switch from record to playback mode on command from Programmer No. 1. Decade oscillators and counters fix the playback time and automatic return to record function. The Tape Recorder records the output of the Low-speed Encoder and upon command plays back the data at 48 times the recorded speed.

Record

The recorder operates during the Record Mode with a tape speed of 0.25 inch per second. The recorder has a tape capacity of at least 110 minutes.

Playback

The recorder operates in the Playback Mode for 138 seconds. The playback-to-record speed rate is 48 to 1 (±0.5 percent). Playback tape speed is 12 inches per second. The motor operates in the reverse direction to that during record.

Playback Operation

The recorder remains in the Record Mode continuously until it is switched into playback by command. Playback time is controlled by an external electronic timer in Programmer No. 1. At the end of the playback period the timer in Programmer No. 1 switches the recorder back to the record mode.

Frequency Range

The recorder has a playback frequency range of 3600 to 16,800 cps at 3 db down.

Noise Ratio

Dynamic signal-to-noise ratio is at least 30 db.

Amplitude Fluctuations

Amplitude fluctuations are no greater than 10 percent variation of the amplitude of the highest recordable frequency.

Flutter

Flutter is no greater than one percent peak-to-peak for any one discrete flutter frequency component or for any 200 cps bandwidth notch between 3600 cps and 25,000 cps during playback.

TRANSMITTER

The transmitter is designed to operate with phase modulation and to be compatible with phase-lock receiving systems. The modulating signal is in the form of tone bursts which are square-wave in nature. A phase deviation of ±57 degrees is used to provide a total sideband power, which is twice the carrier power. The incidental frequency-modulation of the carrier is kept less than 5 cps in order to ensure compatibility with ground station equipment. The carrier frequency is located in the 136 to 137 mc telemetry band, and the output power to the antenna system is 0.25 watt. The oscillator operates at approximately 68 mc and has a temperature frequency stability of ±0.002 percent or less. The oscillator signal is fed through a buffer amplifier to a constant-impedance phase modulator. The modulator output is put through a frequency doubler to an amplifier, which provides the proper output power to a 50-ohm antenna system.

COMMAND RECEIVER – RECEIVER DECODER

The receiver is of the superhetrodyne type and has an i-f bandwidth of 220 kc with a good selectivity curve. The interrogation frequency is the standard NASA command frequency in the 120 mc region. The interrogation signal is amplitude modulated by the assigned audio tone. Two audio tones are used to provide security against accidental playback of the Tape Recorder with attendant loss of all stored information. The Receiver has a sensitivity of -100 dbm.

Appendix E

Ground Support Equipment

GENERAL

This section contains descriptive information concerning the ground support equipment for the S-52 satellite. Each item of special test equipment and ground handling equipment comprising the ground support equipment is described as to its purpose, use, and operation. Special emphasis is placed on the test stand and data reduction unit, an item of special test equipment, in that use and operation of each Westinghouse-manufactured panel is explained separately. Descriptive information for each item of commercial test equipment contained in the test stand and data reduction unit is given in the handbook supplied by the equipment manufacturer, in most cases Hewlett-Packard Co., Palo Alto, California.

SPECIAL TEST EQUIPMENT

The special test equipment supplied to test and maintain the S-52 satellite is composed of the following items:

Satellite Rotator
Antenna Deployment Mechanism Test Fixture
Solar Simulator
Test Stand and Data Reduction Unit

Descriptive information pertaining to each of these equipments is contained in the following paragraphs.

Satellite Rotator

The Satellite Rotator is used to test the S-52 satellite under simulated spin conditions encountered in flight. The satellite is rotated at 5 ± 0.5 rpm by the rotator drive motor with solar paddles and antenna and inertia booms extended. A worm gear and crank on the rotator positions the spin axis of the satellite between the horizontal and the vertical attitudes. Horizontal orientation of the satellite spin axis facilitates simulation of sunrise and sunset, and in conjunction with the experiment stimulators, permits simulation of various sunline inclinations up to ± 90 degrees.

A slipring assembly, containing 40 rings, is mounted on the drive shaft of the rotator. These sliprings pass power to the satellite and permit monitoring of signal voltages while the satellite is rotating. Electrical connections to the satellite are made with the connectors and cables on the rotator. The satellite rotator is caster mounted and contains adjustable position locks for leveling.

Antenna Deployment Mechanism Test Fixture

The Antenna Deployment Mechanism Test Fixture is used to test deployment of the galactic noise antenna. The fixture simulates the centrifugal and unreeling forces exerted on the antenna wires while the satellite is in flight and tests for smooth deployment of each antenna wire. Tension on the antenna wires is produced by suspending weights from the end bobs of the antenna. Since the Antenna Deployment Mechanism employed in the S-52 satellite is a constant speed reel, the basic test consists of measuring the time required to deploy the first three feet of each antenna wire, and checking for slack buildup during deployment. When the test fixture is in use, the satellite is installed on the satellite rotator, and the rotator is adjusted, by means of the handcrank, until the spin axis of the satellite is in the horizontal position. The satellite is then rotated on its spin axis until the antenna booms are horizontal.

Solar Simulator

The Solar Simulator is used to simulate the sunlight environment of the satellite for test of solar paddles during various satellite system and power supply subsystem tests. The simulator contains twenty-one 300-watt, tungsten-filament bulbs arranged in such a pattern above the solar paddle shelf as to provide maximum uniformity of light over the entire shelf area. Between the bulbs and the solar paddle shelf is a water-filled window assembly with inlet and outlet water connections. The purpose of this window assembly is to insulate the solar paddle placed on the shelf from the heat produced by the tungsten bulbs, but still pass light to the solar cells of the paddle. Water flowing through the window assembly is always in contact with both glass plates, upper as well as lower, so that light variations due to water ripples cannot occur. A solar paddle support is supplied with the simulator to keep the solar paddle placed on the shelf level during testing.

To exhaust heat from the Solar Simulator, five fans are employed above the tungsten bulbs and two fans keep air circulating over the solar paddle shelf. The lamp, window assembly, and solar paddle shelves are adjustable in the vertical direction and each contains two spirit levels to obtain parallel alignment. The adjustable shelves, in conjunction with the nine variacs located on the lower panel, are used to obtain a uniform light intensity over the solar paddle shelf. Each of the four large variacs controls light intensity of a corner group of four lamps. The five smaller variacs individually regulate intensity of each lamp in the centrally located group of five. For calibration purposes, a standard solar cell provides light intensity readout in terms of short circuit current. The entire solar simulator is mounted on casters for mobility.

Test Stand and Data Reduction Unit

The Test Stand and Data Reduction Unit is a five-bay console, the four right-hand bays comprising the test stand and the left-hand bay the data reduction unit. A separate mobile stand supports the Tektronix 545 oscilloscope. Each bay of the console contains a blower which supplies cooling air to the internal components.

Test Stand

The Test Stand consists of commercially available test equipment that is supplemented by the special equipment required to provide rapid system checkout in conjunction with the data reduction unit. The Test Stand also enables the isolation of malfunctioning units. Incorporated into the Test Stand are provisions for (1) checking satellite programming sequence, (2) simulating inputs to subunits, (3) supplying power to the satellite in substitution for batteries, power converters, and/or solar paddles, (4) measuring critical current and voltage values without adversely affecting system operation, (5) producing simultaneous strip chart displays of significant circuit parameters, and (6) tape recording and reproducing of system characteristics. All cables and connectors necessary to interconnect the Test Stand and the satellite, and to provide power inputs to the satellite for a missing power supply unit are supplied with the test stand. The power supply that replaces the solar paddles is variable to simulate voltage changes resulting from sunrise and sunset conditions encountered in flight.

Patch Panel—The Patch Panel, which is located on the right-hand bay of the Test Stand over the storage drawer, permits the connection of all Hewlett-Packard equipment and the data reduction unit to the HP562A analog printer. The printer is used to record various measurements provided by the associated equipment. Connectors arranged and wired to agree with equipment connectors on the rear of the panel provide interconnection capabilities at the front of the panel. The Patch Panel permits various combinations of information to be fed to the analog printer and enables control of the printer by the connected instruments or the data reduction unit. A two-gang switch at the lower center of the panel is used to select the print command required by the printer. The negative command is required for all Hewlett-Packard equipment, and the positive command is required for the data reduction unit.

Instrumentation Panel—The Instrumentation Panel provides convenient test points for monitoring satellite power supplies and interfaces. In addition, the panel provides electrical access to the tape record and reproduce amplifiers, the telemetry receiver, and the data reduction unit. During satellite system integration, the Instrumentation Panel supplies external power to the satellite, thereby eliminating power drain from the satellite supplies, and preventing possible damage to the supplies. The external power is limited, monitored, and switched on or off by controls on the panel.

Unit Functional Test Panel—The Unit Functional Test Panel provides separate connectors and associated circuits for testing each component of the satellite system. The panel monitors power supply current, provides loads for units under test, simulates inputs to the units, provides test points for monitoring equipment operation, and, in conjunction with the instrumentation panel, supplies monitored power to the satellite from external power sources. Cables for connecting the satellite units to the unit functional test panel are supplied with the test stand.

Blockhouse Control Panel—The Blockhouse Control Panel enables control of programmer speed-up, programmer reset, sunrise simulation, undervoltage detector reset, and undervoltage simulation. Provisions for monitoring and recording battery voltages and currents are also incorporated. Two power supplies on the panel provide umbilical power to the satellite for simulation of batteries A and B. These power supplies can be used in a constant current mode to check the Battery Control and Protection Unit. Functional testing of the Battery Control and Protective Unit is performed, using the Blockhouse Control Panel, since most of the controls required to operate the unit for battery substitution and monitoring are mounted on the panel.

Meters are mounted on the blockhouse control panel to indicate the amount of umbilical power being supplied to the satellite by the panel, and the amount of umbilical bus current supplied. Metering of umbilical bus current enables measurement of solar paddle current. A timed application of power to the satellite is provided by the panel to simulate various daylight-to-darkness ratios of solar paddle power.

Data Reduction Unit

The Data Reduction Unit receives data from the S-52 satellite and enables the operator to monitor this data through use of the selector switch on the front panel. Data selected is printed out with an identifying number on the analog printed HP562A.

GROUND HANDLING EQUIPMENT

The ground handling equipment for the S-52 satellite consists of the transfer and assembly dolly and the satellite lifting fixture. These are described in the following paragraphs.

Transfer and Assembly Dolly

The Transfer and Assembly Dolly is used to support the satellite whenever the satellite is not mounted on the booster vehicle or the satellite rotator. The dolly contains casters for mobility and a position lock which can be lowered to prevent movement of the satellite during test, assembly, and maintenance operations. A handle is provided on the dolly for movement of the satellite from one test area to another. When not in use, the handle is stored on the base assembly.

The satellite is attached to the mounting ring of the support assembly, which is shockmounted to the base assembly. Pins in the mounting ring engage holes in the satellite interface to ensure correct orientation of the satellite on the transfer dolly.

Included on the Transfer and Assembly Dolly is a hydraulic pump and a cylinder for raising and lowering the satellite. An arm assembly on the cylinder supports the satellite when the hydraulic pressure is released.

When the dolly is used to ship the satellite from one location to another the satellite appendages are placed in their stowed positions and the satellite is lowered so that the antenna and inertia booms enter tubes in the mounting plate. The cover assembly is enclosed around the satellite and attached to the mounting plate by 16 stud fasteners.

Satellite Lifting Fixture

The Satellite Lifting Fixture is used to transport the satellite from one test fixture to another or to hoist the satellite into the booster vehicle. The lifting fixture consists of a two-piece ring assembly which tilts on the lower portion of the frame assembly. One portion of the ring assembly is removable so the lifting fixture can be passed under the satellite when the satellite is attached to the Transfer and Assembly Dolly or the Booster Vehicle Separation Mechanism. The ring assembly tilts to permit easy attachment of the satellite to the horizontal spin axis of the satellite rotator. The position of the ring assembly on the frame is maintained by two pins which are inserted through the frame to engage holes in the ring assembly. Mounting fingers and captive screws for attaching the lifting fixture to bosses on the lower dome assembly of the satellite are included on the ring assembly.

A spring load bearing, attached to the top of the frame assembly, permits rotating the entire fixture 360 degrees about the vertical centerline of the fixture. The spring load feature of the bearing eliminates sudden lift surges that could be imposed by the hoisting device.

It should be noted that the S-52 satellite can support its own weight (approximately 155 pounds) at only two places. These are the four bosses on the lower dome assembly to which the lifting fixture attaches, and the lower surface of the satellite interface that mates with the separation mechanism on the booster vehicle. Structural damage could result if the satellite is permitted to rest on any of its other surfaces, or if any attempt is made to carry the satellite by grasping appendages or structural parts not intended to support the satellite weight.

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