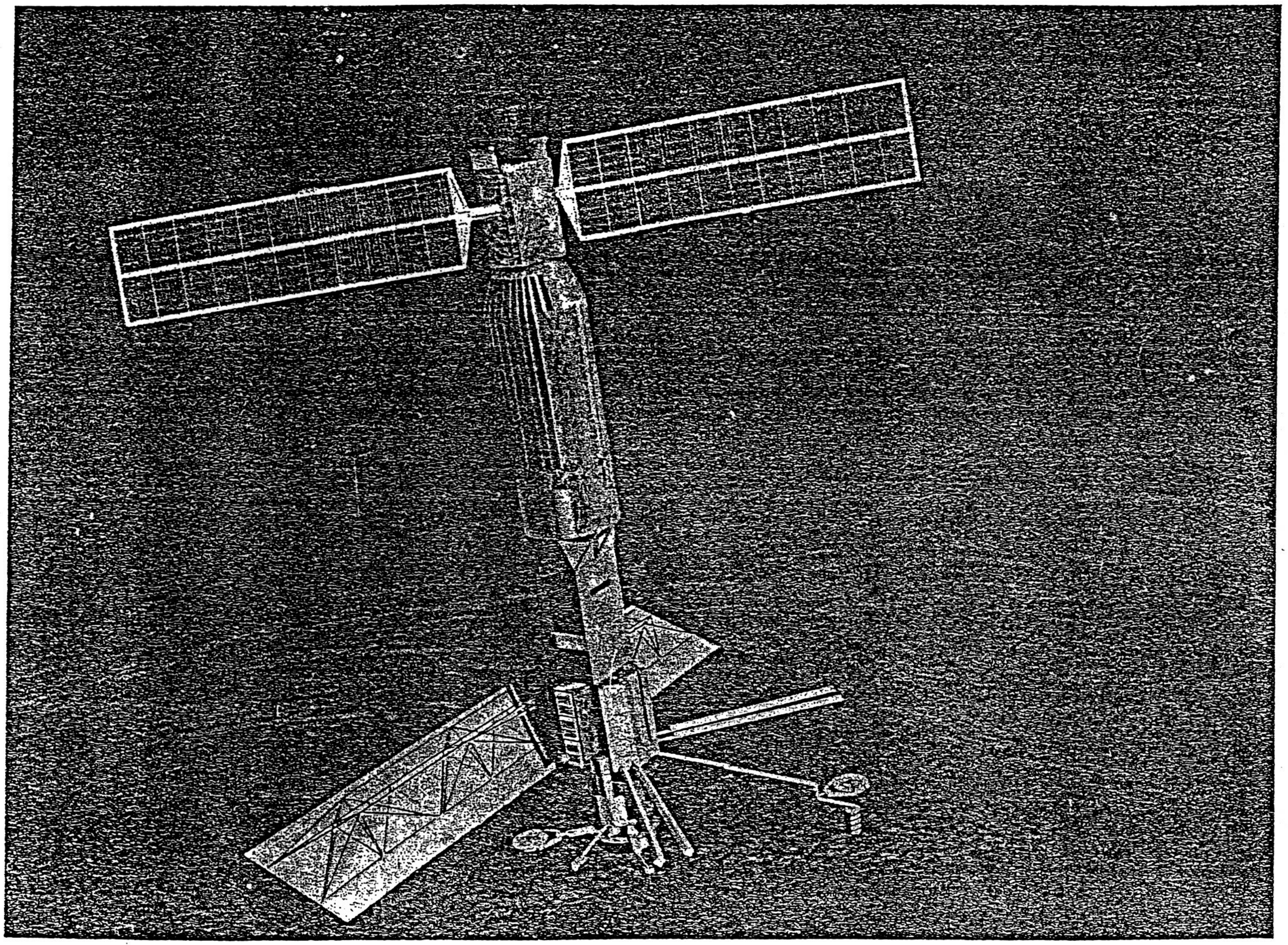
E. FRANK

Report of the Seasat Failure Review Board

DECEMBER 21, 1978



National Aeronautics and Space Administration

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SUMMARY

The Seasat spacecraft failed on October 9, 1978, after satisfactory operation in orbit for 105 days, as a result of a loss of electrical power in the Agena bus that was used as a part of the spacecraft. This loss of power was caused by a massive and progressive short in one of the slip ring assemblies that was used to connect the rotating solar arrays into the power subsystem. The most likely cause of this short was the initiation of an arc between adjacent slip ring brush assemblies. The triggering mechanism of this arc could have been either a wire-to-brush assembly contact, a brush-to-brush contact, or a momentary short caused by a contaminant that bridged internal components of opposite electrical polarity.

The slip ring assembly, as used in the Seasat spacecraft, was connected into the power subsystem in such a way that most of the adjacent brush assemblies were of opposite electrical polarity. This wiring arrangement, together with the congested nature of the design itself, made the Seasat slip ring assembly a unique, first-of-a-kind component that was particularly prone to shorting.

The possibility of slip ring failures resulting from placing opposite electrical polarities on adjacent brush assemblies was known at least as early as the summer of 1977 to other projects within the contractor's organization. Furthermore, failures of slip ring assemblies due to shorting between brushes had been experienced by the prime contractor on slip ring assemblies used by other programs. That the Seasat organization was not fully aware of these potential failure modes was due to a breakdown in communication within the contractor's organization.

In addition to this small, though fatal, breakdown in communications, the failure to give the slip ring assembly the attention it deserved was due, in large part, to an underlying program policy and a pervasive view that Seasat's Agena bus was a standard, well proven piece of equipment that had been used on other programs. In actuality, however, three major subsystems—the electrical power subsystem, the attitude control subsystem, and the data subsystem—were substantially modified for use on Seasat's Agena bus. So firmly rooted was this principle of using a "standard Agena bus" that, even after the engineering staffs of both the government and the contractor were well aware of the final uniqueness of their bus, the words, and the associated way of doing business, persisted to the end.

The point of view that the Seasat bus was flight proven, standard equipment proved to have far-reaching consequences. It became program policy to minimize testing and documentation, to qualify components by similarity wherever possible, and to minimize the penetration into the Agena bus by the government. It led to a concentration by project management on the sensors, sensor integration, and the data management system to the near exclusion of the bus subsystems. Important component failures were not reported to project management, a test was waived without proper approval, and compliance with specifications was weak. The component that failed—the slip ring assembly—was never mentioned in the briefing charts for either the Consent to Ship meeting or the Critical Design Review.

The Failure Modes, Effects, and Criticality Analysis that was conducted for the electrical power subsystem did not consider shorts as a failure mode and thus did not reveal the presence of single point failure modes in the system nor provide a basis for the development of a full complement of safing command sequences that could be used by the flight controllers in responding to anomalies in the power subsystem. A lack of clarity and rigor in the operating requirements and constraints documents for the power subsystem of the bus, together with this lack of safing command sequences, prevented the flight controllers from having all the tools they needed to do their job. The flight controller for the power subsystem was also new to his job at the time of the failure and thus was not sufficiently knowledgeable of the system he was controlling. While no action of the flight controllers contributed to the failure, they did fail to follow the prescribed procedures in response to the information available to them at the time of the failure.

The advantages of using standard, well proven equipment in terms of both cost and mission success are well recognized. But the experience of Seasat illustrates the risks that are associated with the use of equipment that is classified as "standard" or "flight proven." The uncritical acceptance of such classifications by the Seasat engineering staff submerged important differences in both design and application from previously used equipment. It is therefore important that thorough planning be conducted at the start of a project to fully evaluate the heritage of previously used equipment and to establish project plans and procedures that enable the system to be selectively penetrated.

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I – INTRODUCTION

On October 9, 1978, the Seasat spacecraft failed some 105 days after launch into orbit. To identify the nature of this failure, and to determine the underlying causes that led to its occurrence, the Deputy Administrator of NASA established a Seasat Mission Failure Investigation Board on October 16, 1978. A copy of the Board's Charter, of the letter of authorization to the Chairman and of the membership of the Board is presented in the Appendix of this report.

The Board initiated its investigation on October 23, 1978. The identification of the failure mode was accomplished through an analysis of available flight data and examination of the build history, test results and qualification reports on the relevant spacecraft subsystems and components. Material assistance was provided to the Board in this task by the engineering staffs of both the Jet Propulsion Laboratory (JPL), the responsible NASA installation, and the Lockheed Missiles and Space Co., Inc. (LMSC), the principal industrial contractor. Some valuable research that had been conducted by both LMSC and by the Sperry Flight Systems Co. also contributed to this failure analysis.

To determine the underlying causes of the failure, the Board examined all relevant documentation related to the history of the Seasat program and conducted extensive discussions with the staff and management of LMSC and JPL.

These investigations permitted an identification of a most likely failure mode and some of the principal technical and management actions that led to its occurrence. This report presents the results of the Board's investigation in response to the charge of its Charter.

II – THE SEASAT MISSION AND ITS SPACECRAFT

The Seasat Project was a proof-of-concept mission whose objectives included demonstration of techniques for global monitoring of oceanographic and surface meteorological phenomena and features, provision of oceanographic data for both application and scientific areas, and the determination of key features of an operational ocean dynamics monitoring system.

To fulfill these objectives, the Seasat sensor complement comprised a radar altimeter (ALT), a Synthetic Aperture Radar (SAR), a Seasat-A Scatterometer System (SASS), a Scanning Multichannel Microwave Radiometer (SMMR), and a Visual and Infrared Radiometer (VIRR). All of these sensors except the SAR operated continuously; telemetry from them, as well as from all engineering subsystems, was sent in real-time when over a ground station and recorded on a tape recorder for later transmission to provide data for a full orbit. SAR data had to be transmitted in real-time, without the use of the onboard recorder, to specially equipped stations because of its high data rate. The normal duty cycle for the SAR was 4 percent.

The five sensors were integrated into a sensor module that provided mounting, thermal control, power conditioning, telemetry, and command support to the instruments. The second major element of the spacecraft was an Agena bus which provided attitude control, electrical power, telemetry and command functions to the sensor module. In addition to these on-orbit functions, the Agena bus also provided injection stage propulsion and guidance to orbit. The spacecraft was 3-axis stabilized with all sensors earth pointing and is shown in its on-orbit configuration in Figure 2-1. To provide near global coverage, the spacecraft was injected into a 790 kilometer, near circular orbit with an inclination of 108 degrees and a period of approximately 101 minutes. Design lifetime was one year on orbit, with expendables provided for a three year life.

The sensors were provided by various NASA Centers. The sensor module, the Agena bus and the integration of the sensors, sensor module and Agena bus into a spacecraft was provided by LMSC under contract to JPL.

Responsibility for Seasat project management, mission planning and direction, mission operations and experiment data processing resided at JPL. The Goddard Space Flight Center (GSFC) provided network support and spacecraft orbit and attitude determinations; use was therefore made of the existing Spaceflight Tracking and Data Network (STDN), the NASA Communications (NASCOM) network, and the Project Operations Control Center (POCC) that are operated by GSFC.

To place this failure review in a proper perspective, it is noted that the Seasat spacecraft operated in orbit in a generally satisfactory maneuver for over 3 months and provided a large amount of scientific data. The sensors represented a significant advance in technology and their integration into the sensor module, a large engineering challenge. In addition, Seasat also required the creation of significantly enlarged capabilities in the acquisition and processing of flight data. That the important and significant technical and engineering advancements were achieved is a tribute to the skill and dedication of all who were associated with this program.

The Seasat spacecraft was successfully launched on June 26, 1978, and thus operated for 105 days until the failure occurred on October 9, 1978. During this time in orbit, the spacecraft operation was generally satisfactory with considerable data being obtained from all of the sensors. Three significant anomalies were experienced during the life of Seasat in orbit, one involving sun interference in the attitude control system scan wheels, one caused by a sticking thermostat in a sensor heater circuit, and one in which the spacecraft suffered an abnormally low bus voltage for several orbits. Because of a possible relationship of these latter two anomalies with the failure of October 9, 1978, they were specifically investigated by the Board and are discussed herein.

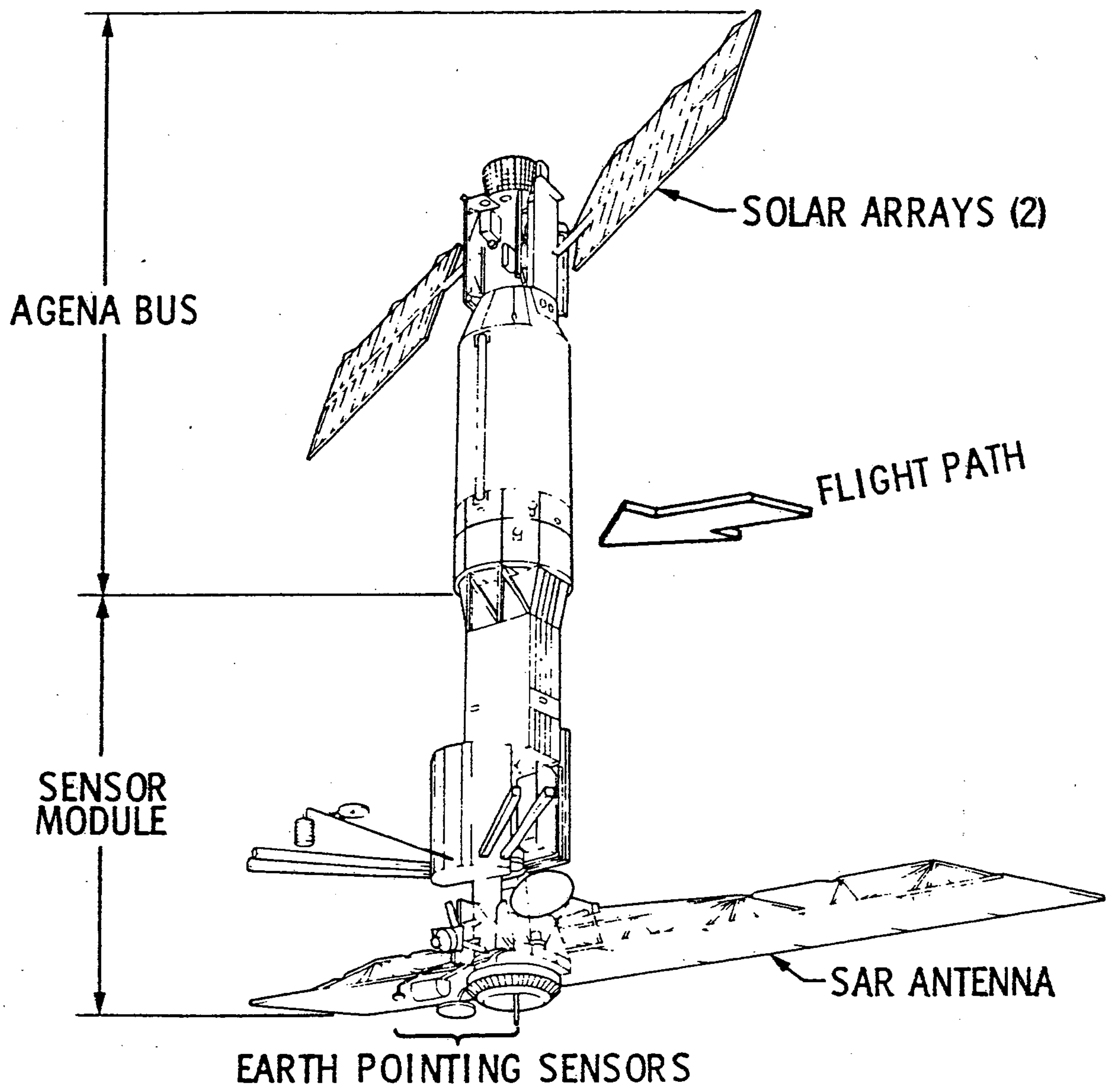

Figure 2-1. On-Orbit Configuration of the Seasat Spacecraft

III – THE FAILURE

The first indication of a spacecraft failure available to the flight controllers occurred when the Santiago STDN station (AGO) acquired the spacecraft on Rev 1503 at approximately 03:30 GMT on October 10, 1978 (11:30 PM EDT, October 9, 1978 at GSFC). At that time, indications of abnormal battery currents, battery voltages, solar array temperatures and unregulated bus voltage were noted. Approximately 13 minutes later, at 03:43 GMT, loss of signal (LOS) occurred at AGO. Because these data were not completely understood at the time, the next STDN station at Orroral (ORR) was called up to receive additional data. Thus, an additional 9 minutes of flight data starting some 17 minutes after LOS at AGO were obtained. The last communication from the spacecraft occurred at LOS ORR about 04:08 GMT; all subsequent efforts to make contact with the spacecraft were unsuccessful.

On October 11, 1978, JPL project management was advised by personnel at the European Space Agency Oak Hanger Tracking Station at Farnborough, England (UKO), that they had obtained flight data from the spacecraft during its pass over their station. Although not a part of the operational Seasat network, this station routinely received data from the Seasat spacecraft for their own purposes. This UKO data on Rev 1503 covered the period from 03:00 GMT to loss of effective data stream at about 03:13 GMT and was made available at JPL for failure analysis on October 12, 1978. As it turned out, this UKO data contained the event of the failure and identified the time of the failure at 03:12:02 GMT, some 6 minutes after the spacecraft entered an eclipse and about 1 minute before the loss of data. A ground track of the flight path of the spacecraft on Rev 1503 over these three stations of interest is presented in Figure 3-1.

Post-event flight data available for failure analysis thus comprised approximately 1 minute of data from UKO, 13 minutes of data from AGO, and 9 minutes of data from ORR. Obviously, the most valuable data for purposes of identifying the failure mode was the 1 minute of data between the initial event of the failure and loss of data from UKO.

Subsequent examination of these data revealed that a massive and rapidly progressive failure occurred in the electrical power subsystem. Accordingly, the identification of the most likely failure mode involved, in addition to the analysis of the flight data, an analysis of the thermal behavior of the spacecraft, an examination of all of the build paper, qualification status reports, test history, as-built drawings and photographs, discrepancy reports, related failure reports and other documentation related to the various components, wiring, assembly and checkout of the electrical power subsystem. Taken altogether, this information was sufficient to identify a most likely failure mode without the necessity of any post event failure mode testing.

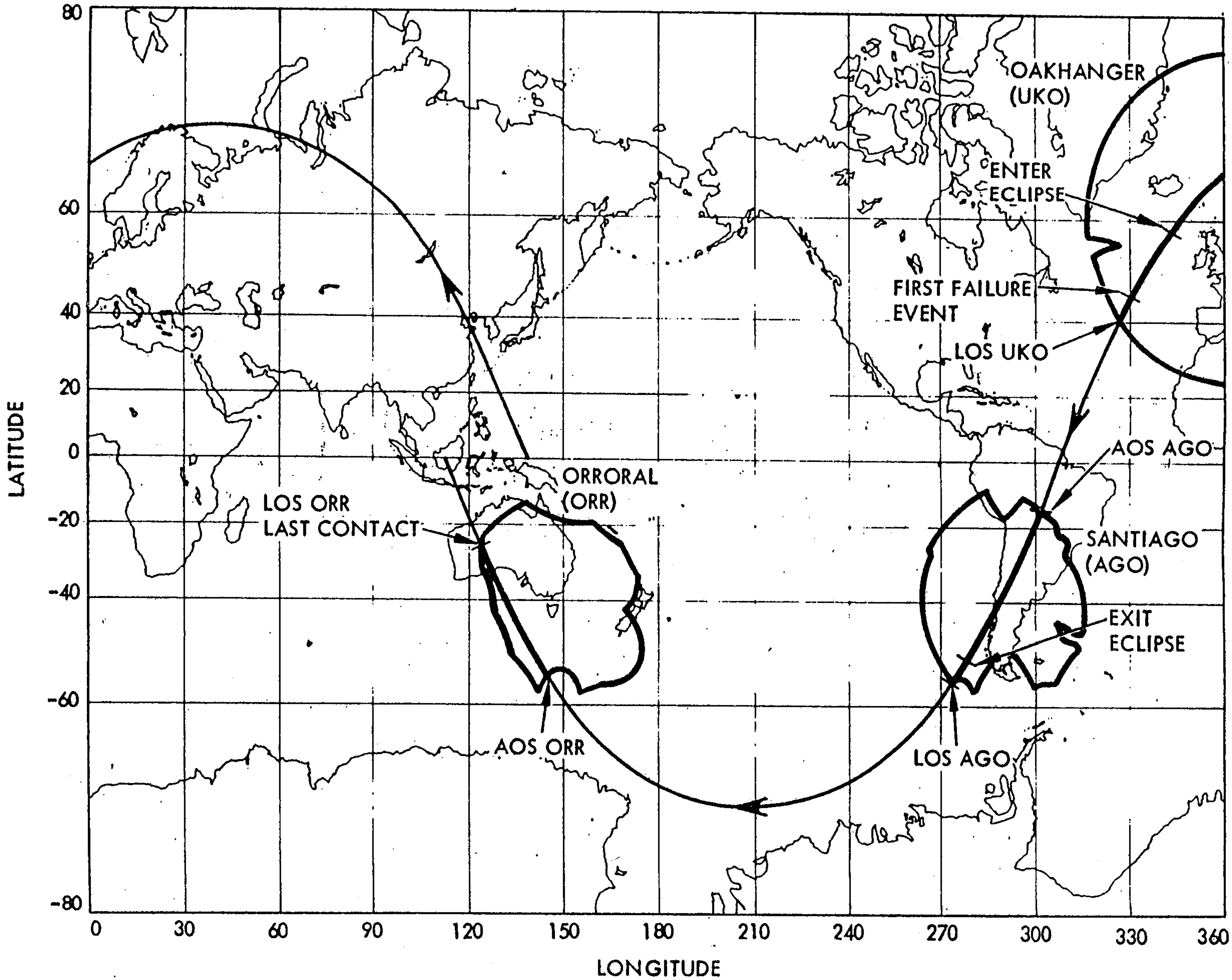

Figure 3-1. Ground Trace of Seasat Revolution 1503

IV – PROGRAM HISTORY AND MANAGEMENT

Environment

The Seasat program was conceived and initiated in a period of transition in the philosophy of management of NASA programs following the Apollo program. Apollo, and to varying degrees other NASA flight programs, were characterized by extensive test programs, large formal documentation systems, and comprehensive and frequent technical and management reviews. A large in-house staff was required in order to implement this approach. The high cost of conducting space programs in this mode severely constrained the future uses of space. During the final phases of the Apollo program, NASA management accordingly instituted a policy aimed at reducing the cost of carrying out space missions. This policy was aggressively pursued by the highest levels of management.

A Low Cost Systems Office was established in Headquarters to oversee a standardization program and to encourage the use of existing hardware. This program included the development of "standard" components as well as a multimission spacecraft.

A major emphasis was placed on shifting work from "in-house" to "out-of-house" in consideration of reducing the NASA manpower base. Design-to-cost techniques and cost benefits of heritage through the use of hardware and software developed for other programs were subjects to be addressed at each step in the approval cycle.

Management Philosophy

The basic philosophy of the Seasat program was thus established in an environment in which management emphasis was shifting from one of demonstrating a national capability to operate reliably in space to one of reducing the cost of utilizing space. Design-to-cost was a fundamental tenet of the Seasat project definition. A cost estimate of $58.2 million was established as a target cost at the end of the feasibility study phase in mid-1973 and was imposed as a design-to-cost ceiling in December 1973 by NASA management. Any overruns were to be offset by descoping the mission content.

In attempting to define a program which would both satisfy the user community and live within the ceiling cost, the concept of making maximum use of proven existing hardware and software was adopted early in the program planning phase. This in turn provided for a reduction in design and development effort and in the size of the in-house staff needed to monitor the activity.

These were key elements of the management philosophy which influenced the structure and conduct of the program.

Program Planning

Feasibility Studies (Phase A) – Feasibility for the Seasat mission was established in 1973 through three studies conducted by the JPL, GSFC, and the Applied Physics Laboratory (APL) of the Johns Hopkins University. These studies were aimed at meeting the set of user requirements generated at a series of meetings held in the first half of 1973 among NASA and representatives of the governmental, commercial, and institutional communities of users of ocean dynamics data. With the user requirements as a basis, the feasibility studies examined the Seasat mission from an overall systems viewpoint, including a review of instrumentation and possible spacecraft (bus) approaches to accommodate the instrumentation.

Subsequent to the submission of the Phase A studies in July 1973, a joint NASA/User Study Task Team was formed to review the Phase A studies, integrate the results, and provide technical and programmatic guidance for more in-depth Definition Phase studies.

As a result of this review, the Task Team recommended a Baseline Mission which included a complement of the five sensor types that actually ended up flying on Seasat.

Based upon cost estimates prepared by the Phase A study participants, the Task Team recommended a target cost of $58.2 million for the Baseline Mission. This included the cost of the spacecraft bus and instruments, the launch vehicle, and tracking and data acquisition. An Alternate Payload Mission of reduced capability, excluding the synthetic aperture radar, was also recommended for further study with a target cost of $43.2 million.

There was some discussion in the Seasat Study Task Team Report (October 1973) of the use of an existing bus to minimize cost. The idea, however, was addressed with some skepticism. While it was believed that the use of subsystems with a high degree of inheritance from existing programs was desirable and possible, it was not clear at that time that an existing bus could be adapted economically.

Definition Studies and Preliminary Design (Phase B) – Definition Phase Studies of the Baseline and Alternate Payload Missions recommended by the Seasat Study Task Team were conducted from November 1973 to the summer of 1974. Wallops Flight Center (WFC) managed the Definition Phase Study of the Baseline Mission which was conducted by APL. The JPL, assisted by various aerospace companies familiar with earth satellite design, conducted the Definition Phase Study of the Alternate Mission.

In December 1973, NASA management adopted the $58.2 million figure recommended by the Task Team as a *not to exceed ceiling* for the Seasat Baseline Mission. The efforts of the Definition Phase Study participants were accordingly intensified to develop the most economical satellite system possible that would best suit the user requirements within the cost ceiling.

The GSFC declined to participate in the Definition Phase activity as they had serious doubts as to their ability to structure a full Baseline Mission within the design-to-cost ceiling.

With the stimulus of the design-to-cost ceiling, and management emphasis on the maximum use of existing subsystem hardware, the JPL Definition Phase Group proposed the idea of building a spacecraft system comprising two major elements: a sensor module designed specifically for Seasat, and a spacecraft bus based on an existing, flight proven bus developed for other Air Force or

NASA programs. The JPL viewed the results of the Phase A studies as indicating that the requirements of the sensors could be satisfied by standard support subsystems for attitude control, power, structures, thermal control, etc. On the other hand, the area of largest uncertainty was seen to be the definition of the sensor's operating capabilities, data requirements, and sensor system integration. It was therefore proposed that if a suitable spacecraft bus were available, the design and development effort could be concentrated on the sensors and their integration with a sensor module that could then be mated to the bus through a mechanical-electrical interface.

The JPL entered into four $15,000 study contracts with aerospace companies (Boeing, General Electric, Lockheed, and TRW) that had existing spacecraft designs with capabilities in the range of Seasat requirements to evaluate the concepts that: (1) There are existing buses that could be used, without modification, to supply the necessary support functions for the sensor payload, and (2) new-design functions could be incorporated in a separate module along with the sensors and thereby reduce the systems development task to a *sensor system* development task. The studies were conducted from November 15, 1973, to March 30, 1974. The sensors were described to the study contractors as they were developed on December 15, 1973, with updates as appropriate until the end of these studies.

It was concluded as a result of these studies that basic sensor support requirements could be satisfied by the existing spacecraft bus designs studied with "no major changes," although "minor modifications" were acknowledged to be required. It was contemplated, for example, that minor modifications would be required of the attitude control, power, and temperature control subsystems. Telemetry, tracking and command subsystems were reported to be "off-the-shelf" designs, but required significant modification. It should be noted that the contractor bus studies were concerned almost solely with mission performance requirements. The reports did not sufficiently define the subsystem design or component selections to provide a basis for an adequate penetration of heritage. The JPL Definition Phase Final Report nevertheless concluded that the existing bus approach had significant cost, schedule and risk advantages, and permitted a concentration of development efforts on the sensor system.

Mid-term reports in May 1974 of the JPL and WFC/APL Definition Phase study groups demonstrated that neither the Baseline nor Alternate Payload Mission was achievable within the $58.2 million ceiling. WFC/APL's estimate for the Baseline Mission, which included an in-house designed spacecraft, was $85.2 million. At this point in time WFC/APL adopted the sensor module/existing bus concept that JPL was pursuing. JPL's mid-term estimate for the Alternate Payload Mission using the existing bus concept was $65.9 million.

Both JPL and WFC/APL searched for ways to descope the project in order to stay within the cost ceiling. Each group performed a number of iterations wherein sensor performance and sensor combinations were varied in order to decrease the cost and yet meet the basic user requirements.

A final presentation of the JPL and WFC/APL Definition Phase studies to NASA Headquarters management in August 1974 resulted in a reduced baseline payload at the $58.2 million ceiling which eliminated the microwave radiometer and combined the altimeter and scatterometer into a single instrument, but which retained the synthetic aperture radar, as well as the visual and infrared radiometer.

Program Implementation

In the summer of 1974, at the completion of the Definition Studies, the concept of using an existing spacecraft bus was firmly established. An analysis by the Program Manager stated that:

> "The most important Seasat-A Program objective is: To place most of the program money on the project peculiar sensors and direct sensor support subsystems. To accomplish this, the selection of an exisiting Low Cost spacecraft bus supplying power, structure, attitude control, orbit adjust, etc., is essential; and selection of the specific existing spacecraft to be used for Seasat-A must necessarily be made early in the program so that sensor and sensor support subsystems design and integration planning can take advantage—Low Cost—of the peculiar attributes of the specific existing spacecraft."

The program had been submitted to the Congress in January 1974 for approval as an FY 1975 "new start," with a run-out cost of $58.2 million. Congressional action was completed in September 1974. Responsibility for project implementation was not assigned to a NASA field installation at that time and thus no project office was established for another two months.

Project Initiation – In November 1974, JPL agreed to accept the management of the Seasat Project at the $58.2 million ceiling with the reduced payload resulting from the Definition Phase descoping effort discussed above. The JPL acceptance was, however, made subject to a review in early 1975 of the project scope and user community reaction to it. (Letter from JPL Director to NASA Associate Administrator for Applications, November 7, 1974.) The above JPL letter further recognized the fiscal constraint of the $58.2 million ceiling by suggesting that the projected April 1978 Seasat launch be "treated like a unique planetary opportunity . . . in order to hold costs and . . . achieve the proper schedule/cost relationships." On November 25, 1974, the Program Approval Document (PAD 61-600, Sub PAD No. 61-630-655) was signed, officially designating JPL as the Project Management Center to be assisted by WFC and APL. The sensor procurements were to be accomplished by designated NASA Centers and furnished to WFC/APL for integration with the Sensor Module. The Sensor Module was then to be supplied to JPL to be integrated with the bus.

The Seasat Program represented a major departure from the predominant JPL experience base. It was the first applications mission for which JPL was responsible, the only earth-orbital mission since Rangers 1 and 2, the first mission in which operations were to be conducted at a location other than JPL, and only the third mission in which an out-of-house contractor was given the overall systems integration responsibility.

The "existing bus" concept became an official requirement of the Seasat Project as the authorizing language in the PAD specified that "a(n) 'existing' bus be competitively acquired by JPL from industry with minimal modification." The next sentence of the PAD specifies that the " 'standard' bus module" be procured on a Fixed Price type contract. The descriptions "standard bus module" and "existing bus" were apparently intended to be equivalent.

The Seasat Project Office was formed in November 1974, with a staff comprised of nine people. The Financial Manager and Contract Negotiator were collocated with the Project, but were organizationally responsible to other Divisions and are not included in the nine. The Project Staff was also assisted by personnel from the JPL Technical Divisions which varied from 26 to 88 personnel over the life of the Project. Out of the approximately 15 people involved in the JPL Definition Phase Study, only one was carried over into the Project Staff. Approximately five more people from the Definition Phase Study Group continued to give support to Seasat from the JPL Technical Divisions.

In negotiation of the staffing level by the Project Office with each of the JPL support divisions, it was made clear that the JPL role was one of monitoring the contractor's activity and that maximum reliance was going to be placed on the use of existing contractor management systems and procedures. It was further made clear that technical monitoring would be based on limited penetration of subsystems with emphasis to be placed on the major interfaces. This is manifested in a JPL Interoffice memo (S-75-23) from the Seasat Project Manager to the JPL Technical Divisions supporting Seasat which states in part:

> "The spacecraft system is based on the use of an existing earth orbiting bus modified to meet mission unique requirements. It will be provided on a Fixed-Price-Incentive contract. With this base and without the need for technology transfer from JPL to the contractor, it is believed manpower levels required to manage and monitor the contractor are justifiably less than that necessary on past projects."

There was substantial desire on the part of all concerned to proceed with obligation of the funds that had become available in September 1974, although the Definition Phase Study left much to be desired. Preliminary design efforts were not only insufficient to generate meaningful component selections, schedules, cost estimates, test plans, qualification requirements, etc., which should be developed during a Phase B activity, but also were conducted with respect to a management structure and sensor complement that were significantly different from those ultimately implemented.

During the first few months after the Project Office was formed, a primary concern of the Project Staff was to prepare and issue an RFP for the spacecraft bus and integration support. The RFP was issued on January 13, 1975. Sensor contracts were let about this time and during the next few months the sensors, sensor module and sensor interface became increasingly defined and cost estimates were updated. In addition, reaction of the user community to the reduced capability payload resulting from the design-to-cost constraints stimulated a reassessment of the Baseline Mission. In July 1975, it was decided to increase the baseline payload by adding back the microwave radiometer and a full performance altimeter. These and other program adjustments resulted in increasing the ceiling cost from $58.2 million to $74.5 million. This last figure was considered by both Project and Headquarters management as tight but achievable.

At this time the functional organization of the project was also reassessed. It was decided that the WFC/APL role would be reduced to include supplying the full performance altimeter, imaging radar data link, and system engineering support. The scope of the contract effort was increased to include not only the spacecraft bus, but also the sensor module and its integration with the bus. The sensors being developed by the various NASA Centers were to be Government Furnished Equipment (GFE) to the contractor. This scope change to the contract effort necessitated a re-solicitation. The second RFP was issued on September 10, 1975.

As with the first RFP, LMSC and General Electric were the only contractors that submitted proposals. On November 20, 1975, JPL selected LMSC with which to negotiate for the procurement of the Seasat Satellite System.

The following were among the LMSC proposal strengths identified in the selection process:

(1) The flight experience of Agena as a spacecraft and launch vehicle was found to be very good. The experience was found to comprise over 300 flights the last of which were 98 percent successful. The Agena was therefore found to be an extremely well proven, reliable space vehicle.

(2) Most of the proposed hardware at the assembly level was found to have previous flight experience in that it was identical to previous Agenas.

(3) Fabrication and test of modified designs were found to be based on proven approaches.

For the next few months the efforts of the JPL Project Staff and JPL Technical Support Divisions were primarily directed toward the fact finding and negotiation process in connection with LMSC's proposal. The contracts between JPL and LMSC for the Seasat Satellite System were signed on February 12, 1976. A Fixed-Price-Incentive (FPI) contract called for LMSC to furnish an Agena bus appropriately modified to meet Seasat unique requirements. Under a Cost-Plus-Award-Fee (CPAF) contract LMSC was to provide the sensor module and system integration effort.

As might be observed from the above, the pace of business in the Project Office was fast and hectic from the day it was initiated. In fact, the Project Office was, in the words of the Project Manager, in the position of "playing catch-up since day one." To recapitulate the foregoing, the project had been approved by the Congress in September of 1974, prior to assignment of project responsibility to JPL. In early November 1974 JPL management had accepted the project assignment with a $58.2 million cost ceiling subject to a review in early 1975 of the project scope and to the user community reaction to it. The Project Office was accordingly established in November 1974 with a minimum of infusion of personnel from the Definition Phase (Phase B) study team. It became a first order of business for the Project Office to issue an RFP for the bus and integration contracts. An RFP was accordingly issued in January 1975 even though JPL was aware of several problems with the program as structured including the concerns of the user community with the proposed payload content. In the spring of 1975, two major perturbations resulted in increasing the sensor complement and reassigning the organizational responsibilities which necessitated the issuance of a new RFP. The program was redefined, recosted, and approved by Headquarters in mid-1975, and the new RFP issued in September.

While the initiation of the program was delayed some 8 months as a result of this restructuring, the launch date was delayed only 6 weeks, thus compounding an already difficult schedule. The restructuring also invalidated much of the already meager Phase B work.

The major technical and management challenge of Seasat was to develop a set of advanced sensors, to integrate them successfully into a sensor module, and to create an expanded capability in data acquisition and processing within a tight schedule. The Project, therefore, proceeded at a fast pace, and at times under hectic conditions. There was little time to reflect, to explore alternative approaches to problems, and to thoroughly develop plans and procedures. That the Seasat sensors and the data system were apparently very successful is a tribute to the dedication and skill of the project staffs of both the government and the contractors.

Project Implementation – At the time of contract go-ahead, the sensor complement, the overall budget and the launch date were firmly established as was the use of an existing "flight proven" Agena bus to be procured under an FPI contract with minimal government involvement and monitoring.

The principal technical and engineering challenges were viewed to be in the sensor development, systems integration and the development of a data handling system. The JPL in-house support had been sized for a limited penetration of the Agena bus and the personnel advised of this policy.

The contracts with LMSC contained 15 deliverable data items which required JPL signature. These included specifications for the spacecraft bus, as well as plans for qualification, configuration control, reliability, systems test, operations, etc. For implementation of these plans, and consistent with Program policy, the Project Office was largely dependent on the contractor for configuration control, mass properties control, compliance of component specifications with systems specifications, subcontractor compliance, and qualification certifications. The Project Office was solely dependent on the contractor for assurance that the flight and test history of similar components used on other programs was brought to the attention of the proper Seasat contractor personnel and acted upon.

There is no indication that the JPL technical or management personnel did not maintain free and open communications with their LMSC counterparts. To the contrary, they appeared in general to be well aware of the status of the project on a timely basis to the extent of knowledge of their counterparts. Their efforts to selectively focus attention on the areas of likely difficulty were hampered by the lack of a more thorough Phase B study which, among other things, should have explored the modifications required to the Agena bus in greater detail.

Penetration into the bus was also inhibited as a result of the cost growth of the program. At the time the LMSC contracts were signed in February 1976, the ceiling cost for Seasat was budgeted at $74.5 million. As of June 30, 1978, the cost, including scope changes and overruns was projected to be $94.0 million. Major reasons for the escalation included significant scope changes to the CPAF contract for the sensor module and system engineering as well as a $6.5 million (40%) overrun by LMSC on this contract. Another major item was an increase in the cost of the launch vehicle from $5 million to $12.6 million in order to accommodate weight growth and to avoid potential launch vehicle buffeting.

As a result of this cost growth, there were frequent admonitions from the Headquarters Program Office during the course of the project to hold costs, including, among other things, suggestions to cut back or even eliminate JPL penetration of the Agena bus. The extent of infusion of this pressure on the project prompted JPL management to express a concern to NASA management in September 1977 regarding the proper balance between performance and cost goals. Although cost control was recognized as important, JPL was concerned that the reaction of the Project Staff would result in an overemphasis of cost at the expense of hardware quality and adequacy. In point of fact, however, no evidence was revealed that would indicate the Project Staff ever consciously sacrificed hardware quality and adequacy for the sole sake of costs. Nevertheless, cost control was a constant concern resulting in an active and sometimes tension-filled dialogue between the JPL Project Office and Headquarters Program Office. The resulting atmosphere was less than conducive to optimum project management.

In this same vein, a word must be said about the LMSC contract overruns. Both the FPI and the CPAF contracts incurred cost overruns. On the FPI contract, the overrun amounted to approximately $145,000 above the contract ceiling. LMSC accordingly earned no fee under this contract and absorbed the $145,000 as a loss. The CPAF contract to date is known to have an overrun of approximately $6,500,000, or 40 percent. Final figures may increase this slightly. Despite their poor profit position, no evidence was found of any instances where LMSC sacrificed technical performance solely to reduce costs. In particular, no connection could be found between the flight failure and the poor profit environment that resulted from either of these contracts.

In retrospect, the environment that was created by holding so firmly to the concept of using an existing bus produced a disarming level of confidence in the heritage of the Seasat Agena bus. The Project Staff, to be sure, was aware of the various design changes to the Agena bus that were made to accommodate Seasat unique requirements. In spite of this awareness, the concept of the existing or standard bus, as it became interchangeably referred to, was so far embedded in Program policy, that this awareness became submerged. Changes to the bus were not perceived significant enough to violate the principle of the Standard Agena Bus. The mindset was that to the extent subsystems were not identical, they all used previously flight proven components or components close enough to be qualified by similarity. As a result of this environment, JPL personnel were not sufficiently sensitized to exploring the differences between the existing standard Agena bus and the one used for Seasat.

As it turned out, the Agena Bus finally used for Seasat carried a unique power subsystem, attitude control subsystem, and telemetry subsystem. As far as orbital functions are concerned, all that was really standard was the basic structural configuration and the primary propulsion system.

For example, with regard to the power subsystem, the solar arrays were the first application of a rotating array on the aft end of the Agena, the slip ring assembly had no applicable flight experience, and the solar array drive electronics (SADE) had undergone extensive redesign.

In spite of the significant differences in both design and application from prior experience, the basic concept of the standard, existing bus persisted. For example, the Pre-Launch Mission Operations Report (No. E-655-78-01) dated June 23, 1978, (three days prior to launch) described the Seasat Satellite System as comprising two major hardware elements:

- "A standard support bus which is based on the Lockheed Missiles and Space Company (LMSC) Agena . . . ."
- "A sensor module . . . ."

The Agena, the Report further notes, "has had over 300 flights and has previously achieved orbit operations of over two years."

V – THE ELECTRICAL POWER SUBSYSTEM

General Description

The Electrical Power Subsystem (EPS) was designed to provide power at 28 ±4 VDC to the spacecraft subsystem and sensors using solar arrays and rechargeable batteries. Figure 5-1 is a block diagram of the power subsystem and Figure 5-2 shows the general location of the power subsystem on the spacecraft. A simplified schematic diagram of the EPS is presented in Figure 5-3. The basic design philosophy was to provide functional redundancy, i.e., interconnected systems, but both necessary to provide the required capacity to handle the full load. Capability for component isolation (removal from circuit), cross-strapping, charge control (automatic and manual), and system protection via by-pass functions was provided by commandable relays.

Energy Source and Storage Subsystems – The primary energy source for the spacecraft was the Solar Array (SA) which consisted of two modules (wings) mounted on either side of the aft rack (Figure 5-2). With the vehicle in the normal orbital attitude and the SA deployed in the X-Y plane, the wing axes lay 40° ahead (toward the direction of flight) of the +Y axis and 40° behind the –Y axis. The wings tracked the sun through 360° about this axis using error signals generated by sun sensors located on each solar array wing. Signals generated by the sun sensors were processed in the Solar Array Drive Electronics (SADE) which provided power to control the array drive motor speed. During periods of eclipse the array was driven at a fixed angular rate by signals from the SADE. In addition, the rotation direction and rate could be controlled by commands. Each wing contained 11 panels. The average power output of the SA varied during the life of the spacecraft due to the seasonal intensity of the sun, the angle to the sun (beta angle), eclipse periods, and various factors which degrade the power output capability of the solar cells. During full sun, the SA supplied power to all the loads as well as for charging the two Type 40 nickel-cadmium batteries. The batteries supplied the total spacecraft load requirements during eclipse and supplied the surge loads when they exceeded the instantaneous capability of the SA.

Power Distribution and Control – Power, instrumentation, and sun sensor signals were brought in from each SA wing through a Slip Ring Assembly which was coupled with the Drive Motor Assembly. A Charge Current Controller (CCC) was provided for each battery to control charging rate as a function of voltage and/or battery temperature. Manual charge control was provided by commands that operated relays to disconnect various segments of the solar array from the batteries. Power to the spacecraft subsystems and sensors was distributed, fused, and controlled in the Main Power Control and Distribution Unit (MPCDU). The MPCDU also contained the main power transfer switch that connected the batteries to the 28 VDC bus, the power transfer relay, pyro test switch, and inverters, control relays and other vehicle power functions. The SA panel disconnect relays and the CCC backup manual charge control relays were also located in the MPCDU.

Power Subsystem Requirements – Figure 5-4 is a summary of the Seasat power requirements.

The total average power requirement of 711 watts assumed a 100 percent duty cycle for all sensors except the SAR where a 4 percent duty cycle was assumed. Figure 5-5 is a comparison of the power required with the power available for one year of spacecraft operation. During parts of the eclipse periods, the power available provided for the average power required but with no margin for contingency. The pre-launch plan required operational restriction of the sensor duty cycle during certain beta angles in the event adequate power was not available to maintain battery charge. However, the standby power (all sensors off) was only 100 watts less than the full load (approximately 600 watts compared to the full load of 711 watts) because considerable heater power was required for the altimeter and scatterometer even in a standby mode.

Power Subsystem Instrumentation – Over 208 parameters of the EPS were monitored by telemetry. Many of these were relay on/off, SA deployment monitors and similar measurements. The key instrumentation channels pertinent to this review are shown in Figure 5-6. This table gives the range and sampling rates for these critical measurements. The locations of temperature sensors are of importance to this investigation and are discussed below. Battery temperature was monitored by thermistors bonded to straps that connected various cells of the battery. Solar array temperatures were monitored by sensors bonded to the outboard panels (panel 11) of each array. The temperature of the SADE box was measured by a thermistor bonded to a printed circuit card located near the center of the box.

Power Subsystem Operation

All transfer of power and signals between the rotating solar arrays and the spacecraft's electronics was through a slip ring assembly associated with each array. Both power and returns from each of the 11 panels on each array, as well as the array temperature and sun sensor inputs, were connected through this assembly. Figure 5-7 is a simplified schematic of the primary power control system (solar arrays, batteries, charge current controllers, and power switching logic).

Main Power Control and Distribution Unit – The control and distribution of the unregulated power to the various subsystems aboard the spacecraft were accomplished in the MPCDU. Fuses located in the MPCDU provided the power interface for sensors and other subsystems, except for a few mission critical loads that were supplied unregulated power directly from the 28 volt main bus. Power for the main bus was provided both directly from the solar array panels and also from the batteries through isolation diodes. Control of the spacecraft bus voltage to within the specified 28±4 VDC range was accomplished during sunlight by open-circuiting sections of the solar array as the batteries were charged. During eclipse, the bus voltage was to be maintained by proper planning of loads so that the battery voltage would remain above 24.5 VDC. The MPCDU also contained the solar array panel interconnection and by-pass relays, enabling individual solar array sections to be connected in a large number of configurations to provide flexibility in charging the spacecraft batteries and supplying the load demands. A bus was provided from the output of battery 2 to supply power directly to the pyrotechnic loads on the spacecraft; this pyro bus could be disconnected via ground command through the K3 relay. The MPCDU also provided power to the remainder of the power subsystem components, including the SADE and the CCC. Instrumentation was included in the MPCDU for the measurement of unregulated bus voltage and current, structure (ground) current, battery and pyro bus voltage, and relay status monitors. As shown in Figure 5-7, panels 9 and 10 on both the +Y and −Y array were connected directly to the main bus when their respective relays, K28 and K34, were closed. On this schematic, these relays are shown in the status mearsurement (open) position, but in normal operation were closed. The unnumbered square symbols on the schematic simply indicate connection to the main bus. Provisions existed for connecting all the remaining panels directly to the main bus and by-passing the batteries in event of battery malfunction. This capability was provided through commandable relays K32, 33, 38, and 39. In normal operation, however, output from panels 1, 2, 3, and 11 were connected to the batteries through the K1 relays and output from panels 4, 5, 6, 7, and 8 were connected to the batteries through the K2 relays, both located in the Charge Current Controllers. Thus, considerable operational flexibility was provided in the design of the power subsystem.

Charge Current Controller (CCC) – Battery charging was controlled by a separate CCC for each battery. The CCC consisted basically of two relays (K1 and K2) that controlled battery charge by connecting or removing sections of the solar array to the battery terminals dependent upon the battery's voltage and temperature. A maximum of four array panels were connected to the battery by the K1 relay and a maximum of five array panels were connected to the battery by the K2 relay. The number of solar array panels connected through each relay was commandable by means of interconnecting relays (K29, 30, 31, 35, 36, and 37) located in the MPCDU. Each of the CCC relays (K1 and K2) had its own driver circuitry, which determined the appropriate battery voltage level at which to remove sections of the solar array from the battery charge circuit. This voltage disconnect set point was dependent upon temperature such that the higher the battery temperature, the lower the voltage set point. The set point characteristics of battery voltage versus temperature for the K1 and K2 relays are shown in Figure 5-8. The design of this circuitry was predicated on a knowledge of battery state-of-charge (SOC) as a function of battery voltage, with K2 designed to open at a 90 percent SOC and K1 designed to open at a 95 percent SOC. Correspondingly, each relay's driver circuitry reconnected the appropriate solar array panels to the battery when the battery voltage fell below a pre-set level that was dependent upon battery temperature. As a back-up safety feature, each relay was also controlled by battery temperature alone: K2 was designed to open at approximately 93°F and K1 at approximately 98°F. Closing of K1 and K2 with decreasing temperature was automatic. Another safety feature was the interleaving of panels from each array to each charge controller to provide some charging capability for each battery in case of failure of a complete solar array. This interleaving of panels from each array is shown on Figures 5-3 and 5-7.

Normal Operation – The power subsystem was designed so that during full sun the solar array provided power to all of the loads as well as for charging the batteries. Both the K1 and K2 relays were normally closed allowing the batteries to charge. When the batteries achieved approximately 90 percent of full charge the K2 relays opened, reducing the charging rate. The K1 relays were set to interrupt charging when essentially full charge (98 percent) voltage was obtained. When the spacecraft entered an eclipse, the batteries would assume the load, would start to discharge, and the voltage would drop. First K1 would close and, as the batteries were further discharged, K2 would close. As the spacecraft emerged from the earth's shadow the solar array became illuminated, charging was initiated, and the cycle repeated. The system was designed for the batteries to reach essentially full charge just prior to going into eclipse. During the eclipse, it was planned that the loads would remove about 12.5 percent of the rated capability of the batteries.

Provisions for Malfunctions – In the event of anomalous automatic performance of the CCC, additional controls were provided so battery charging could be controlled by manual commands. The following is a list of possible malfunctions and corrective actions.

1. *Relays Fail Closed:*
   Relay driver circuit fails to open K1 or K2 relays. Separate override commands for each K1 and K2 relays were provided which closed a latching relay in the MPCDU which applied power to the selected CCC relay through back up circuitry. If sending the override command failed to open the malfunctioning relay, the proper "SA panel off" relay command would disconnect the appropriate panels from the charging circuit.

2. *Relays Fail Open:*
   Relay driver circuit fails to remove power from K1 or K2 relays; i.e., relays stay open as battery voltage drops. Power to the offending CCC could be removed by sending the proper "CCC off" commands. If removing power from the CCC did not correct the anomaly, a back up parallel latching relay could be activated to by-pass the K2 relay.

In the event of a battery failure, its CCC K1 and K2 relays could be commanded open by the override command removing the battery from the system. The solar array panels assigned to that battery could then be connected directly to the bus by closing the proper "diode by-pass relays" (K32 and 33 for battery number 1 and K38 and 39 for battery number 2).

Any failed solar array panels could be disconnected in groups. Panels 2, 3, and 11; 4 and 5; 6, 7, and 8; 9 and 10 could be disconnected by opening the appropriate interconnecting relay. Panel 1 could be disconnected by opening K1, which also removed panels 2, 3, and 11. The sensors and other loads could be commanded off individually.

Electrical Power Subsystem Flight History

The EPS generally performed as designed from launch up to the time of the failure on October 9, 1978, except for two anomalies, one on July 6 and one on August 28.

Heater Thermostat Anomaly – On July 6, 1978, the radar altimeter high mode thermostats cycled rapidly and there were indications that relay contacts intermittently stuck in the closed, heater power-on, position. The heater thermostats for the SASS also locked on but this was not discovered until the heater bus was cycled to relieve the high temperature condition on the radar altimeter. Subsequent investigation revealed that the thermostat was rated for an upper current limit of 3.0 amps while in operation the radar altimeter high mode heater was operated at 4.7 amps, and the SASS heater was operated at 6 amps. Laboratory tests duplicated the failure and revealed that the failure mode was one of material transfer at the contact points. This transfer of material changed the gap between points and also caused the thermostat to stick. This failure necessitated manual rather than thermostatic control of heater power which, because of the less efficient operation, increased the average heater power consumption. This increased load was a contributing factor to the low voltage anomaly on August 28, but did not result in any damage to the electrical power subsystem.

Bus Low Voltage Anomaly – On August 28, 1978, the altimeter transmitter turned itself off between active station passes because of a low voltage condition. This undervoltage developed because of a slow depletion of battery power as a result of the loads being slightly in excess of solar array capability. Figure 5-9 shows the unregulated bus voltage for several revolutions prior to the low voltage anomaly. These data indicate that, at least for six orbits, the batteries did not reach a fully charged state; i.e., K2 did not open, thus indicating less than a 90 percent full charge. In fact, examination of the data after the failure of October 9 revealed that for three days preceding this low voltage anomaly none of the K1 or K2 relays had opened as they would have if the batteries were charged to a high level. As shown by the data of Figure 5-9 the battery charge was progressively depleted for each succeeding orbit during the eclipse and heater on periods. On revolution 892, with the altimeter off, the battery began to recover its charge. This information was not available to the flight controllers in real time because the low voltage point for these orbits did not occur during station passes. General information on the status of battery charge was, however, available to the flight controllers for several days from K2 relay status data. The lowest bus voltage experienced during this period was approximately 21.8 VDC; while the maximum depth-of-discharge cannot be accurately determined from the available data, it is estimated that the observed minimum battery voltage would be equivalent to that expected with almost 100 percent of rated capacity removed from a 22 cell battery. This problem probably was due to a combination of the extra heater loads and insufficient understanding of the state-of-charge of the batteries. As a result the operation of the altimeter and SAR was restricted until normal operations were resumed on September 7, 1978. No equipment damage was sustained as a result of this anomaly and the EPS performed normally in subsequent orbits.

EPS Status at Failure – From launch to the time of the failure on Rev 1503, no failure of any component of the power subsystem was detected. Anomalous behavior such as discussed above did not result in any significant degradation of the power subsystem. All indications are that the batteries were in a healthy state and all elements of the electrical power subsystem were operating properly prior to the failure.

Slip Ring Assembly

A more detailed description of the slip ring assembly is provided at this point, as a thorough understanding of this component is central to the development of the following sections of this report.

General Description – The slip ring assembly flown on Seasat was produced by the Poly-Scientific company of Blacksburg, Virginia. Its design, development, and parts fabrication had been completed for another earlier program that had been cancelled prior to flight. This design had required 14 power transfer rings, which were 0.17 inch wide, and 38 signal rings which were made 0.07 inch wide. For the Seasat mission, a larger number of power rings were required but their individual electrical loads were small enough to be accommodated by the narrower rings. This design was adapted for Seasat by simply using both wide and narrow rings for power. For Seasat, 30 of the slip rings were used for power transfer. The flight units were assembled from existing parts inventory. Slip ring serial number 1001 was assigned to the +Y array and serial number 1002 was assigned to the –Y array.

A photograph of a slip ring assembly of the Seasat design that has been partially disassembled to reveal the rings and brushes is presented in Figure 5-10. As can be noted, 26 of the brush assemblies were mounted on a brush block in the bottom half of the assembly. These brushes interleaved with the other 26 brushes that were attached to an upper brush block shown removed from the assembly in Figure 5-10.

For reference in this investigation, the assignments of the 52 slip rings as connected for Seasat are listed in Figure 5-11. This figure identifies the solar array panel to which each ring was wired, and specifies polarity and function of each ring. All brushes were wired to exit connectors. Brush wires that were not functionally used were terminated at these connectors. All slip rings were connected to wires routed through the inside of the slip ring and drive motor shafts to the solar array assembly strut shown in Figure 5-12. Wires from the rings that were functionally used were spliced to leads from the applicable solar array panel. The wires that were not used were cut off approximately 8 inches from the exit of the slip ring assembly and each end of the wire was dead ended with shrink tubing. These cut-off wires were placed back into the wire bundle to assure non-interference with solar array deployment. The numbering system for the rings was 1 through 52 in sequence starting with the drive end of the assembly. The cable bundle at the drive end of the assembly contained the wires from the slip rings and was routed through the solar array drive assembly strut and outward to the solar array harness as shown in Figure 5-12.

Significant Design Features – The array normally turned slowly with respect to the bus in a sun tracking mode. It turns in either direction, may remain stationary, or be slewed at rates of up to 15° per minute while transmitting power and signals. The electrical load through the slip ring assembly was approximately 25 amperes of current at voltages up to 33.0 volts or about 2.3 amperes output from each solar array panel through its assigned slip ring(s) in the assembly.

The slip ring rotor assembly consisted of individual slip rings on a fluted shaft, held in place by a potting compound that provided structural support and positioning to the rings and their soldered leads as well as dielectric insulation. The two types of brush and spring assemblies in the design flown on Seasat are shown in detail in Figure 5-13 (a and b). From these figures it is clear that the wide brush assembly, when mounted, straddled the slip ring assembly in a manner to bear on its mating slip ring from opposite sides. This resulted in having essentially a full overlap between the interleaving wide brush assemblies that were mounted alternately from the top and bottom brush holders in the slip ring assembly. The narrow brushes shown in Figure 5-13b sprung out when placed over the rotor so that a vertical gap of 0.04 inch typically resulted at assembly between the tips of narrow brushes. An appreciation of brush spacings when fully assembled may be seen by the photographs in Figure 5-14.

In order to determine worst case clearances between adjacent brush assemblies, a stack up tolerance study was done by Poly-Scientific at the request of the Board. This analysis showed that mechanical interference between brush assemblies was possible. This analysis verified a judgment by the Board that the design was very congested and prone to trouble when minor discrepancies in assembly, inspection, or operating disturbances occurred.

Materials Used – Space applications of rotary power or signal transmission has been accompanied by a long term, highly specialized development effort. In this effort on slip ring and brush combinations, Poly-Scientific has played a major role and has supplied various types of devices for most of the western world's space missions. From this considerable experience, the coin-silver rings, which contain 90 percent copper, used together with brushes of 85 percent silver, 12 percent molybdenum disulfide, and 3 percent graphite were judged by the Board to be adequate and likely were the best available for the Seasat application. Questions of wear debris are ever present in these devices but actual wear for the Seasat application was expected to be minor compared to the spinning drum type of solar array applications that have been widely used with success for several years' lifetime.

Critique of Slip Ring Assembly – In the rather limited and specialized market that exists for slip ring assembly devices for space applications, the Seasat units were procured from one of the most proven sources available. The material choices were proper and had been well proven in flight. There were adequate numbers of quality personnel in the Poly-Scientific organization (about 10 percent of the 500 person workforce).

The design value for brush contact pressure (5 to 6 psi assuming full contact) appears to have been proper in this application and the assigned tolerances were reasonable. Cramped space and limited access precluded pressure measurement in the preferable fully assembled configuration so the use of tooling fixtures was required. This technique results in less pristine measurements. The electrical load of 2-3 amperes applied to both narrow and wide rings was the same because of the improvised approach to adapt the existing design but resulting current densities were still conservative. The Poly-Scientific personnel, as expected from their experience, were generally aware and sensitive to potential problems from voltage surges, corona, and intolerance to earth ambient exposure or operation when using the Seasat brush material. The slip ring assembly was sensitive to earth ambient operation because the molybdenum disulfide constituent serving as the lubricant in the composite could pick up moisture and exhibits a characteristic termed "filming." This could cause electrical noise during earth ambient operation which would continue for a while in vacuum before the brush-to-ring contact surfaces became reconditioned and the noise was again reduced to acceptable levels.

The overall adaptation of the mechanical design was unnecessarily crowded for the functions it had to perform for Seasat. It was inherently more prone to minor errors in positioning critical parts and to problems resulting from minor disturbances to those parts than necessary, even for a device of equal volume and weight. Poly-Scientific personnel were aware that the design was congested and did recommend relief in length and weight specifications from LMSC during the original development in 1973. This relief was not granted at the time because of program constraints. In the case of Seasat, however, these same restrictions did not exist. All of the slip rings could have been of the narrow type, the dual wiring to more than one ring for a solar array panel lead was not required, and the eight rings that were not connected externally in the flight configuration could have been omitted. In summary, the policy of using existing equipment without modification introduced a significant risk in the Seasat application of the slip ring assembly.

The major fault, however, was not in the design of the device as produced by Poly-Scientific, but in how it was wired into the power subsystem in application. In the case of Seasat, the wiring, as flown, was alternately plus and minus between most of the adjacent brush and ring pairs along the rotor axis in the assembly. The reason given by both LMSC and JPL for this wiring arrangement was that it was done to reduce magnetic moments. Consideration of magnetic moments and means of their compensation are normal design considerations. It is, however, only *within* the slip ring assembly that magnetic moments would dictate a particular wiring sequence, and here the wire routing was unspecified, and by virtue of the assembly process used, was to a degree unknown and uncontrolled. A test failure on another program, covered in Chapter VI of this report, demonstrated the vulnerability of this design to electrical shorts when adjacent brush assemblies are of opposite polarity.

In the electrical area, it was also noted that Poly-Scientific used Teflon insulated wiring whic[illegible] is more prone to cold flow than other popular insulation materials. They did, however, have a [illegible] rationale for its application within the slip ring rotor assembly itself because in the curing [illegible] potted assemblies the temperatures involved tend to crack the alternate insulating materials. For this application, Teflon was judged by the Board to be acceptable because this portion of the assembly was relatively free from sources of wire to metal pressure, being potted in place within smooth channels. For the brush assembly wires, there was some susceptibility to occurrences of wire to metal pressures. For the brush wiring it would appear that a better insulation choice could be made for future applications as no high temperature potting is involved; however, more space would have to be provided for wire routing when the more rigid type of insulated wires is used.

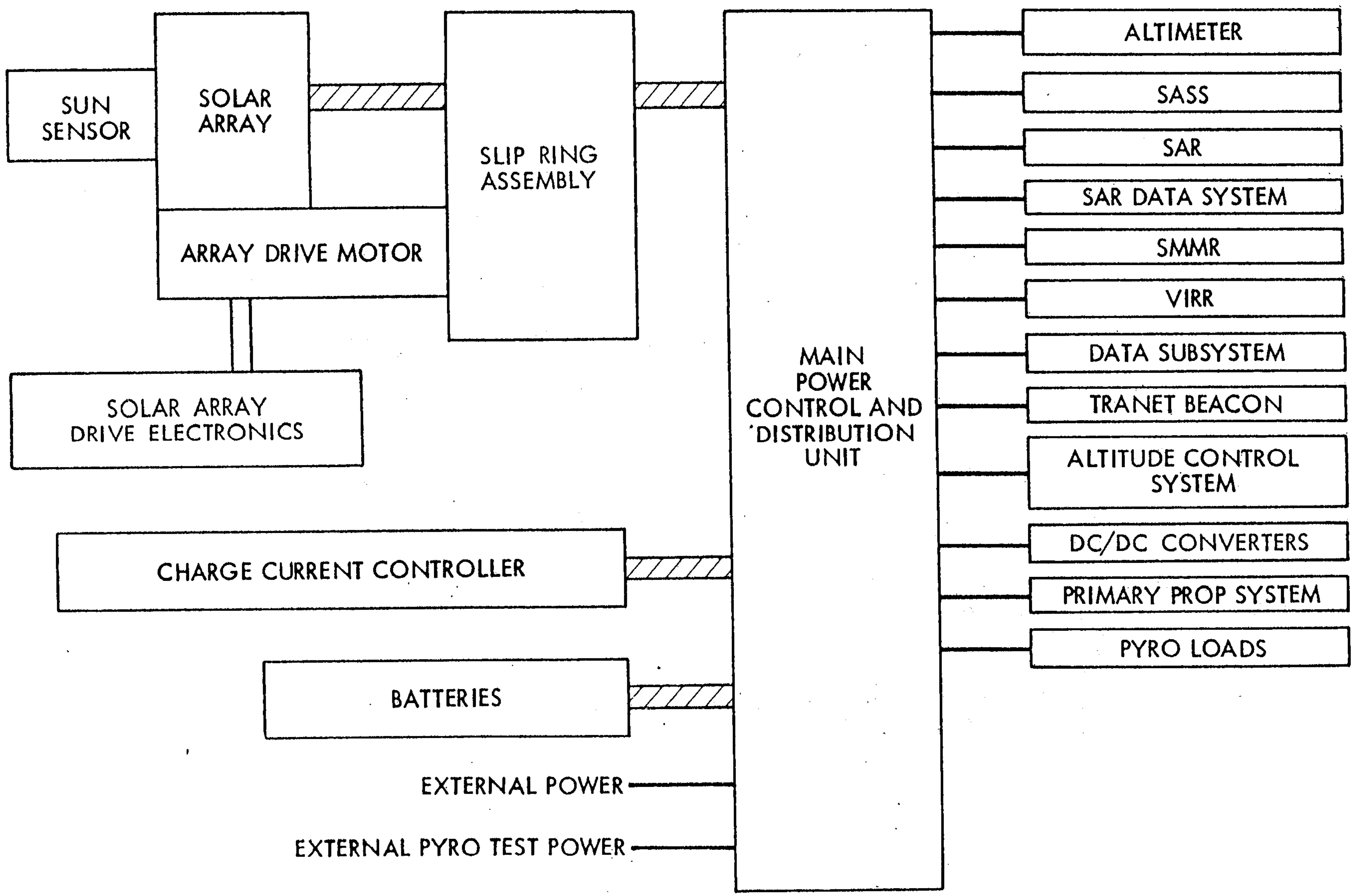

Figure 5-1. Block Diagram of the Electrical Power Subsystem

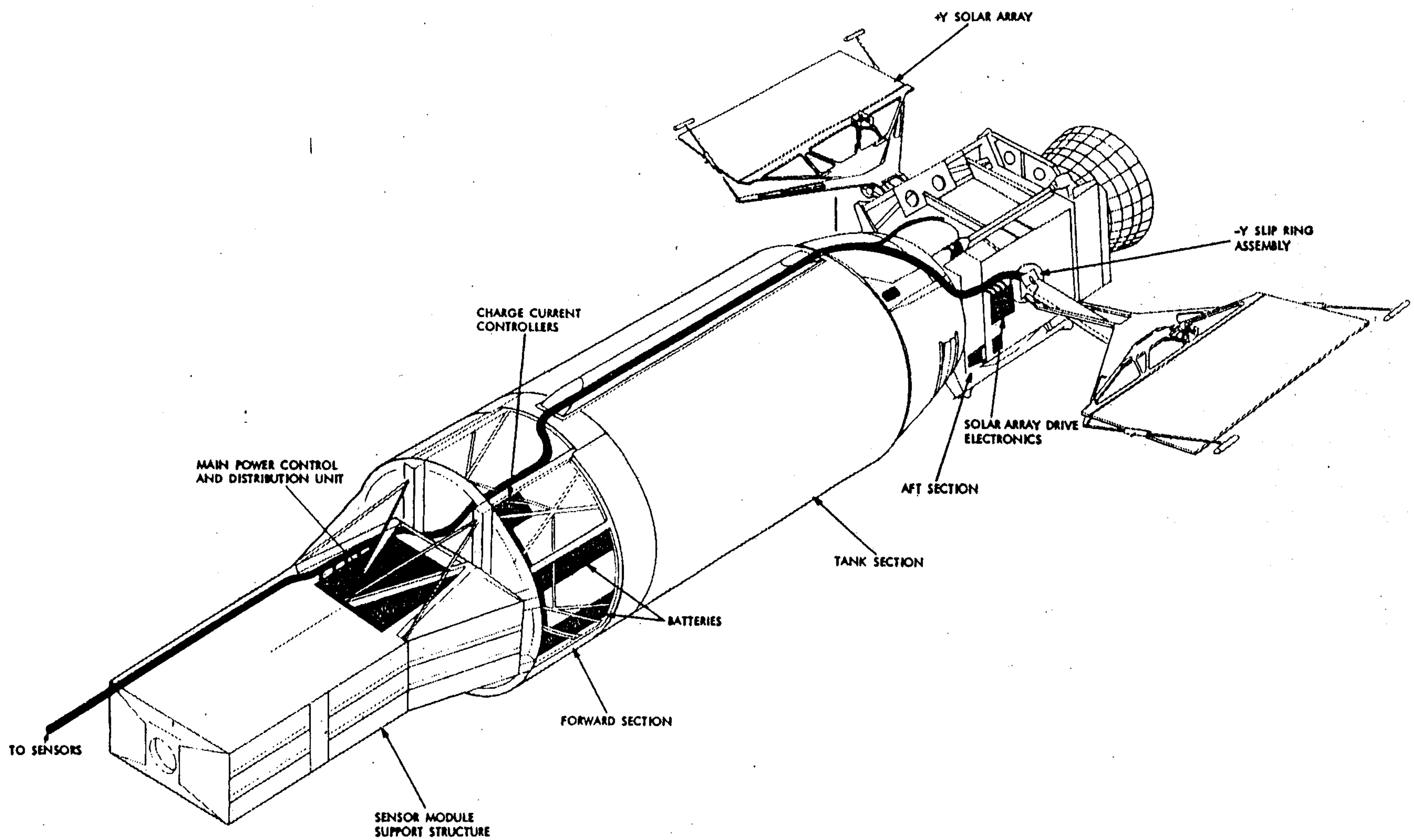

Figure 5-2. Location of Major Components of the Electrical Power Subsystem

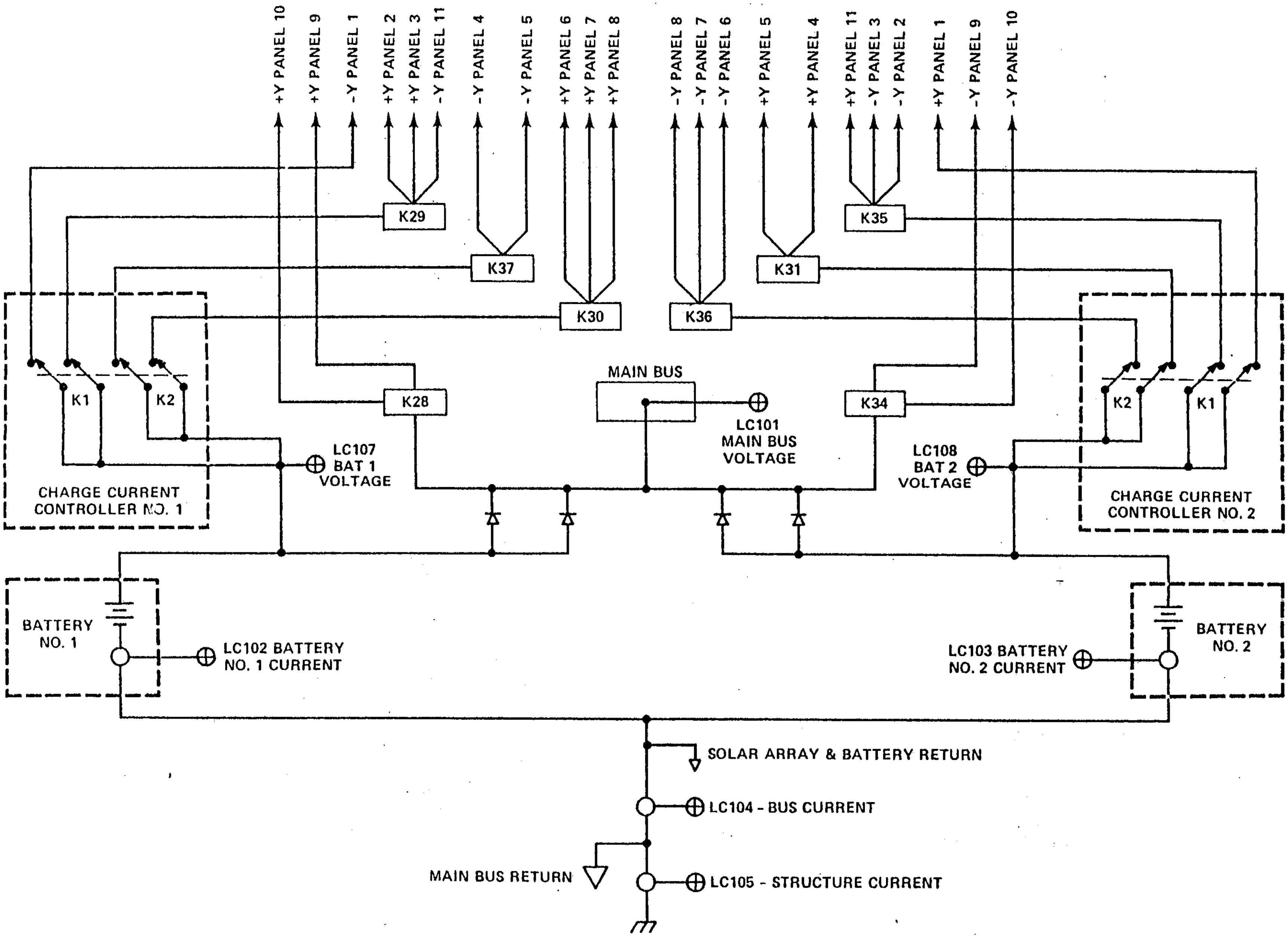

Figure 5-3. Simplified Schematic Diagram of the Primary Power Control System

	AVERAGE POWER, WATTS
• ELECTRICAL POWER SUBSYSTEM	42.4
• DATA SUBSYSTEM	61.2
• ATTITUDE CONTROL SUBSYSTEM	37.2
• THERMAL CONTROL SUBSYSTEM	63.2
• SUBSYSTEM TOTAL	204.0
• SENSORS TOTAL (100% DUTY CYCLE EXCEPT SAR AT 4%)	473.0
• TOTAL EQUIPMENT LOAD	677.0
• DISTRIBUTION LOSSES	13.0
• TOTAL SPACECRAFT LOAD	690.0
• BATTERY DIODE LOSSES	21.0
TOTAL	711.0

Figure 5-4. Average Power Requirements of the Spacecraft

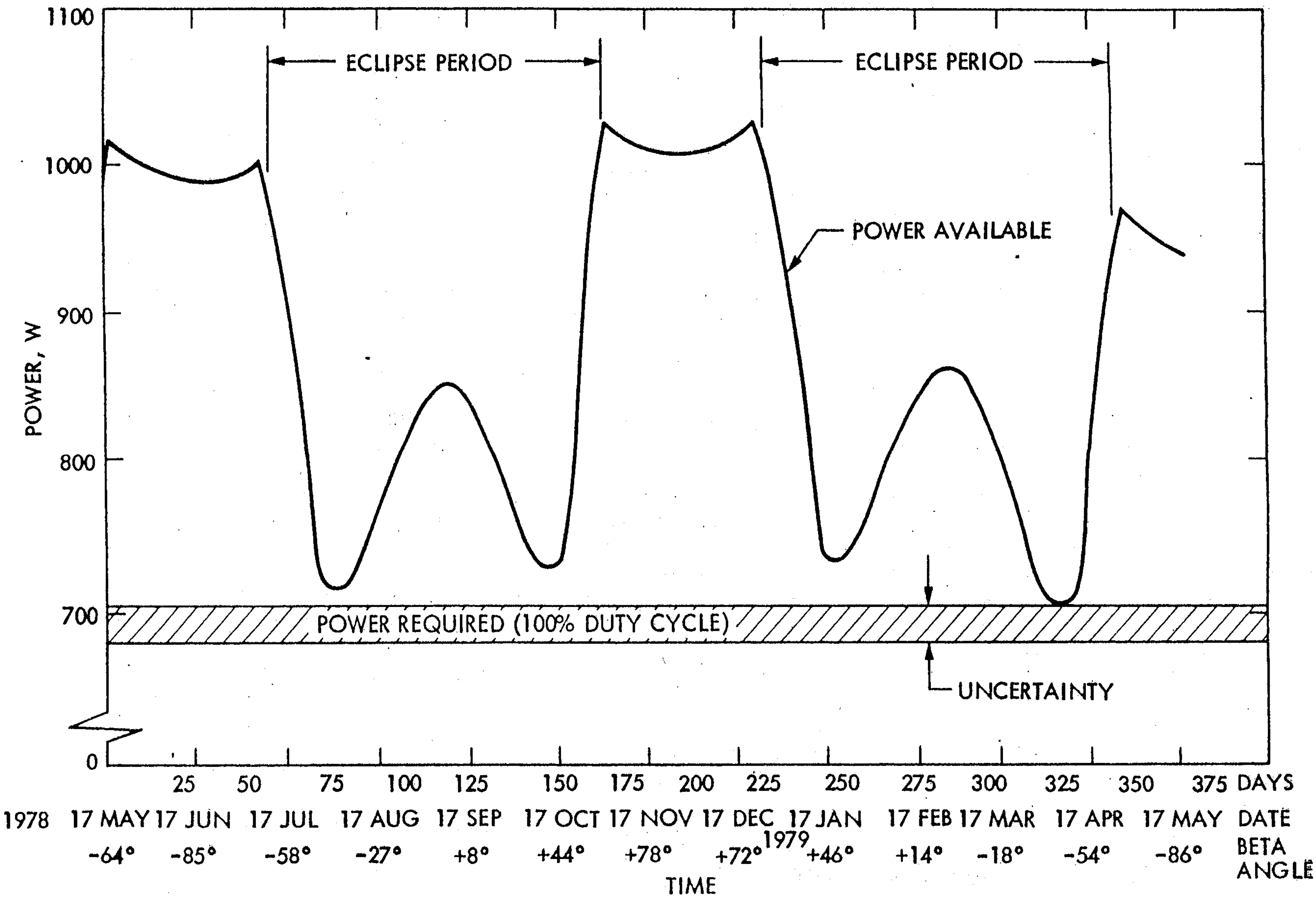

Figure 5-5. Representative Variation of Power Available

CHANNEL No.	TITLE	RANGE	SAMPLES PER SECOND	
			LOW RATE SYSTEM	HIGH RATE SYSTEM
LC-101	UNREG BUS VOLTAGE	0 TO +40 VDC	1	8
LC-102	BAT No. 1 CURRENT	-20 TO +50 A	1	
LC-103	BAT No. 2 CURRENT	-20 TO +50 A	1	
LC-104	UNREG BUS CURRENT	0 TO 100 A	1	32
LC-105	STRUCTURE CURRENT	0 TO 50 A	1	8
LC-107	BAT No. 1 VOLTAGE	0 TO +40 VDC	1	
LC-108	BAT No. 2 VOLTAGE	0 TO +40 VDC	1	
LC-109	BAT No. 1 TEMP	-5 TO +140°F	1	
LC-110	BAT No. 2 TEMP	-5 TO +140°F	1	
LC-122	S/A + Y_A TEMP	-150 TO +250°F	1/8	
LC-123	S/A - Y_A TEMP	-150 TO +250°F	1/8	
LC-124	SADE TEMP	-35 TO +180°F	1/8	
LC-507	CCC No. 1K1	ON/OFF	1	
LC-508	CCC No. 2K1	ON/OFF	1	
LC-509	CCC No. 1K2	ON/OFF	1	
LC-510	CCC No. 2K2	ON/OFF	1	

Figure 5-6. Summary of KEY Electrical Power Subsystem Instrumentation

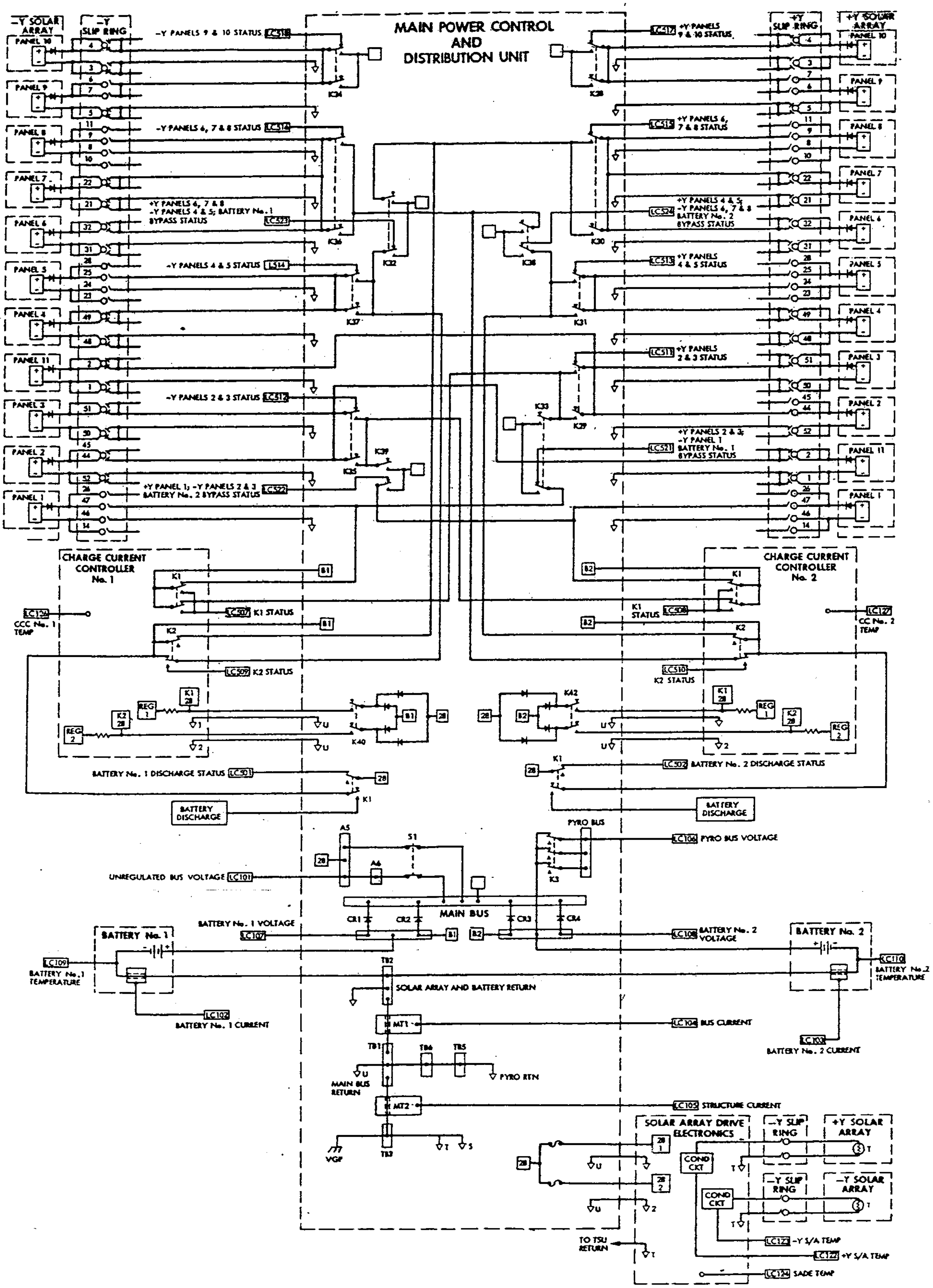

Figure 5-7. Schematic of the Electrical Power Subsystem

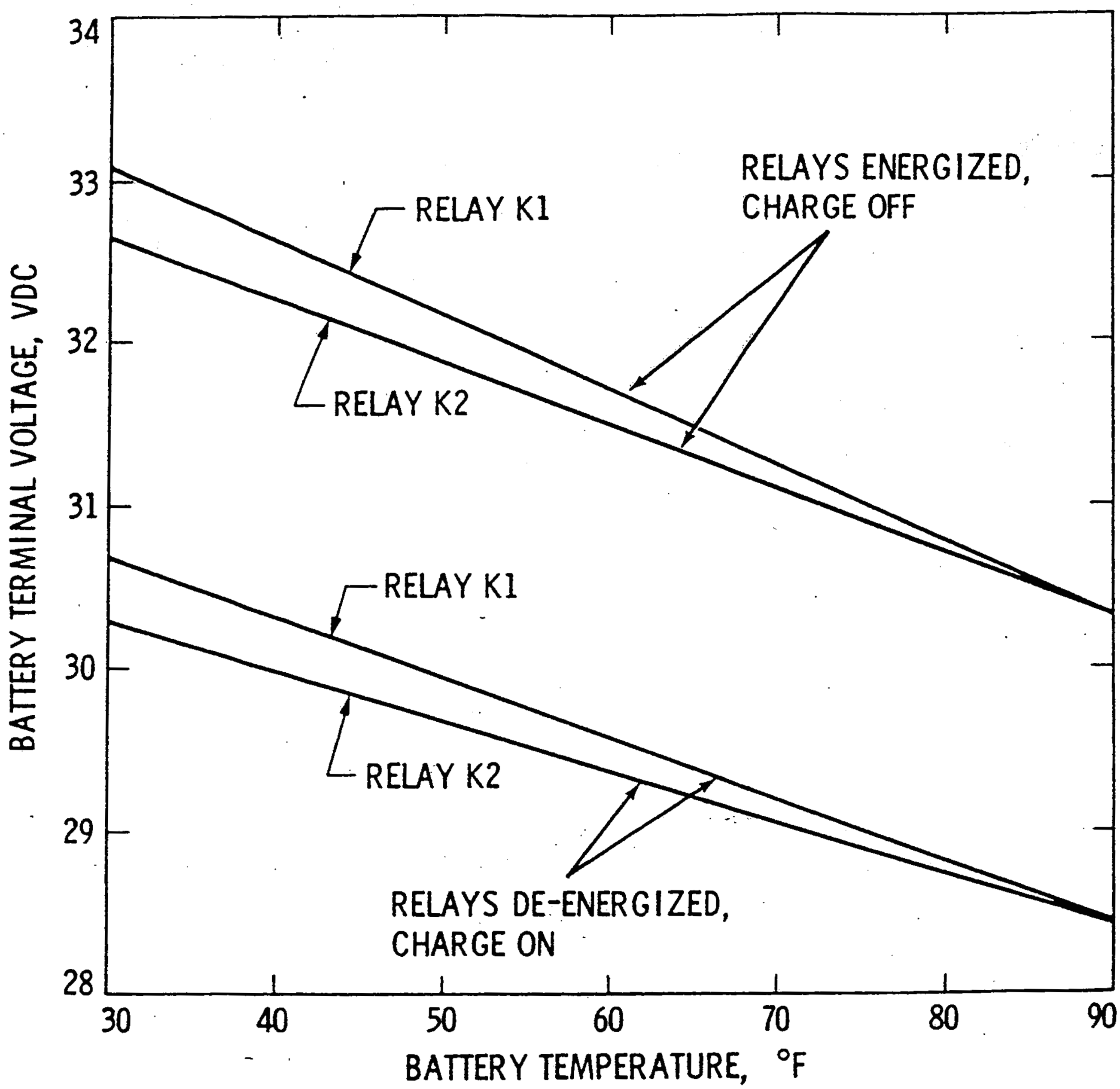

Figure 5-8. Characteristics of the Charge Current Controller

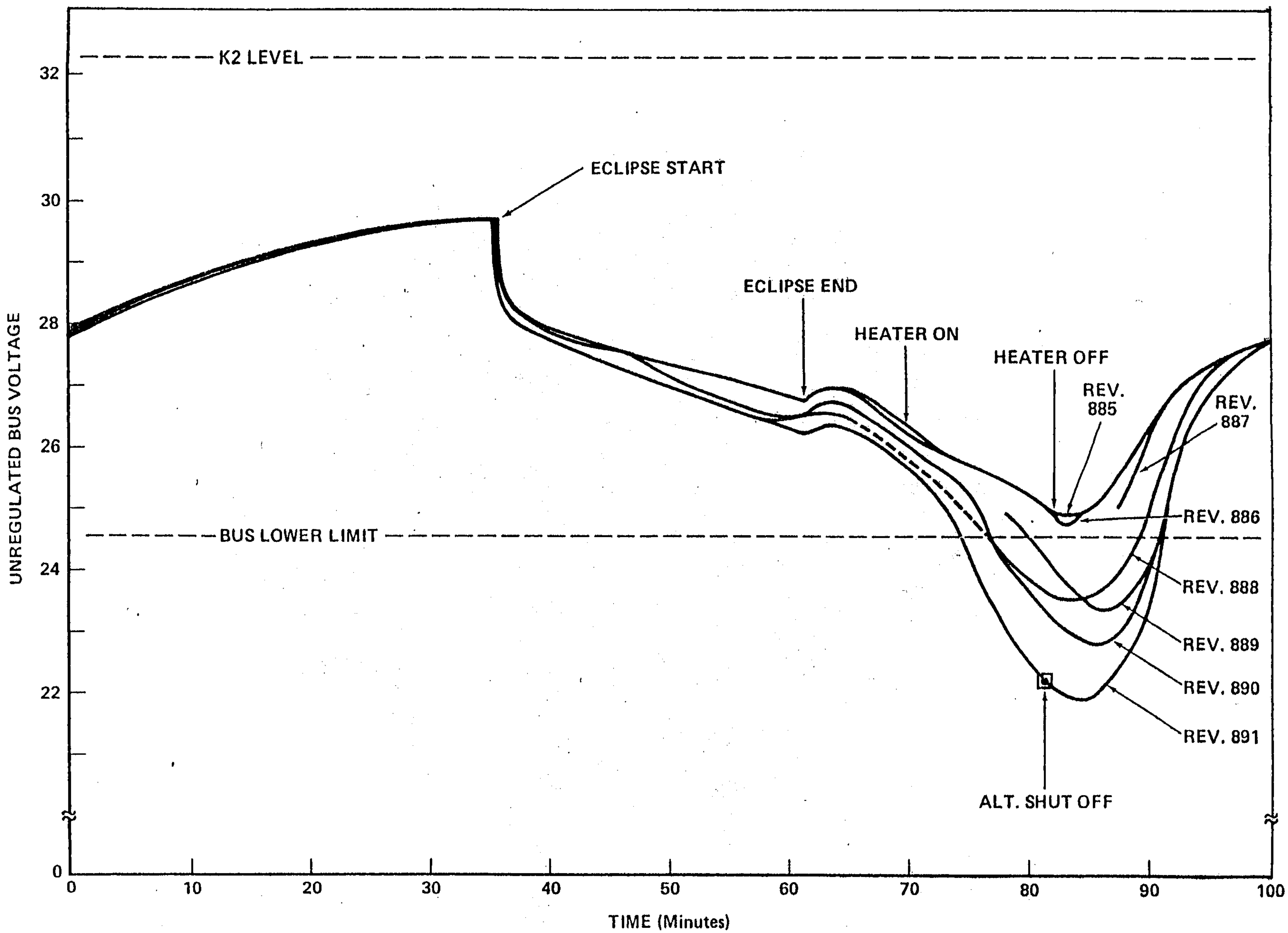

Figure 5-9. Unregulated Bus Voltage During the Low Voltage Anomaly

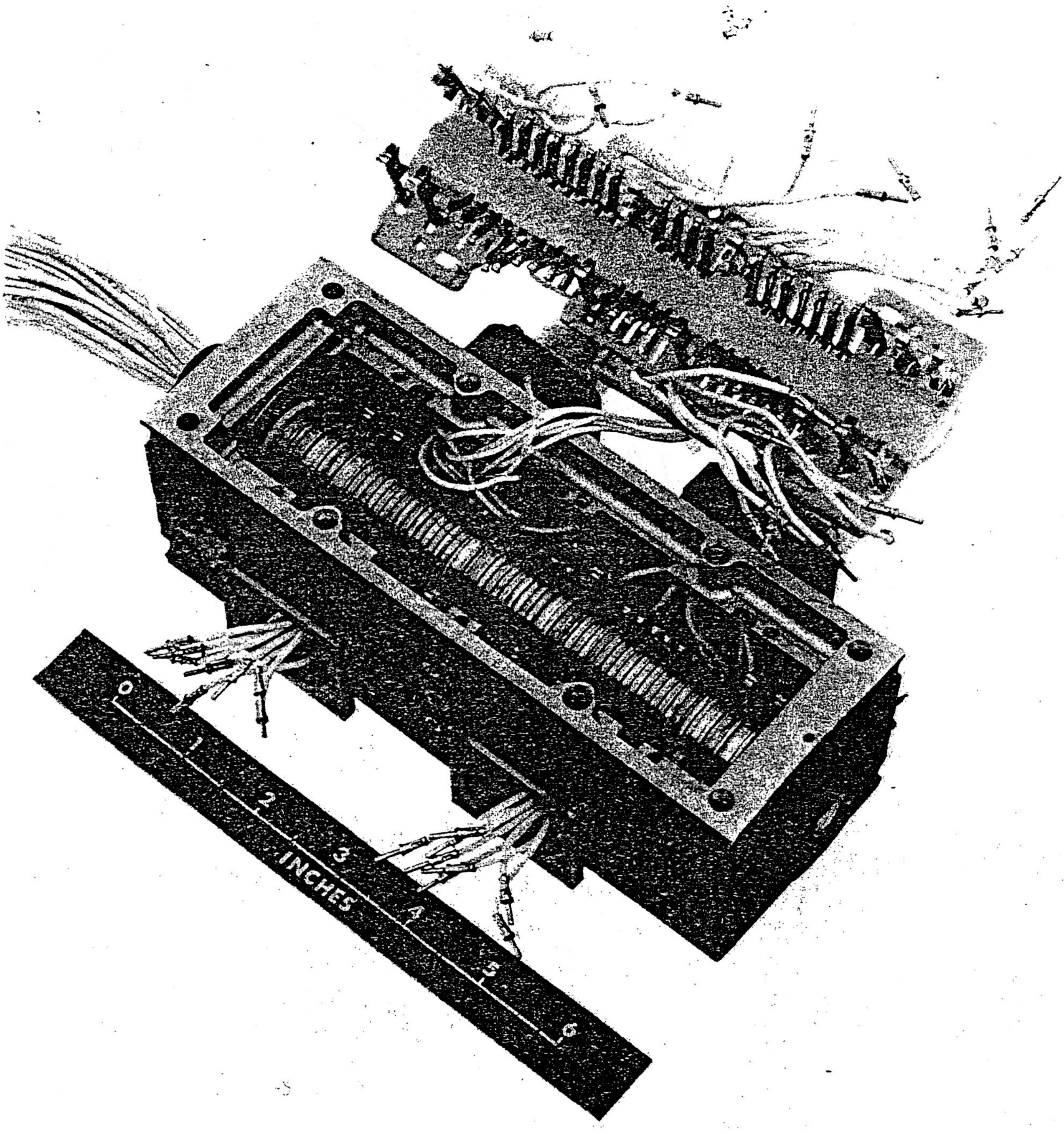

Figure 5-10. Slip Ring Assembly with Top Brush Block Removed

Slip Ring Number	Size Slip Ring	Connection to -Y Panel Number or Function	Polarity
1	Wide	11	-
2	Wide	11	+
3	Wide	10	-
4	Wide	10	+
5	Wide	9	-
6	Narrow	9	+
7	Narrow	9	+
8	Narrow	8	-
9	Narrow	8	+
10	Narrow	8	-
11	Narrow	8	+
12	Narrow	No Connection	
13	Narrow	No Connection	
14	Narrow	1	-
15	Narrow	Thermistor	
16	Narrow	Thermistor	
17	Narrow	Potentiometer	
18	Narrow	Potentiometer	
19	Narrow	No Connection	
20	Narrow	No Connection	
21	Wide	7	-
22	Wide	7	+
23	Narrow	5	-
24	Narrow	5	-
25	Narrow	5	+
26	Narrow	1	+

Slip Ring Number	Size Slip Ring	Connection to -Y Panel Number or Function	Polarity
27	Narrow	Shield	
28	Narrow	5	+
29	Narrow	No Connection	
30	Narrow	No Connection	
31	Wide	6	-
32	Wide	6	+
33	Narrow	Sun Sensor	
34	Narrow	Sun Sensor	
35	Narrow	Sun Sensor	
36	Narrow	Sun Sensor	
37	Narrow	Sun Sensor	
38	Narrow	Sun Sensor	
39	Narrow	Sun Sensor Ground	
40	Narrow	Deploy Mechanical Grd	
41	Narrow	No Connection	
42	Narrow	No Connection	
43	Narrow	Rotor Ground	
44	Narrow	2	+
45	Narrow	2	+
46	Narrow	1	-
47	Narrow	1	-
48	Wide	4	-
49	Wide	4	+
50	Wide	3	-
51	Wide	3	+
52	Wide	2	-

Figure 5-11. Assignment of Seasat Slip Rings to Solar Array Panels and Functions

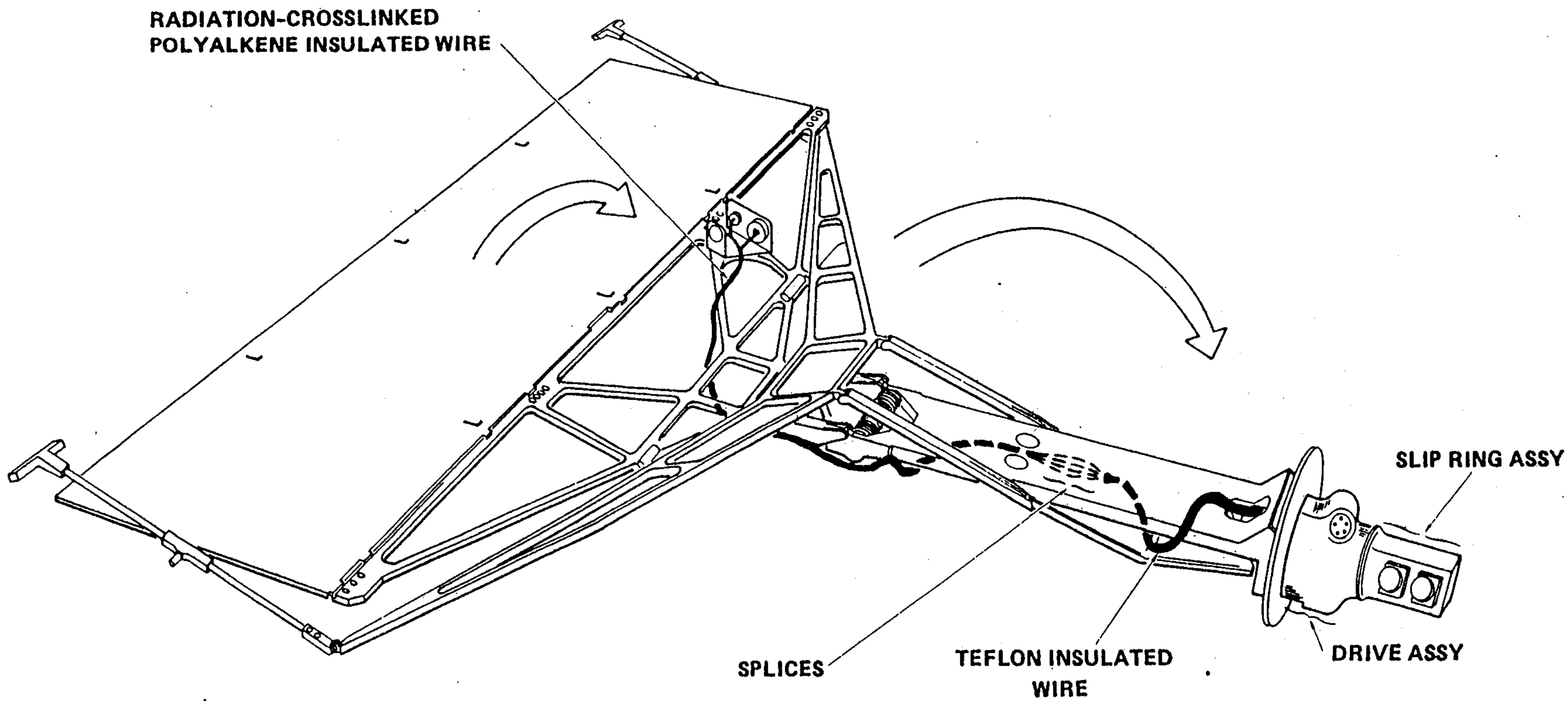

Figure 5-12. Inboard Portion of Solar Array Module

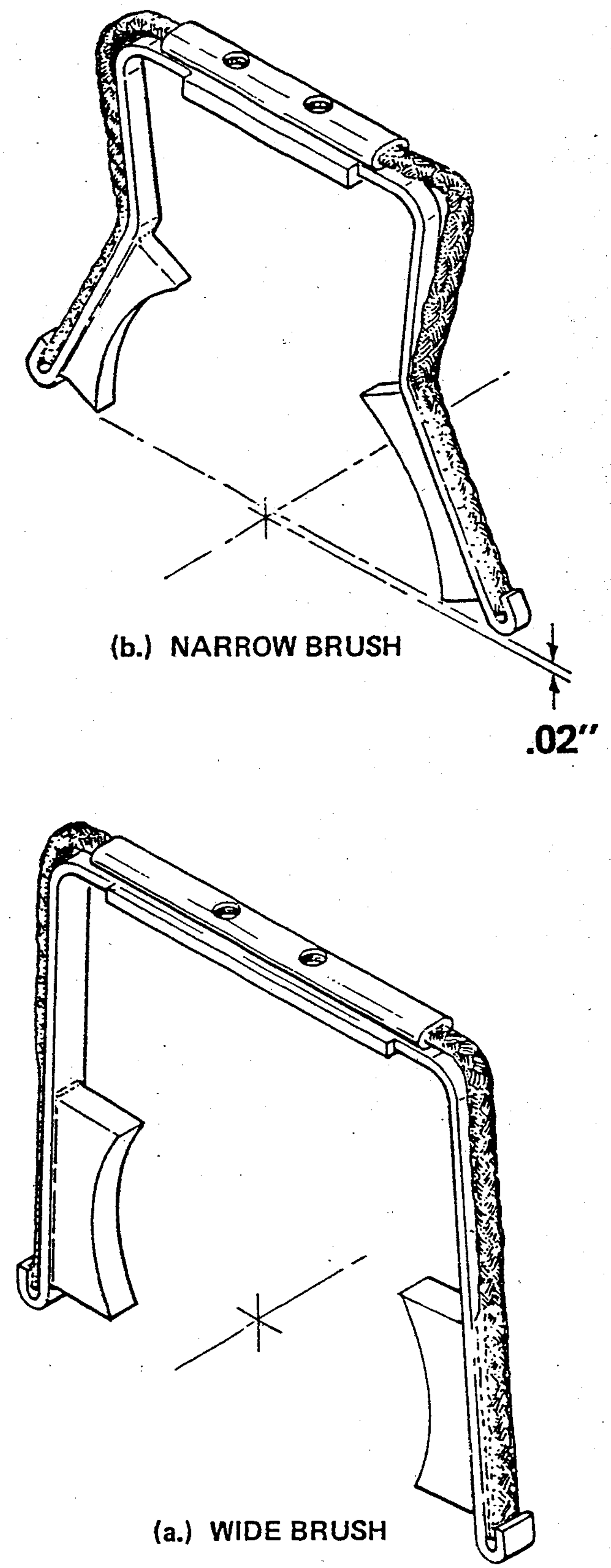

Figure 5-13. Slip Ring Brush and Spring Assemblies

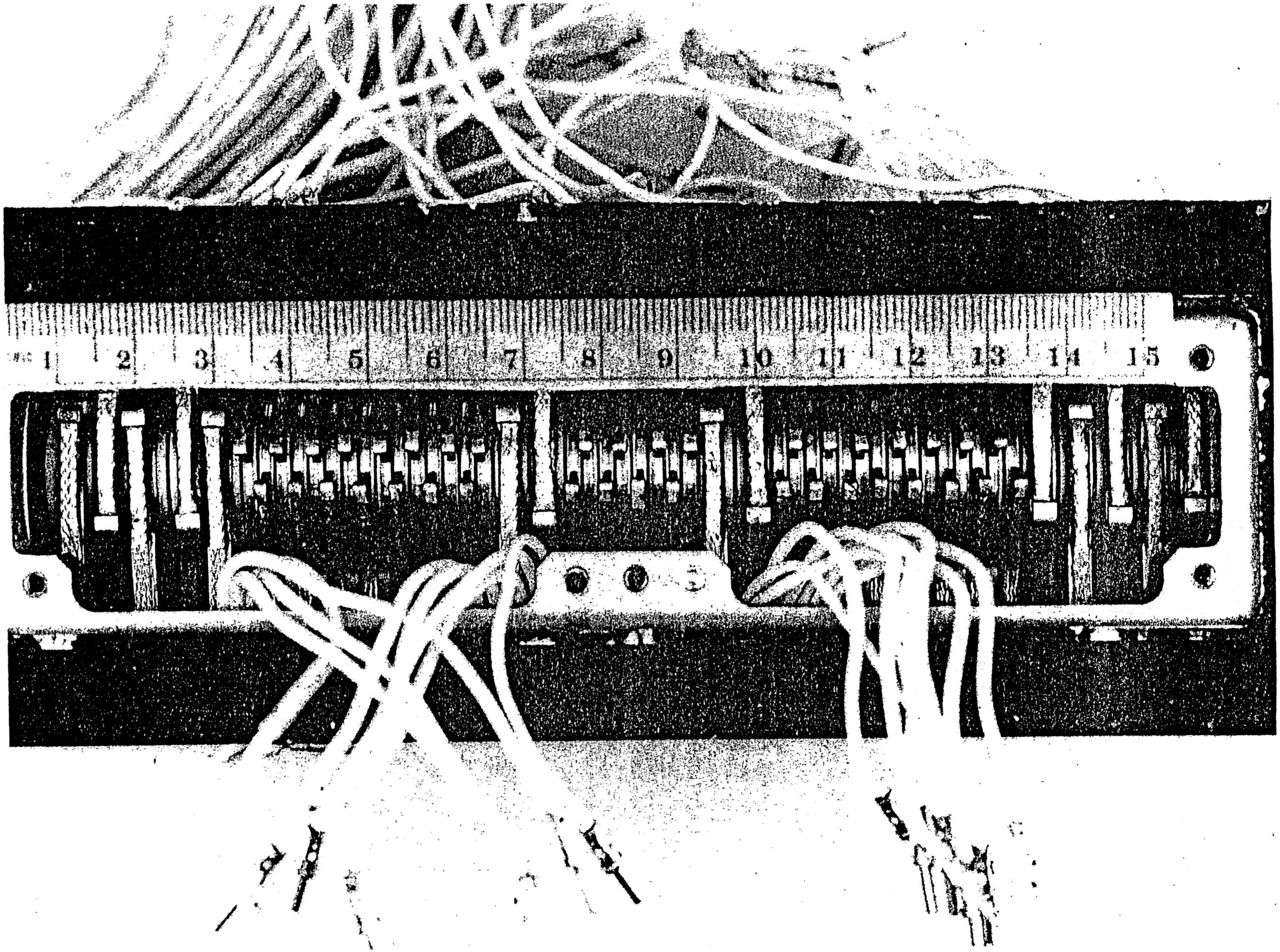

Figure 5-14. Assembled Slip Ring Assembly (Side Cover Removed for Viewing)

VI – QUALITY ASSURANCE AND FLIGHT READINESS

Introduction

This chapter will discuss the quality assurance and flight readiness requirements for the Seasat Program and the processes used to ensure compliance. The spacecraft requirements were contained in two contractually approved documents: (1) the Satellite Vehicle Specification (Part I and Part II), and (2) the Satellite Vehicle System Test Plan. The first section of this chapter will summarize the flow of requirements from the contractual documents through qualification into flight readiness and will document the Board's evaluation of the Seasat "build" paper and the potential problems in the quality assurance systems actually used.

During the early stages of the Board's investigation of the Seasat failure, it became apparent that the Slip Ring Assembly was a prime failure candidate. The second section of this chapter will, therefore, discuss the "heritage" of the Seasat slip rings. This will include a discussion of previous flights, ground tests and failure experience of similar units, and a detailed history of the specific Seasat slip ring assemblies from vendor assembly through certification of qualification for flight.

Spacecraft Requirements and Documentation

The two primary contractual documents on Seasat were the Satellite Vehicle Specification (Part I and Part II) and the Satellite Vehicle System Test Plan. There were 13 other documents which required JPL approval, but these were primarily implementation and operations type plans; i.e., Data Management Plan, Quality Assurance Plan, etc. One of these plans, the Reliability Assurance Plan, is relevant to this chapter and will be discussed herein.

Part I of the Satellite Vehicle Specification established the performance, design, development, and qualification requirements for the Seasat mission. Part II of the specification established the product configuration and system test acceptance requirements. This specification is similar to a typical Part I, Part II Contract End Item (CEI) specification used for most NASA programs.

The Satellite Vehicle Systems Test Plan established the test program for assembling, testing, monitoring and operating the Seasat spacecraft from manufacturing through launch. The Satellite Vehicle System included all LMSC furnished hardware and GFE installed in the Agena bus assembly and the sensor module. The test plan was the controlling test document and subordinate only to the Satellite Vehicle Specification. An evaluation was made regarding this flow of requirements and the interrelationships of LMSC and JPL relative to control and the visibility of requirements.

Compliance with Requirements – During the Board's review, it was determined that a significant test required by the JPL approved test plan was not conducted. The Satellite Vehicle Test Plan required electronic assemblies to be subjected to eight cycles in thermal environment of which, as a minimum, two cycles should be in a vacuum chamber (acceptance test). The Slip Ring Assembly Component Specification, however, did not require a thermal vacuum test. This noncompliance was not recognized by JPL or LMSC Systems Engineering until the present failure investigation was begun. Discussions with LMSC and JPL personnel revealed that there was not a "closed loop" system to assure compliance with contractual requirements identified in the test plan.

The fact that a component specification could be issued which violated a contractual requirement is indicative of a lack of checks and balances in the system. Another such indication surfaced in reviewing the qualification requirements. In at least two cases, to be discussed below, qualification requirements noncompliance was not documented. In fact, in the areas where the Board performed an in-depth evaluation, inconsistencies in requirements were noted in many cases. Most inconsistencies were minor; however, the impression left was that both compliance with requirements by LMSC and the "check and balance" system at LMSC and JPL were deficient.

Engineering Memoranda – Environmental derivations, test criteria, and detailed test requirements were documented in Engineering Memoranda (EM's). The LMSC stated that EM's were used to allow early generation of requirements while the spacecraft design was being finalized. A considerable number of EM's were developed during the course of the Seasat program, and it accordingly became very difficult to establish a documentation "trail" as to how test requirements were established, modified, and satisfied. In fact, two particular incidents were uncovered during detailed evaluation into the qualification status of the EPS components that point out the weakness of the EM system.

In one case, the Seasat environmental requirements specified a 5 minute per axis random vibration level but several components were qualified by similarity to a program that required only a 3 minute per axis vibration. This 5 minute per axis requirement was also specified in Part I of the Satellite Vehicle Specification. There was no documented evidence that this noncompliance was acceptable. In the second incident, pyro shock levels for Seasat were not enveloped by the program to which the Seasat slip ring assemblies were "qualified by similarity." While an EM stated that the slip ring assemblies are "not highly sensitive to pyro shock," there was no documentation or analysis to support the stated conclusion.

Because Seasat was a one-of-a-kind vehicle, LMSC did not summarize the requirements contained in the various EM's into a single "baseline" document. A "baseline" document, with change control, would have been a systematic approach to assuring requirements were satisfied and would have provided a "feed-back" mechanism to all parties. The large number of EM's produced in the Seasat program made it very difficult for LMSC to use the EM's to manage the program and to assure continuity in requirements, as exemplified above, and equally difficult for JPL to effectively penetrate the system.

The Failure Modes, Effects, and Criticality Analysis – The Failure Modes, Effects, and Criticality Analysis (FMECA) prepared for Seasat utilized the Fault Tree Analysis Technique. In effect, this was a method for studying the factors that could cause an undesired event to occur and inputting these factors into a computer model to which probability data could be applied to determine the most critical and probable sequence of events that could produce the undesirable event.

The Reliability Assurance Program Plan required that a FMECA be performed at the system level. Further evaluation revealed that "critical/new equipment" would also be subjected to an FMECA. Out of the 74 critical items identified on Seasat, only three were judged to require component level FMECA's. These were the Command Timing Unit (CTU), the Telemetry Sensor Unit (TSU) and the Synthetic Aperture Radar (SAR) Antenna (supplier performed).

The FMECA for the EPS stated that there were "no single point failures" and listed a number of redundancies, including main bus power supply channels, batteries, charge controllers, and others. Electrical shorts were, however, *not* included as possible failure modes; almost all of the effort was directed toward consideration of failure modes that would result in loss of solar array power, and the only slip ring assembly failure mode considered was "slip ring contact failure." The lack of consideration of electrical shorts in effect prevented the FMECA from serving as a tool for directing attention to those portions of the system where electrical shorts could occur and led to the erroneous conclusion that there were no single point failure modes in the EPS.

Component Specifications – Component specifications were used on Seasat to define the design, performance, acceptance, and qualification requirements of the major hardware items and subassemblies. Because the program intent was to utilize as much off-the-shelf hardware as possible, many existing specifications were red-lined and updated for the Seasat Agena bus. These red-lined specifications were then converted into component specifications by the Responsible Equipment Engineers (REE's). After April 1976, a Program directive established that all component specifications on Seasat required the signature approval of Reliability Engineering, of Space Technology, and of the Chief Systems Engineer (CSE) in addition to the REE and the Program Engineer. Two specifications were released prior to April 1976 and never received the full complement of signature approvals. These two specifications were for the Slip Ring Assemblies and the Solar Array Drive Motors. Had the other three Engineering organizations reviewed the specifications, quite possibly the Slip Ring Assembly thermal vacuum test deletion may have been prevented and inconsistencies in the qualification requirements may have been avoided. The component specifications were not reviewed and approved by JPL.

Qualification for Flight – The Seasat Program used the classical methods of qualifying hardware for flight. These were:

a. Qualification by test to demonstrate the capability of an item to meet specification requirements.

b. Qualification by design similarity whereby an unqualified item is compared with an item qualified by test to determine whether the requirements for both items and their configurations are sufficiently similar to justify not testing the unqualified item.

c. Qualification by engineering analysis, independently or in conjunction with test and/or similarity, to meet a specific qualification in the specifications. The use of engineering analysis alone could not be used to satisfy all qualification requirements.

In September 1976, the LMSC Seasat Project issued a directive creating an Equipment Qualification Review Board for the purpose of reviewing and approving all qualification and design similarity certificates. The primary membership of the board included the Program Engineering Managers, the Chief Systems Engineer, the Program Reliability Engineer, the Quality Assurance Manager, and the Applicable Space Technology Manager. This Board met every two weeks to review the status of the qualification program and to determine what additional tasks were required to qualify a given item. Status reports were issued by Program Reliability Engineering who tracked the qualification progress and documented open items.

The qualification cycle concluded with a meeting to review all test data, design similarity statements, engineering analyses, and individual component pedigree packages. Individual Certificates of

Qualification were issued stating that the specific component had been qualified to the intended environment and was acceptable for flight. A JPL engineering representative attended these qualification review meetings but was not required to approve the qualification certificate. A JPL reliability representative attended approximately 25 percent of the review meetings.

Review of Build Paper – An evaluation of the Seasat "build" paper was made with primary attention focused on the EPS. The review encompassed the electrical harness fabrication and installation, the "pedigree packages" on electrical components and assemblies, Nonconformance Reports (NCR's) on anomalies encountered in assembly and test, vehicle assembly log books, and the vehicle acceptance summary.

Because the Board's failure analysis eventually identified the slip ring assembly as the component responsible for the Seasat failure, detailed "build" paper associated with this component only will be discussed in the next section. However, some brief observations are presented below that deal with other findings made during the course of the investigation.

The NCR's are used by LMSC to document nonconforming conditions and resultant dispositions and correction actions. In general, the NCR system at LMSC was found to be acceptable. At the Board's request, LMSC reviewed, cataloged, and summarized all EPS NCR's and made a conscious decision as to the possible effect of the anomaly in contributing to the Seasat failure. None of the nonconformances were judged to be contributory to the failure.

Evaluation of the spacecraft build paper of the EPS indicated that the Air Force Plant Representative Office (AFPRO) involvement, operating under delegation from JPL, was shallow. Inspection coverage was concentrated at the system level with few in-process mandatory inspection points.

Early negotiations surfaced the fact that AFPRO could neither provide the number of personnel nor the required skill levels to perform electronic inspections. As a result of these negotiations, JPL elected to send three JPL inspectors on extended TDY to perform 100 percent solder joint inspections and electronic component acceptance testing. While it cannot be stated that a more in-depth involvement by the government would have prevented the failure, it is the opinion of the Board that the depth of penetration was inappropriate and a more selective penetration would have been in order rather than a nearly total reliance on system level audits and shakedown inspections for the bus assembly operations.

Slip Ring Heritage

Previous Experience – Consistent with the basic philosophy of the Seasat program to use, to the maximum extent possible, standard flight-proven equipment, the solar array drive motors and slip ring assemblies for Seasat were adapted from another LMSC program. At the time of initial contract negotiations, this other LMSC program had just developed a slip ring assembly and was in the process of performing qualification testing. This slip ring was also being considered for still other LMSC programs and it was anticipated that the assembly would be a qualified and flight-proven design by the time Seasat was flown. As it turned out, however, the program for which the design was originally developed was cancelled after completion of slip ring qualification but prior to flight; however, one other LMSC program did fly a slip ring assembly of this design shortly before Seasat was launched. While the design of the slip ring assembly for Seasat and this "previously flown" program were identical, the wiring sequence of the individual rings and brushes was different in the two programs. As noted in Chapter V, the Seasat slip rings were wired such that most of the adjacent power brushes were of opposite DC polarity while the other LMSC program was wired such that adjacent power brushes had the same polarity. This difference in how the slip ring assemblies were connected into the EPS thus became crucial to the heritage of the Seasat slip ring assembly; when the Seasat slip ring assembly became, in its application, connected in a manner that was different from its sole predecessor it became a unique, first of a kind component.

Two significant problems were noted as a result of random vibration testing of the slip ring assemblies used for the other LMSC flight program. An isolation failure was found after vibration testing in two adjacent brush/ring circuits. The corrective action was to separate the brushes. Also, when the assembly was opened for this operation, a crack was noted in the brush mounting block at a mounting hole. This block was replaced on the failed unit and a "T" strengthener was added to all identical slip ring assemblies, including the Seasat units, to distribute the mounting loads away from the mounting point.

Failure History – Slip ring assemblies of the design flown by Seasat experienced two "nonconformances" that provide evidence of two separate failure mode possibilities. One of these was the isolation failure noted above on the other LMSC flight program that was indicative of a possible failure mode due to contact between adjacent brushes of opposite polarity. Another failure mode identified on one of the Seasat assemblies was caused by shorting of a wire to ground due to cold flow of the Teflon insulation in the region where high stresses were imposed on the wire. This incident will be described in a later section of this Chapter.

Considerable evidence exists in published reports that the sliding friction between brushes and rings will generate debris particles that can accumulate and produce electrical noise or, in some cases, short circuits between adjacent rings and brushes. The LMSC experienced a shorting failure in a slip ring assembly used in ground tests of a control moment gyro prior to June 1977, which was attributed to accumulation of brush-generated debris and subsequent arcing between adjacent power brushes. Discussions with engineering personnel from TRW, Ball Corporation, and Sperry Flight Systems have indicated that other aerospace contractors have experienced similar slip ring shorts in ground tests. As a result of their experience with slip rings, Sperry initiated an experimental study of the possible effects of debris which will be described in Chapter VIII. While the Board recognizes that there are significant differences between the design and application of the Seasat slip ring assembly and these other units, experience illustrates a third possible failure mode due to shorting caused by contaminants or debris within the assembly.

Seasat Slip Ring History – A portion of the build history of components is assembled by LMSC into "Pedigree Packages." These packages contain component drawings, a component specification including acceptance and qualification test requirements, NCR's, and some vendor documentation including specified test plans and test records. Component selection for pedigree packages was determined by the Seasat Program Office and the Quality Assurance organization at LMSC. The Seasat slip ring assemblies were documented by such pedigree packages. Relevant component history *not* contained in the slip ring Pedigree Packages included vendor assembly and test NCR's (including failure reports), assembly test procedures and records (including brush alignments and pressure checks and brush "run-in" procedures), and relevant vendor and customer correspondence.

The timing of the Seasat contract was such that LMSC was able to acquire two partially assembled slip ring assemblies when another LMSC program referred to herein as Program A, was cancelled. Program A had initially contracted for 10 assemblies and, at the time of termination, had accepted delivery of one qualification unit, one development unit, and two production units leaving six partially assembled units at the vendor. The Seasat Program picked up two of these units and LMSC Program B picked up the additional units. Reference will be made to Program B in other portions of this report relative to test experience and use of Program B qualification testing as a basis for qualifying the Seasat slip rings by "similarity."

Program A personnel were informed by Poly-Scientific in late 1973 that the constraints placed upon the length of the assembly were found to be restrictive and that relief of the specifications would enhance reliability. Program A, however, could not relax the specification. Although the Seasat application was not constrained by length, the program desire to use available "off-the-shelf-hardware" precluded the development of a new unit having increased dimensional tolerances between the rings and brush assemblies with possibly enhanced inherent reliability.

Seasat personnel initiated discussions with Poly-Scientific in late 1975 using the LMSC Program A specification as a baseline. On February 3, 1976, Poly-Scientific submitted its first written quote for two assemblies to be fabricated and tested per the Program A specification. This initial quote was not acceptable to LMSC and the REE and buyer responded on March 5, 1976, with a Seasat red-lined version of the Program A specification. It was in this March 5, 1976, specification that the Program A requirement for 10 cycles of thermal vacuum acceptance testing was deleted. This deletion occurred even though: (1) the majority of the Seasat electronic assemblies and electro-mechanical assemblies were subjected to a thermal vacuum acceptance test; (2) neither Seasat Reliability personnel, Systems Engineering personnel, nor JPL personnel were aware of this deletion until the present failure investigation; and (3) the thermal vacuum test was contractually required and a waiver of the requirement was never issued.

Upon pursuing the thermal vacuum deletion further, it was determined from interviews with involved personnel that the test was deleted during verbal negotiations between both the REE and the buyer at LMSC, and the vendor in order to reduce unit cost of the slip ring assemblies. The responsible LMSC program engineer approved the deletion but, at that time, there was no requirement to coordinate specifications with the Seasat Program Reliability Engineer or the Chief Systems Engineer. The fact that a waiver was not issued on this and other contract noncompliances is indicative of a weak compliance system between Lockheed and JPL.

On March 25, 1976, LMSC issued a formal Request for Quote (RFQ) to Poly-Scientific for two Seasat slip ring assemblies built to the March 5, 1976, specification with a requested delivery date of 1 year. On May 26, 1976, LMSC authorized contract go ahead for two slip ring assemblies at a unit price of $8,953.50.

Researching the manufacturing history and fabrication and test anomalies at Poly-Scientific resulted in the following:

a. There were four anomalies noted on slip ring unit 1001. Three were minor and appear to have had no real impact on assembly reliability. The fourth anomaly was a Teflon wire short to an adjacent ground lug. The repair action, approved by LMSC engineering, was to insulate the ground terminal and repot with ES 222-2 cement. The damaged insulation on the wire was not repaired. This discrepancy report was not included in the vendor's data package and consequently this failure was not contained in the LMSC Pedigree Package.

b. Slip Ring Unit 1002 (–Y Solar Array) had the more significant anomalies noted during fabrication and test. These anomalies are summarized as follows:

(1) 9/20/76 – 80 minute run-in of brushes to rings at 100 ± 10 RPM. Run-in time should have been for 100 to 115 minutes. This discrepancy was missed and not documented.

(2) 9/23/76 – discrepancy No. 146522–discolored rings noted after above run-in test. Unit had to be completely disassembled, brushes and rings recleaned, unit reassembled and another run-in performed. The exact run-in time was not recorded nor entered into the log book.

(3) 11/12/76 – discrepancy No. 151887–excessive noise noted caused by moisture pick-up in the brush material. Corrective action was to run the unit in vacuum at 14.4 RPM for 1½ hours. No vacuum cleanup was performed after this 14.4 RPM run-in test. This run time was not entered into the log book.

c. Review of vendor documentation and subsequent teleconferences with Poly-Scientific personnel revealed the following assembly technique and procedures:

(1) The assembly planning documentation specifies that the brushes were to be aligned "in center of the rings." This requirement was verified visually by the inspector, but no dimensional checks were made. Proper alignment of the brushes is dependent, therefore, on the inspector's judgment.

(2) Poly-Scientific stated that the tolerances within the slip ring assembly could allow adjacent brushes to touch. It is noted here that an identical slip ring assembly experienced an isolation failure during acceptance testing which was probably caused by adjacent brushes touching. (Program B hardware).

Both Seasat slip ring assemblies were shipped from Poly-Scientific on February 22, 1977. These units were received and accepted at LMSC on March 11, 1977, where they remained in storage until required for installation on their respective Solar Array Modules.

In approximately July 1977, LMSC Program B, which utilized identical slip ring assemblies, made a wiring change external to the slip rings that separated the polarity arrangement of adjacent slip rings. By changing connector pin functions, the power applied to individual rings was changed from a configuration in which adjacent rings were of opposite polarity to one having positive contacts on one end of the slip ring assembly and negative contacts on the opposite end. This wiring change significantly reduced the possibility of internal shorts within the slip ring assembly.

The Seasat Chief System Engineer was contacted by a system engineer from Program B about this change in wiring in August 1977. The explanation given for the wiring change was a concern that the ascent vibration environment could cause adjacent brushes to make contact and thus produce an electrical short because Program B slip rings had power applied during launch. The Chief System Engineer discussed this change with the Seasat program engineer and they decided not to make a similar wiring change because Seasat did not see the same launch vibration levels and because Seasat slip rings were not planned to be powered during launch. It is noted that in April 1978, a change in launch relay configuration was made which did apply power to the slip ring assemblies. In retrospect, the decision not to change the wiring sequence for Seasat was a crucial one. When the other program changed its wiring and Seasat did not, Seasat became the first program to fly a 52-brush slip ring assembly with adjacent brushes of opposite polarity. Had there been better visibility to the problems experienced with slip rings by both the vendor and by other organizations within LMSC, the Seasat engineering managers may have been more sensitive to the failure-prone nature of this

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complicated device and to the importance of the electrical polarity of adjacent brushes. Unfortunately, such visibility, which may only have needed to have been slight to have been effective, was lacking.

Slip Ring Assembly Serial Number (S/N) 1002 was installed on the –Y Solar Array Module (SAM) on August 17, 1977. On August 30, 1977, an NCR was written because the mechanic "lost" an undetermined number of shim washers.

Review of the installation drawing revealed that four number 10 washers were required between the solar array mounting structure and the slip ring assembly. The cover of the assembly is made of thin sheet metal and is prone to "bow up" during installation operations. Because the mounting bolts go through the cover plate into the threaded holes in the slip ring body, the mechanic had to place the round washers over the bolts between the structure and the cover plate. It was during this operation that the mechanic "lost" the washers. The S/N 1002 Slip Ring assembly was removed from the solar array module, the cover plate removed and three washers were found. Because some areas were still obscured, an X-ray of the slip ring was taken. No additional washers were located. An NCR was then written against Slip Ring Assembly 1001 and no washers were found by either visual or X-ray inspection. It is interesting to note two things: (1) there were no downstream electrical functional checks after installation of the slip ring assembly which could have detected missing washers in the slip rings, and (2) it was never conclusively determined if all "lost" washers were found.

The Solar Array Modules, including the slip ring assemblies, were shipped to the launch site in April 1978. The last reported anomaly on the slip rings was high contact resistance on unit 1002 during interface tests performed when the Solar Array Modules were mated to the vehicle. The resistance reading recorded was 2.38 ohms; the specification value was 2.00 ohms maximum. The engineering disposition in the NCR was "use-as-is" because in-flight operation would decrease the contact resistance.

VII – MISSION OPERATIONS

Overview

Seasat mission operations were conducted under JPL management from a POCC located at GSFC. Spacecraft telemetry, tracking, and command support were provided by GSFC, utilizing the STDN through the NASCOM Network. Real-time POCC telemetry, display, and command processing were also provided by GSFC, along with off-line orbit attitude and maneuver computations, and computer support for power profiling. Figure 7-1 depicts the mission operations interfaces.

Processing of whole orbit spacecraft tape recorder dump data was provided by the GSFC Information Processing Division (IPD) to allow playback in the POCC for spacecraft housekeeping "quick look" purposes. For non-real-time mission planning activities at JPL, GSFC prepared a project data package containing a complete 24-hour file of data from the tape recorder dumps, along with associated orbit, attitude, and command histories.

The JPL mission planning system provided the POCC a weekly command request profile, from which GSFC generated the daily mission sequence of events and the command load file. This information then led to the development of the station pass plans used on the console.

The LMSC provided GSFC with specialized attitude determination and power profile software programs that were used off-line in the POCC for spacecraft system management.

The Seasat pre-mission operations phase began at JPL in March 1976, when mission operations planning and design started. The activity was transferred to the POCC at GSFC in January 1978, at which time team formation was accomplished and flight controller training began.

Real-time operations support began on the day of launch, June 26, 1978, and continued on a 24-hour per day, 7-day per week basis until operations were officially terminated on November 10, 1978.

Mission Control Team Organization

The real-time mission control organization operating out of the POCC was the Mission Control Team (MCT) shown in Figure 7-2. Four MCT's were organized to support Seasat mission operations around the clock. They functioned on a 4-day on, 4-day off basis, operating on two 12-hour shifts daily. All four MCT's reported to the JPL Seasat Chief of Mission Operations (CMO) or his designee, located at GSFC. Each MCT was composed of five people: one command operator, a Satellite Performance and Analysis Team (SPAT) of three engineers, and one Assistant Chief of Mission Operations (ACMO) serving as team leader. The ACMO was a JPL employee, while a Bendix employee on contract to GSFC served as the command operator who performed all the spacecraft command functions.

The LMSC provided the three SPAT engineers to each MCT. The SPAT consisted of the Lead Monitor Analyst (LMA) and two Assistant Monitor Analysts (AMA's). The two AMA's functionally split their responsibility between the spacecraft power subsystem and the attitude control subsystem. In addition, both AMA's monitored specific sensor housekeeping data. The SPAT was responsible to the ACMO for spacecraft systems and sensor status, for health monitoring, for analysis of attitude control and power subsystem performance, for spacecraft anomalous condition response, and for provision, but not transmission, of commands for spacecraft configuration control. Critical mission phases required augmentation of the SPAT by two system or sensor specialists, depending upon the mission phase.

There were several support areas provided the MCT by the GSFC POCC Operations Support Team (POST), including uplink command processing, data operations, network support, data processing, orbit computation, attitude determination and maneuver computation, and power profiling computer support.

The Seasat MCT operated from consoles located in the Mission Operations Room (MOR) in the POCC. The primary real-time data display system was a series of digital data formats of telemetered parameters, command functions, and other information, presented on console-mounted cathode ray tubes (CRT's). Snapshots (hardcopies) of these displays could be made by a high speed printer at any time. Also, a series of data formats referred to as "snaps," different from the CRT displays, could be printed out on demand by a high speed printer. These were used primarily as historical data to maintain manual trend plots and for power and attitude evaluation. Analog strip chart recorders were used by the MCT to monitor selected parameters in the power and attitude control subsystems. Interactive console keyboards provided the method by which the controllers communicated with the POCC computer and transmitted commands to the spacecraft.

Mission Operations

Low earth orbit missions require a different philosophy of mission operations than do deep space missions where the spacecraft is in continuous communications with ground stations for long periods of time. Earth orbit operations planning has to accommodate the "snapshot" nature of the data viewing concept inherent to this type of mission, where the flight controllers receive short periods of real-time telemetry data as the spacecraft passes in and out of station contact on each revolution. This situation requires designing an efficient mission planning system in order to properly schedule flight events and activities to coincide with STDN station coverage on an orbit-by-orbit basis. It results in more complex data management plans and procedures in order to recover uninterrupted spacecraft telemetry information over a specific period of time. For example, onboard tape recorders have to be managed such that one is always recording data while the other is ready to be played back ("dumped") to the ground. This continuous information is necessary for daily attitude, thermal and electrical power management, and for long term trend analyses; provisions therefore have to be made to recover the dumped data in a minimum turnaround time.

In order to maximize the chances of survivability in case of a spacecraft anomaly in this kind of operating environment, flight controllers require mission rules and associated contingency procedures to "safe" the spacecraft before the anomaly cascades into total failure. To be effective, these rules and procedures must be well thought out, be as simple and clear as possible, and very importantly, be capable of rapid execution. All creditable single point failure modes should be accounted for in the development of these procedures, using the results of FMECA's or their equivalent. The controllers, of course, have to be well-trained in spacecraft subsystems to be able to recognize the symptoms as presented on their displays and to quickly associate them with the correct anomalous condition in order to respond with the proper procedures. Finally, the flight controllers must rehearse all contingency operations with a sufficient number of reasonably high fidelity simulations to demonstrate their readiness to support flight operations.

Seasat was the first low earth orbit project conducted by JPL since Ranger 2, and was also the first JPL project flown from the GSFC POCC. Investigation revealed that the Seasat operations personnel, with extensive deep space mission background and little experience in earth orbit operations, were sensitive to the inherent differences between the two concepts, and, with two exceptions, were able to satisfactorily account for this in their Seasat mission operations planning. These exceptions relate to deficiencies in the flight controller training and in the development of mission rules and procedures associated with the Seasat Agena bus, especially the EPS. There were also some areas of concern which adversely affected smooth operations. Discussion of the deficiencies and the areas of concern follow.

The geographical separation between the project office and mission planning activity at JPL and the operations control center at GSFC, and between the spacecraft engineering activity at LMSC Sunnyvale and the operations control center, did present the obvious problems of travel and associated costs during the period when the JPL/GSFC and LMSC/GSFC operational interface requirements were being established. These were overcome satisfactorily. One serious problem due to this separation surfaced during the investigation which concerned the difficulty LMSC had in providing subsystem experts familiar with the Seasat spacecraft to relocate from Sunnyvale to the GSFC area in support of mission operations. This will be discussed in more detail in the following.

Training

The Seasat MCT personnel received informal and formal classroom training before and after team formation at GSFC in January 1978, followed by "hands-on" simulation and test exercises, all conducted while at GSFC. The final objective of the training was to demonstrate readiness of the Project Operations System (POS) to support Seasat operations. This included the ground data systems and associated personnel as well as the MCT, to which this discussion is limited.

The classroom training included orientation briefings and classes in various levels of detail on the POCC support facilities (hardware and software), project information (mission profile, project organization, data system, etc.), and spacecraft subsystem instruction (physical and functional description, constraints, etc.).

The "hands-on" training was composed of six different kinds of exercises with all four MCT teams participating. During the first two, *intra-team exercises* and *inter-team exercises,* the MCT teams simulated the procedures required to conduct nominal mission phases (launch and early orbit, attitude trim, orbit adjust, etc.) exercising appropriate interfaces and procedures. Test tapes from LMSC of actual telemetry data and a GSFC POCC bit stream simulator provided the simulation sources for these exercises. Real-time and play-back data, along with command capability and with limited telemetry response, were provided the MCT although no spacecraft memory was simulated. The intra-team exercises which followed extended to outside support functions.

In March 1978, an 8-hour *Satellite/POCC End-to-End Data System Compatibility Test* was conducted using the actual Seasat spacecraft at Sunnyvale, the STDN compatibility test van at

Sunnyvale to simulate an STDN station, the NASCOM circuits, and the GSFC facilities including the POCC. The POCC exercised command functions such as spacecraft clock control, command and data system configuration, tape recorder dumps, telemetry modes, sensor sequence tests, and spacecraft memory dump tests. Live telemetry was provided the MCT. All appropriate team interfaces were exercised.

A series of *combined POS simulated exercises* for various mission phases were conducted that were very similar to the inter-team exercises, except that Portable Simulation System (PSS) equipment located at STDN stations and at GSFC simulated Seasat telemetry, thus bringing actual network personnel and equipment into the simulation. Limited telemetry responsiveness to command functions was available, and spacecraft memory capability was provided in this case to allow exercising memory loading and dumping procedures.

A 12-hour long *Operational Demonstration Test (ODT)* was conducted in May 1978 which simulated launch through early operations (antenna and sensor deployment), with the actual WTR and STDN teams participating. The spacecraft test tapes and the PSS equipment provided the simulation sources. Four ground systems and two spacecraft anomalies were simulated during the ODT, which was the only test exercise in the program where any anomaly simulation was performed on the console.

The last series of "hands-on" tests was a 5½ day *Operational Readiness Test (ORT)* conducted a week before launch. This mission "dress rehearsal" stepped through all mission phases, beginning with launch. It combined all elements of the operations system, tested all interfaces, and used all three simulation sources mentioned above. The first 12 hours duplicated the ODT, exercising the WTR launch teams and an actual STDN station. In addition, an LMSC operational readiness audit team provided a number of spacecraft anomaly cases that were exercised by the SPAT during the ORT time frame. These were "paper simulations" rather than actual hands-on simulations.

Observations – Flight controller training and simulations were considered adequate in ground systems and data flow procedures.

The spacecraft subsystem classroom training, however, was considered inadequate in that it went only to the level necessary to acquaint the flight controllers with the general characteristics of the subsystems. This level of training was insufficient to insure the capability of rapid real-time anomaly assessment. The contents of the LMSC Operations Training Manual were only to block diagram level and the LMSC Vehicle Schematic Document essentially contained a collection of wiring diagrams. There were no intermediate level schematics that presented subsystem end-to-end functional signal and power flow, which would have been ideal for operational training.

The spacecraft subsystem "hands-on" training was also considered inadequate because there were only two spacecraft anomaly conditions simulated on-console during the exercises; these were a "satellite clock anomaly" case and a "no downlink" case. These anomalies were accomplished by inputs from the telemetry simulator at an STDN station during the ODT where one or two individual parameters were varied to represent the specific fault. They were considered "open loop" from a flight controller response standpoint. The Seasat project did not have a high fidelity spacecraft math model as part of its simulation system which would have provided realistic "closed-loop" exercises to the flight controllers. Therefore, extra emphasis should have been placed on overcoming this situation as much as possible by conducting more on-console anomaly simulations. As a minimum, all anomaly cases listed in the operations documentation should have been run at least once for each team.

The *Satellite/POCC End-to-End System Compatibility Test,* which was not originally in the program, provided the Seasat flight controllers a chance to see live data on their displays and to send actual commands to the spacecraft before launch. This type of test has proven very valuable in other programs in shaking out data flow discrepancies and enhancing the operations team readiness. It has also uncovered occasional spacecraft system discrepancies. In the case of Seasat, the test revealed several telemetry and command discrepancies onboard and within the ground system software, which were corrected.

It was originally planned by the JPL project that LMSC would provide the operations team at GSFC with subsystem specialists who had followed the design, development, and testing of the actual Seasat vehicle from the beginning. However, LMSC could not comply with this plan because of the reluctance of these personnel to relocate for long periods of time from Sunnyvale to the GSFC area. In January 1978, LMSC was able to hire several engineers who were already living in the GSFC area. Most of these personnal were available to the Seasat Project because they had terminated employment with RCA, whose ground systems Maintenance and Operations (M&O) contract with GSFC was being renegotiated at the time. For this reason, at 5 months before launch, the Seasat operations team received LMSC personnel who were proficient in ground systems operation but had little familiarity with the Agena bus. These former RCA employees, along with one engineer formerly with TRW and a Lockheed engineer from another facility on the East Coast, formed the four SPAT teams. The LMA's were temporarily sent to Sunnyvale for briefing and test monitoring but the AMA's remained at GSFC. All these personnel received spacecraft classroom training at GSFC, performed by LMSC engineers using the Seasat Operations Training Manual. This entire situation resulted in the flight controllers responsible for subsystem health monitoring and contingency procedures not having the in-depth knowledge of the satellite systems Agena bus subsystems considered necessary to adequately cope with real-time failures.

The M&O contract that RCA held with GSFC was put out on bid in 1977. After several months of negotiation, Bendix was notified on May 1, 1978, that they had been awarded the contract. On June 1, 1978, the switchover from RCA to Bendix occurred. A key member on each of the four MCT's was the command operator, a GSFC-supplied contractor employee. In early January 1978, all four of these positions were filled by RCA personnel, who then went through the Seasat training program with their teammates. By Seasat launch, however, of the four trained command operators, three had left the teams. Other turnovers in the command operator positions also occurred. The full complement of four command operators, i.e., one for each MCT, was never regained through the remainder of the mission; three Bendix operators therefore supported four MCT's. With only three operators available, the MCT's never had a dedicated command operator to grow in experience as a member of a closely knit team. As a command operator would become accustomed to his teammates and their uniqueness (and vice versa), team shifting would cause him to have to move to another team and start the process all over. This complicated the ACMO's task of building and maintaining proficient teams.

Procedures

Extensive capability for ground control of subsystem management and reconfiguration was provided the Seasat spacecraft by use of 317 real-time commands (RTC's) that could be used for subsystem management and contingency operations. The normal usage was for reconfiguring the attitude control subsystem, the solar array tracking subsystem, and the sensors as required. Tape recorder management was also partially controlled by RTC's. Contingency operations involved transmission of RTC's that reconfigured systems, switched in redundant equipment, or safed systems in the event of an onboard failure.

There were cases where multiple RTC's were required to effect the proper reconfiguration. Therefore, in order to minimize the time required for the flight controllers to pick out the proper RTC's in real-time, arrange them in the proper order of execution, and transmit each of them separately, sequences of properly grouped RTC's were designed for both normal usage and contingency operation. These sequences were loaded into the ground system software so that the flight controller needed only to initiate a single execute command to transmit all the particular RTC's required for configuration changes.

Four sequences were created for contingency operation, with three of these associated with spacecraft safing for EPS anomalies and a fourth for attitude tumble recovery. These were as follows:

Sequence 1 – "Programmer Inhibit"
Prevented further on-board stored program commands from clocking out by disabling all memory banks in the operating command processor unit.

Sequence 2 – "Sensor Power Down to OFF"
Removed Sensor loads from unregulated bus.

Sequence 3 – "Maximize Array Output"
Provide full solar array output by closing all panel connect relays.

Sequence 4 – "Condition RCS"
Thermally conditioned and powered up the reaction control subsystem in case of an attitude control subsystem failure.

Eleven anomalous conditions were selected for use in contingency operation planning which were as follows:

a. Low battery voltage
b. Array not tracking
c. High battery temperature
d. High current
e. Over voltage
f. Large attitude errors
g. Attitude component failure–small errors
h. Clock anomaly
i. Programmer failure
j. No downlink
k. No response to uplink

Flight controller responses to these anomalous conditions were prepared and published in May 1978 in two documents. One was the LMSC Seasat-A Operations Requirements and Constraints Handbook (ORCH), used by the SPAT personnel as their primary real-time procedural document for spacecraft control, and the other was the JPL Seasat-A Space Flight Operations Plan (SFOP) used by the JPL and GSFC personnel as their primary operations document.

The ORCH presented some of the flight controller responses in "decision logic tree" diagrams while others were written statements in paragraph form. A chart was also included that attempted to summarize the reactions to all 11 anomalous conditions. The chart was not as complete or as specific as the narrative material. There were also some inconsistencies between the narrative material and the summary chart.

The corresponding information in the SPAT section of the SFOP was more step-by-step "console procedural" oriented, with tabulated listings of specific procedures which included those for data collection.

Both the ORCH and the SFOP specified safing sequences 1, 2, and 3 to be transmitted for anomalous conditions 1 through 4 (electrical power conditions). In addition, the ORCH stated that, if the condition were considered critical for conditions 1 and 2, additional loads should be removed by transmission of the "heater bus off" RTC. It was not specified in either the ORCH or the SFOP which "current" was intended in condition 4, the two battery currents or the unregulated bus current, because it was to cover all three cases. The operator was to decide at the time, based on which telemetered current readings appeared abnormal. The prescribed action was the same, however, for all three current cases, that is, (1) determine the load source and disconnect the "offending" equipment, (2) transmit safing sequences 1, 2, and 3, if the load source was not apparent, and (3) continue troubleshooting. One item open to interpretation in the documentation was the length of time the flight controllers should take in attempting to determine the load source before transmitting the safing sequences.

Assuming that condition 4 applied to the two battery currents, the red-line values selected for the operators use in evoking the response to the conditions applicable to the failure were:

Condition 1 (Low Battery Voltage) – 26.5 volts (minimum)
Condition 3 (High Battery Temperature) – 60°F (maximum)
Condition 4 (High Battery Current) – 18 amps (maximum)

If these values were exceeded, the POCC computer automatically notified the operators by placing an asterisk next to the CRT parameter readout.

Observations – The ground systems and the normal spacecraft operating procedures specified in the Seasat operations documentation (the ORCH and the SFOP) appeared adequate for proper conduct of the mission. However, the spacecraft contingency procedures were not considered adequate for operations. More emphasis should have been placed on identification of additional failure modes in the Agena bus EPS. This would have led to a more complete set of anomalous conditions and associated responses in the EPS than was documented. For example, a preplanned command sequence could have been prepared from available RTC's that would have isolated all or portions of the solar arrays from the power distribution and storage subsystem should a short circuit arise in the solar array subsystem.

There was no preplanned true emergency safing sequence prepared to rapidly remove *all* loads except those absolutely essential to keep the vehicle alive until troubleshooting could isolate a serious electrical failure. One such sequence could have been prepared from available RTC's that would have quickly turned off all sensors, the heater bus, and all nonessential loads. The only equipment that needed to be on in this critical situation was the S-band transponder, telemetry and command equipment, and some minimal attitude stabilization equipment.

Three of the anomalous conditions listed in the ORCH and the SFOP matched the symptoms of the Rev 1503 flight failure. These were low battery voltage, high battery temperature, and high current. (The documented high current condition should have been more specific as to which current was intended.) The specified flight controller response to these three anomalous conditions was the same–inhibit further clocking out of onboard stored program commands (transmit safing sequence 1), turn off all sensors (transmit safing sequence 2), and maximize solar array output (transmit safing sequence 3).

The contingency operations section of the ORCH should have been prepared with more attention to consistency between the anomalous condition/response summary chart and the narrative. Also, the procedures should have been more specific as to how long the flight controllers should take in attempting to determine the anomalous load source before transmitting the safing sequences, or whether the safing sequences should be transmitted first in all cases. It was not very clear in either case. Logic tree diagrams should have been prepared for all the conditions, not just a few.

The availability of the Seasat spacecraft contingency operation data such as the anomalous conditions, reaction to these conditions, the safing procedures, and the red-line data with which to evoke these procedures, was very late in the premission operations period. This factor made it impossible for the flight controllers to properly analyze the data, learn the procedures, and prepare themselves for spacecraft contingency operation before launch.

Operational Concerns

Turnaround of Whole Orbit Data – The Seasat project had a requirement that "whole orbit data" be available in the POCC within 4 to 6 hours after GSFC IPD receipt of raw data, on a daily basis during certain critical mission periods such as orbit adjust/trim maneuvers and low power periods, and on demand at other times during the mission. Whole orbit data consisted of two revolutions of recorded data on the spacecraft housekeeping tape used by the MCT for spacecraft subsystem "quick look" purposes, and for power and attitude management. It was produced by IPD from spacecraft tape recorder dumps played in from the particular STDN station receiving the dump. Due to various problems within IPD and the STDN, the turnaround time was not met until near the end of the mission. In some cases the turnaround was as long as 1 or 2 weeks.

This problem hampered the MCT SPAT personnel throughout the mission in performing proper trend analyses within a reasonable time. This excessive turnaround time could have been alleviated if the Seasat project had used the standard STDN-to-POCC direct reversed tape recorder dump technique for POCC "quick look" data common on other space programs.

POCC Data Processing – The effectiveness of any mission operations team is only as good as the quality of the data presented to them on the consoles. Team indecisiveness due to erroneous data indications during a spacecraft anomaly could result in a failure cascading into total loss of the spacecraft that might otherwise be manageable. A normal mission might even be severely jeopardized if the team takes improper action because of this kind of problem.

Consistently high data quality was lacking in the Seasat operation for the entire mission. The problem centered around the POCC software program that processed the spacecraft real-time telemetry digital data transmitted from the STDN stations. This processing consisted of the conversion of the incoming raw data into parameters with engineering units. In most cases, the discrepancy that caused the erroneous display required one or more POCC computer reloads ("reboots") to clear the problem. Although the situation did improve as the software program progressed through two major updates after the launch, random display problems continued to the end of the mission. For example, during the 2-week period just prior to the October 9 failure, 11 display anomalies arose where the computer required reloading to clear the problem. Three of these required two reloads, while two others required three reloads. The typical display characteristics of these problems would include parameters randomly off-scale.

Documentation – As stated previously, there were two Seasat operational documents containing spacecraft procedures. One, the Operations Requirements and Constraints Handbook (ORCH), was published by LMSC specifically for use by the LMSC SPAT personnel responsible for the spacecraft operations. The other, the Space Flight Operations Plan (SFOP) published by JPL, was a much broader document to be used by all Seasat project personnel (JPL, GSFC, and LMSC).

The existence of two sets of procedures left open the question of which was the governing document. This was complicated by the fact that they were different in format. This situation could conceivably delay timely real-time response to an anomalous condition, although there is no direct evidence that this occurred during Seasat operations. Operations plans are generally published to tell how a project will be operationally conducted, identifying the roles of the participating organizations. It should not list specific console procedures; these should be listed in a single document with more limited distribution, such as a console operating handbook, which is recognized as the authoritative program document for flight controller console operating procedures.

Other than the overlap problem in spacecraft procedures, the ORCH and the SFOP served their purposes adequately, as did the LMSC Command Handbook and the LMSC Telemetry Instrumentation Schedule. The LMSC Operations Training Manual was found to contain technically incorrect data, and the LMSC Vehicle Schematic Document contained wiring diagrams not considered suitable for operations.

Real-Time Operations

Revolution 891 Findings – The first low power period of the mission was expected in late August 1978, and LMSC Sunnyvale thermal and electrical subsystem engineering specialists traveled to GSFC to monitor the spacecraft subsystem performance in the POCC during this period. They also provided technical support to the MCT personnel in those specialties.

On August 26, an orbit adjust maneuver was executed on Rev 863, which required one solar array to be feathered to minimize reaction control jet impingement effects. The other panel had to be fixed also, but placed into position to provide maximum power. This was a standard operation, requiring all sensors to be powered down because of the low power state. The preplanned command sequences were used to conduct the operation. Full solar array output was not available for about 2 hours during this operation.

On August 28, the spacecraft entered the low power period, 2 days earlier than anticipated by the LMSC specialists. The MCT monitored real-time data during the STDN station coverage, which was limited to the peak portion of the charging cycle and the subsequent entry into the Earth eclipse, for six consecutive revolutions (886 through 891). Although neither of the K2 relays were ever observed to open during these 9 hours, there was no apparent concern with the POCC. In fact, on

Rev 891, a planned SAR operation was conducted 20 minutes before entering the eclipse. About 1½ hours after observing peak charging and eclipse entry on Rev 891 over the Hawaii STDN station, the MCT received their first look at Rev 892 live data over the Alaska STDN station. All spacecraft subsystems looked normal, with the exception that the altimeter transmitter was not operating as it had been during previous revolutions. The flight controllers first believed the altimeter had experienced a failure, but after some quick investigation, they concluded a power problem of some nature existed. At this time, they terminated sensor operation until the cause of the problem could be determined.

A spacecraft housekeeping tape of Revs 890/891 whole orbit data was immediately requested by the MCT to provide "quick look" data. When this tape was delivered a little less than a week later and played into the POCC computer system, the MCT personnel and LMSC engineering specialists received confirmation of a very serious power situation that had been encountered due to extremely low unregulated bus voltage during Rev 891. It was obvious at that time that the altimeter had simply turned itself off due to low voltage. Other tape playbacks of previous orbits revealed the progressively lower voltage dips from Rev 886 onward, leading up to a 21.8 volts minimum experienced on Rev 891. Data showed the system recovered during the next revolution. The MCT did not resume normal sensor operation until September 7.

Concluding that the spacecraft subsystem trend monitoring procedures were inadequate, the MCT SPAT personnel and LMSC Sunnyvale personnel immediately instigated a daily manual trend plotting technique using snapshot spacecraft health data from each real-time STDN pass. They recognized that more emphasis needed to be placed on K2 relay status as part of power management, and added a mission role that required both K2 relays to be open for at least two revolutions before operating the SAR. In addition, it was concluded that the power profile program needed recalibration because it did not include an estimated 35 watts being consumed due to the inefficiency of manual heater operation as a result of the July 6 heater thermostat anomaly described in Chapter VI. Software updates were planned to account for this recalibration, but the updated package was not delivered before the October 9 failure. The SPAT personnel had to manually input this increased power loading each time they ran the program.

Revolution 891 Observations – Seasat electrical power management was not well understood by the personnel in the Mission Operations Room, at least not until after analysis of the events of August 28 when the serious low unregulated bus voltage drop occurred. Even though these personnel and the visiting LMSC engineering specialists were fully aware of the first mission low power period, expected in late August, insufficient attention was paid to the battery charging state. This was exemplified by the lack of concern, with no power-down actions, when the K2 relays remained closed over at least the 6-orbit (9-hour) period before the Rev 891 minimum voltage dip, and by the conduct of an SAR operation with the K2 relays closed.

The problem was aggravated by the STDN station geometry during revolutions 886 through 891, which precluded the flight controllers from viewing the voltage drop which developed progressively over those six revolutions. The station acquisition period for each of these revolutions included the entry into the eclipse but did not cover the sharp voltage dip occurring afterwards.

The analysis of the situation was delayed by the long turnaround time experienced by the MCT in receipt of the spacecraft housekeeping tapes. Even if a tape had been requested for Revs 886/887 however and had arrived within the 4- to 6-hour specified time requirement, sufficient time would not have been available for the flight controllers to prevent the Rev 891 minimum voltage dip and altimeter trip-off.

The increased rigor in overall spacecraft subsystem monitoring procedures instigated after this low voltage problem resulted in satisfactory power management techniques for the rest of the Seasat mission.

Revolution 1503

Findings

A sequence of events of activities and observations within the POCC Mission Operations Room (MOR) during Rev 1503 is presented in Appendix B. The sequence begins at 23:30 hours Greenwich Mean Time (GMT) on the Day of the Year (DOY) 282 when Mission Control Team #3 assumed their positions on-console. The STDN station acquisition-of-signal (AOS) and loss-of-signal (LOS) times are shown and are based on station receiver phase lock times. All real-time command (RTC) activity is shown also. This sequence is shown ending several hours after Team 2 relieved Team 3 on-console. Refer to Figure 3-1 for an orbital ground track of Rev 1503.

Pre-Santiago Pass

As Rev 1503 began, tape recorder #1 was commanded by onboard stored program command (SPC) to begin recording, and recorder #2 completed its recording cycle and was ready for playback to the STDN. All vehicle systems had looked normal to the flight controllers during the Orroral station pass at the end of Rev 1502.

Santiago Pass

The first indication of a problem became apparent to the MCT at the Santiago station acquisition-of-signal (AOS). The ACMO was advised of the following out-of-limit indications: both battery currents off-scale high (greater than 51 amps), both battery voltages 24.7 volts, the unregulated bus voltage 24.1 volts, the –Y solar array temperature off-scale high (greater than 860°F), and the +Y solar array temperature on-scale, but about 40°F offset above previous data. The AMCO was advised that all other spacecraft subsystems and the sensors indicated normal operation. Both battery temperatures were reading about 84°F, and the SADE temperature was reading 98°F, but this data was not reported to the ACMO. The battery current, battery voltage, and battery temperature indications were all out of limits at this time, exceeding the mission rule red-lines to be used by flight controllers in exercising documented responses to the appropriate anomalous conditions.

The team strongly suspected a ground data system problem because of the POCC data processing anomalies described previously. Many of these had display characteristics similar to those observed during the Rev 1503 Santiago pass, such as parameters off-scale. As a result, the ACMO requested that the POCC computer be reloaded. This was accomplished with the loss of 29 seconds of live data. Because the data indications did not change as a result, and because the SPAT personnel could not reconstruct or postulate an onboard failure mode that would be consistent with the telemetry indications, ground system problems were still suspected. Accordingly, the Santiago station real-time telemetry decommutation and computer telemetry processing programs were reloaded at the ACMO's request. This action resulted in a loss of 32 seconds of live data. As before, the data indications did not change after this action was completed.

Out-of-limit indications were still present to the MCT at Santiago loss-of-signal (LOS). The battery currents had returned back on-scale although they were very high, the battery voltages and the unregulated bus voltage were decreasing, and the battery temperatures were still abnormally high. Because the operators were unable to identify the specific anomaly, they felt that no spacecraft reconfiguration action should be taken; therefore, no RTC's were transmitted during this 13-minute pass.

Santiago Post-Pass

The flight controllers reviewed the "snapshot" data very intently after the Santiago pass, attempting to the best of their ability to determine the source of the problem, but were unable to do so. During this 16-minute period before Orroral AOS, the team reviewed a "snapshot" of the POCC computer convert tables made during the Santiago pass. These were the calibration data tables used to convert incoming raw data to processed data in engineering units. No discrepancies were noted, although insufficient time was available to do an in-depth analysis.

Orroral Pass

The Orroral station was not originally planned for Seasat support during Rev 1503, but due to the anomaly the ACMO had arranged through the GSFC operations support personnel that the station be available. It was a fringe pass with only about 9 minutes of real-time data available. At AOS, the anomalous telemetry indications were still present and continuing to diverge. The MCT decided to make one final attempt to rule out ground system problems by requesting the POCC computer operators to switch from the on-line disk storing the Seasat data base to the standby disk. This was attempted, but for some reason the computer would not run on the secondary disk, so the ACMO requested a return to the primary disk. This switching activity resulted in almost 4 minutes of lost live data (almost 50 percent of the total pass coverage).

The ACMO had planned a tape recorder playback at Orroral, but by then, there was insufficient time to do so. By Orroral LOS, the team was convinced that the problem was not a ground problem, but was within the spacecraft; because they could not locate it, however, they decided against any real-time safing command sequence transmission.

No further acquisition was made with Seasat after Orroral LOS.

Orroral Post-Pass

A playback of Orroral pass data about 15 minutes after Orroral LOS confirmed that the ground system was not at fault. At 43 minutes after Orroral LOS, the official declaration of Seasat spacecraft emergency was issued. The first transmission of a safing sequence occurred at about this same time, when Sequence 2 ("Sensor Power-Down to OFF") was transmitted without communications with the spacecraft during the predicted Merritt Island STDN station acquisition.

Safing sequences, along with the "No Downlink" command procedures, were transmitted repeatedly from then on until termination of mission support, to no avail.

At 08:30 GMT, the ACMO requested that the GSFC IPD process a "quick-look" housekeeping tape of the Orroral Rev 1502 tape recorder #1 dump data to play into the POCC computer in order to determine if any trends leading up to the failure were present, and to see if the problem occurred during the recording cycle. The tape was delivered to the POCC 12 hours later, and a playback into the POCC computer revealed no onboard anomalies to the flight controllers. In other words, the data through 01:37 GMT, Rev 1502, looked normal. The recorded data subsequent to that time (full cycle on tape recorder #2 and a partial cycle on tape recorder #1) was never dumped to the ground.

Revolution 1503 Observations – The telemetered indications from the Santiago and Orroral stations during Rev 1503 contained the symptoms of a serious failure in the Seasat spacecraft EPS. The MCT on duty, however, was unable to postulate a failure consistent with these indications in time to take possible corrective action. Out-of-limit indications were present for the low battery voltage, high battery temperature, and high current conditions, but no spacecraft safing action was taken in response to those indications. This was a technical violation of the mission rules ("responses to anomalous conditions") published in the Seasat mission operations documentation.

The MCT genuinely believed they were experiencing a ground system data processing problem rather than an actual spacecraft system failure. They were so convinced of this situation that several minutes of real-time telemetry data were lost while pursuing this belief. Contributing directly to this situation were the repeated POCC computer processing problems discussed previously, because many were similar in nature to those observed by the flight controllers during Rev 1503 (off-scale parameters, for example).

A prelaunch failure was experienced in the Seasat telemetry and sensor interface unit (TSU), which, if it were to occur in flight, would have resulted in anomalous data indications somewhat similar to those observed during Rev 1503. The flight controllers were aware of this failure, which certainly influenced their thinking during their attempt to determine the source of the out-of-limits indications.

Contributing to the lack of ability of the MCT to determine the failure mode was the minimal spacecraft subsystems (Agena bus) training received by the entire operations team. In addition, the SPAT member who was on duty serving as the AMA specialist in charge of electrical power was being cross-trained into that position during the time of the failure. His previous experience was in the Seasat attitude control subsystem and he had little, if any, background in the electrical power subsystem.

It was unfortunate that the tape recorder playback was not accomplished at Orroral from Rev 1503. From the standpoint of obtaining the recorded failure event, however, the wrong recorder probably would have been dumped. Tape recorder #2 completed its record cycle and was ready to be dumped about one-half hour after LOS at Orroral, Rev 1502. Tape recorder #1 began recording just before this and actually recorded the failure a few minutes later. All this activity happened out of station contact during the hour between LOS at Orroral, Rev 1502 and AOS at Santiago, Rev 1503. So not knowing where in this hour the failure occurred, the flight controllers would naturally have dumped recorder #2 which was full and in standby, awaiting playback, rather than recorder #1 which was still recording during the Rev 1503 Santiago and Orroral passes. (It takes longer for the ground to set up and dump a recorder still recording than it does for one which is full and in standby to dump.) Even with the minimum time specified for spacecraft "quick look" tape turnaround, the MCT would not have known in time that they had dumped the recorder that did not contain the failure event.

As it turned out, the Oak Hangar (UKO) European Space Agency (ESA) tracking station in England recorded the Seasat telemetry during the failure event; data covering the failure event was therefore available for post-flight failure analysis.

Mission Operations Conclusions

a. There is no indication that any aspects of mission operations were contributory to the Seasat Failure.

b. Although the MCT on duty during Rev 1503 technically violated mission rules by delaying any safing action to the spacecraft, for whatever reasons, it is the judgment of the Board that if this action had been taken during either the Santiago or Orroral station passes, given the existing depth of training, the documented contingency procedures, and the actual failure mode, the Seasat mission would not have been saved or partially saved.

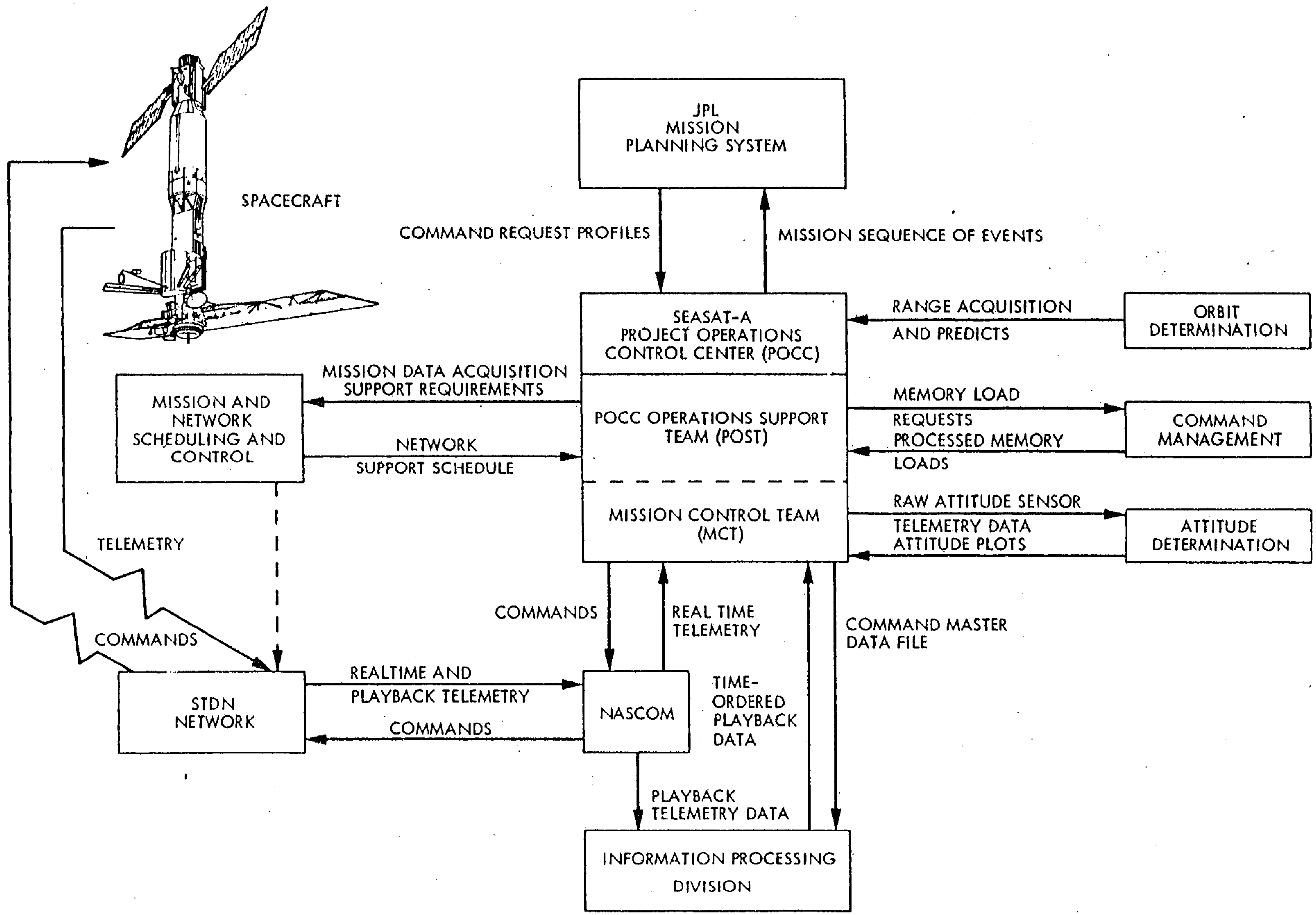

Figure 7-1. Seasat Mission Operations Interfaces

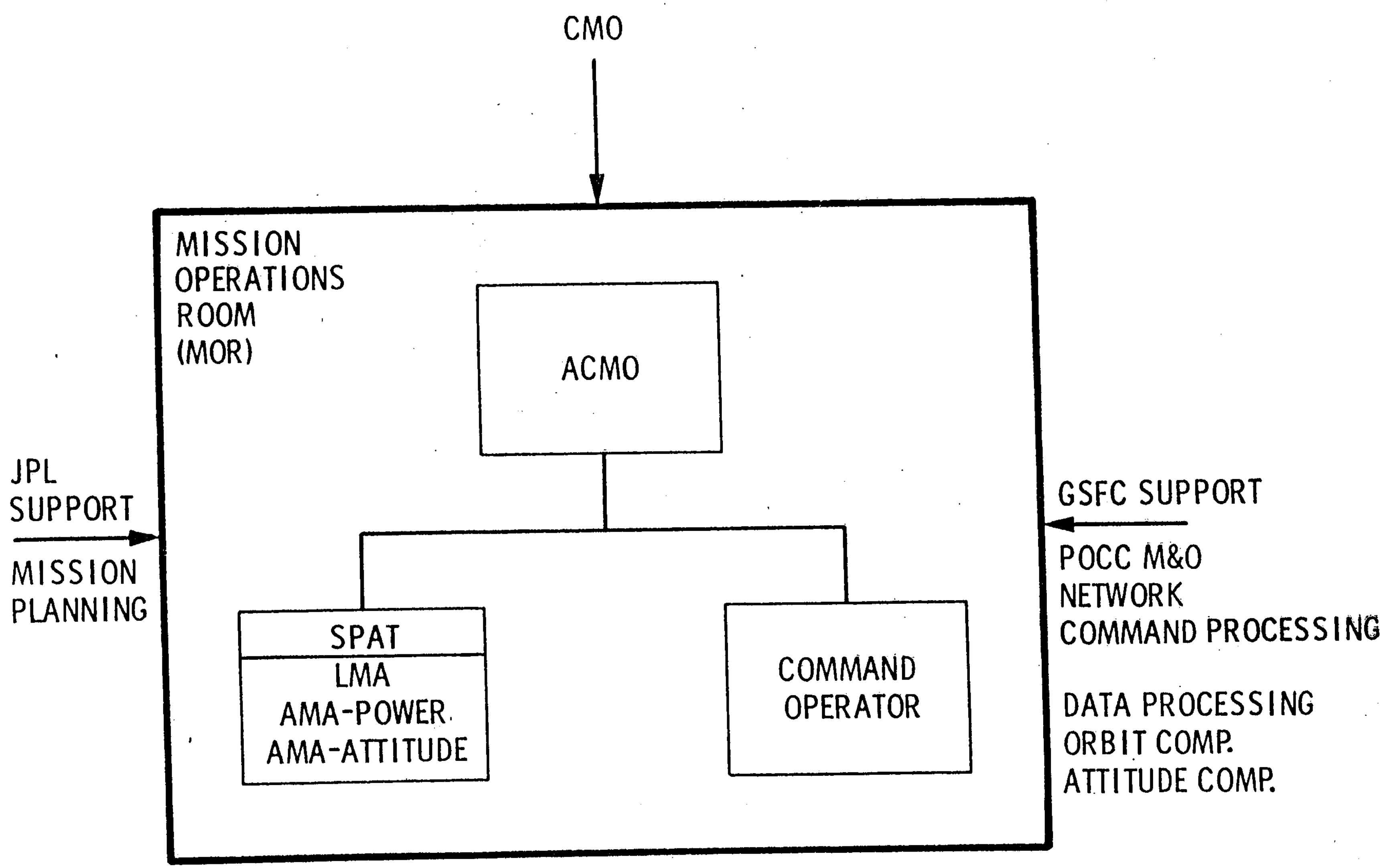

Figure 7-2. Seasat Real-Time Mission Control Team (MCT) (Typical of 4 Teams)

VIII – IDENTIFICATION OF THE FAILURE MODE

Isolation of the Failure

On revolution 1503 of the Seasat mission, the spacecraft data recorded at UKO showed that all subsystems were operating normally at AOS. At 03:06:13 GMT the spacecraft entered eclipse, and all subsystems continued to operate normally with the batteries assuming the spacecraft loads. At 03:12:02 GMT the unregulated bus voltage decreased from 29.2 to 25.2 VDC in approximately 17 seconds, coming to an essentially steady state condition. Battery currents increased from a normal value of 10 to 12 amps each to a full scale high value of 51 amps, with a single intermediate measurement on each battery of approximately 40 amps, Figure 8-1a. Battery voltage followed the unregulated bus voltage and battery temperatures increased sharply in response to a heavy current load of at least 100 amps placed upon them. Solar array temperatures exhibited abnormal behavior with the –Y solar array temperature indication abruptly going to full scale. The +Y solar array temperature indication assumed a steady state offset but continued to function normally, Figure 8-1b.

Examination of the principal power subsystem parameters displayed on an expanded time scale of 1 second per inch, in Figure 8-2, showed that the unregulated bus voltage decreased in a successive series of steps over the first 17 seconds of the event. The battery current indication of 40 amps per battery occurred during the first step of the unregulated bus voltage drop. Unregulated bus current, however, indicated no change in characteristics after the fault. Structure current showed minor abnormal behavior at the initiation of the unregulated bus voltage drop, and again 12 seconds later, and abruptly returned to normal. Except for these two relatively minor and short lived events, the structure current showed no permanent abnormal behavior.

Examination of the EPS high bit rate data displayed at 0.250 seconds per inch, in Figure 8-3, reveals that the initiation of the fault resulted in an unregulated bus voltage drop from 29.06 to 28.2 VDC that lasted for approximately 600 milliseconds. Unregulated bus current indicated only one abnormal data point. A single bit error in the most significant bit of this data word would, however, bring this current indication to within the normal pattern of the unregulated bus current. Also, several data indications of the structure current show slightly different behavior during the initiation of the fault, but these differences were minor compared to the major fault current of 100 amps or more. These flight data clearly indicate that a massive and progressive short circuit occurred within the EPS. The short circuit did not involve the unregulated bus or cause any significant permanent change in structure current indicating that a short circuit did not occur in any of the spacecraft loads, and no short occurred to ground. The short, therefore, occurred between the positive and negative power lines within the EPS.

These conditions existed in the EPS as the spacecraft passed out of view of UKO approximately 1 minute after the initiation of the failure.

AOS at the AGO station occurred at approximately 03:30 GMT, or 17 minutes later. The unregulated bus voltage had dropped to 24.1 VDC and the bus current was normal for this voltage. Battery voltages had also dropped, battery currents were still full scale high, and battery temperatures were high (85°F) and rising. These data indicated a continuing massive fault that was now essentially steady state. The solar array temperature indications remained abnormal with -Y full scale high and +Y operating normally but with a 40°F high offset. The temperature of the SADE had increased to 97.5°F and continued to rise. During the AGO pass, both 1K2 and 2K2 relays opened, a normal response to the high battery temperatures. The opening of 2K2 coincided with an increase in battery and unregulated bus voltages, which indicated that a portion of the short had been relieved. About 1 minute before LOS at AGO, the spacecraft emerged from eclipse, the solar array began sharing the loads, and the battery currents came back on scale but were still very high. All subsystems of the spacecraft except for the power subsystem were normal throughout the AGO pass.

As indicated, the SADE temperature during the AGO pass was elevated and slowly rising at a rate of 1.53°F/min. The SADE was thermally isolated from the structure through vibration isolators and from solar radiation by multilayer insulation. Analysis of the SADE thermal characteristics indicated it would require between 50 and 70 watts of thermal energy to produce the indicated temperature rise rate. The SADE continued to function normally, thus producing no excessive thermal dissipation. As no major power cables passed through the SADE, there was no source available internal to the SADE to produce the required heat. The -Y solar array cable bundle, however, ran the length of the SADE in close proximity to it and underneath the multilayer thermal blanket. A thermal analysis indicated that a fault current of the indicated magnitude passing through four to eight wires of this cable harness would produce sufficient thermal energy to cause the temperature rise rate exhibited by the SADE. This led to the conclusion that the fault current passed through the cable adjacent to the SADE. The fault must, therefore, have occurred on the solar array side of the SADE, because if it had been on the vehicle side, no fault current would have been flowing through the section of the cable adjacent to the SADE and no temperature rise would have resulted.

Localization of the fault to the solar array side of the SADE placed the fault within the cable between the SADE and the -Y slip ring assembly, within the -Y slip ring assembly, or on the solar array or in its cabling. The cable run between the SADE and the slip ring assembly was a short, straight cable of Kapton insulated wire with no splices. A short here would have to have been a wire-to-wire short which is highly unlikely with Kapton insulated wire. In addition, no reasonable propagation mechanism could be postulated for a wire-to-wire short; this section of cable was therefore not suspect.

Flight data of the solar array positions showed that both solar arrays continued to be driven at the programmed rate until considerably after the initial event indicating that both solar array drive assemblies were functioning normally; the -Y solar array drive assembly was therefore not suspect.

Examination of the remainder of the solar array layout indicated three potential shorting possibilities. These were: (1) solar cell string-to-battery return; (2) isolation diode on panel-to-battery return cable; and (3) a wire-to-wire short in the cable. All of these were examined to see if they could match the initial fault flight data but none of the potential failure modes on the array could be made to fit. In addition, the construction of the arrays was such that none of the potential failure modes were considered likely, and no propagation mechanism could be found in any of these areas. It was concluded that the fault was not on the solar array or in its cables.

The elimination of all other elements of the EPS localized the fault to the slip ring assembly. The initial event was then characterized by using resistance values corresponding with the spacecraft cable length and the observed flight data values for current and voltage. Also, the design of the EPS was such that the positive and negative wires in the vehicle cable run were equal in length with the same number of connectors in each line and therefore they had the same electrical resistance. The known electrical resistance in the positive and negative cables with the unknown short resistance between them form a voltage divider, and since the flight data provided the voltage and current of this electrical circuit, the unknown fault resistance and voltage drop could be easily determined. The resulting parameters were as follows:

Fault current	60 amps
Fault voltage	10 volts
Fault dissipation	600 watts
Fault duration	600 ms
Fault resistance	0.2 ohms

Further investigations were directed toward finding a probable initiating mechanism within this assembly.

Slip Ring Failure Modes

A detailed review of the slip ring construction, assembly procedures, and failure history permitted the synthesis of three possible failure modes within the slip ring assembly which were as follows: a wire-to-brush assembly short; a short between adjacent brush assemblies; and a short by particle contamination.

Wire-to-brush assembly short – In this failure mode, the Teflon insulated wire could press against a brush assembly in a way to cause cold flow of the insulation and ultimately cause a short. The application of this failure mode to the fault parameters had two shortcomings. First, a short of high impedance would not be likely, and secondly, the short must dissipate 600 watts of power. A thermal analysis indicated that, if this amount of power were to be dissipated between the wire and the brush spring, the wire would be heated to its melting point in significantly less time than the duration of the observed initial short. Such a short would then burn the wire open, with the shorted wire acting as a fuse. Therefore, while this failure mode could possibly cause the initial shorting event, some additional mechanism would have to be triggered by this mode which could explain the observed flight data. As will be discussed below, this additional mechanism could be a sustained electrical arc.

Contact short between adjacent brush assemblies – As discussed in Chapter V, the clearances between brush assemblies are small at best, and slight misalignments could make these distances even smaller. Figure 8-4 is a photograph of a non-flight Seasat slip ring assembly showing the brush assignments for brushes 1 through 15. The large areas of overlap between the wide brush assemblies and the clearances between adjacent brush assemblies can be seen easily. As noted in Figure 8-4, most of the brush assemblies are wired such that opposite polarities exist on adjacent brush assemblies. Contact between adjacent brushes is considered quite possible and, if such contact occurred between brush assemblies of opposite polarity, this could trigger an arcing short having the characteristics of the initial fault.

Short by particle contamination – Ground tests of slip ring assemblies using sintered composite of silver, molydisulfide, and graphite brushes have demonstrated that debris produced by brush wear can cause failures due to shorting. One failure mode is caused by relatively small debris, principally silver, building up on the insulating barriers between adjacent brushes and producing a conductive path. In this case, a very small deviation from nominal brush position, for any reason, could result in brushes contacting the dielectric material between rings and severely aggravating wear rates. Also, large debris particles that were either undetected before launch or generated by malfunction cannot be dismissed as possible failure modes. This debris could introduce a shorting condition between closely spaced adjacent parts of opposite polarity as was the case for the Seasat slip ring assemblies. Debris-caused shorting has been observed in high speed slip ring applications where many rotations are experienced. The Seasat application was very low speed and the slip ring assemblies experienced a relatively low number of rotations before mission termination; however, the generation of some amount of debris would have been normal and the amount could have been exaggerated by minor faults. This debris or contamination could have led to the above described shorting or could have exacerbated the previously described possibility of shorting between misaligned brush assemblies. If any one of these debris-induced shorts produced electrical arcs, it could explain the observed flight data.

In summary, any of the above described slip ring failure modes could produce an electrical short between either adjacent brush assemblies or adjacent rings having opposite polarity. By themselves, however, none of these shorts could produce the flight data observed during the initial shorting event unless it was a trigger mechanism for a sustained electrical arc. Additionally, this arc would have to lead to still other shorts involving additional circuits which would provide the current carrying capability to ultimately drain the batteries and cause failure of the EPS. That arc initiation is in fact possible due to at least one of the above described failure modes was proven conclusively in a laboratory test program described in a report from Sperry Flight Systems. The subsequent sections of this chapter describe the results of this test program, compare the results with the Seasat flight data, and discuss the possibility of sequential propagation of the arc induced fault. While the Sperry study describes arc initiation caused by relatively large debris particles, the Board considers it possible that similar arcs might also be initiated within the slip ring assembly by either wire-to-brush assembly shorts, contact shorts between adjacent brush assemblies, or from abnormal amounts of small particle debris.

Arc Initiation and Maintenance

Sperry Flight Systems[1], Phoenix, Arizona, who has used slip rings with the same brush material, brush spring material, and similar brush-to-brush clearances has conducted an investigation of a failure mechanism that can be initiated by contaminants. Tests were run in a fixture containing simulated slip ring brush assembly springs, with clearances similar to the Seasat flight hardware (Figure 8-5). The purpose of tests conducted with this apparatus was to determine the nature and extent of arcing shorts that might be initiated by debris.

A schematic drawing of two adjacent slip ring and brush assemblies is shown in Figure 8-6. Low level steady state currents pass in opposite directions in the two brush leads and brushes. If a short is introduced across these two leads, the total current in each lead will increase significantly and a

[1]Substantial portions of this section were derived directly from Sperry Flight Systems, Phoenix, Arizona, Report No. 120181, "Effects of Material Wear Debris on Power/Slip Ring Components in Vacuum," by Mr. P. E. Jacobson, April 1978, and are used herein with the permission of Sperry Flight Systems.

three-dimensional magnetic field will be set up around the conducting elements as shown in the figure. As can be seen, the particle will be forced out of the magnetic field set up in the vicinity of the brush leads by reason of its own field developed by the short circuit. This mechanism would exist even if the two lead wire polarities were reversed.

The inherent particle ejection would be desirable were it not for an arcing phenomenon that is triggered by this ejection. As the shorting particle is ejected, it is possible that an electrical arc may be initiated which would result in more severe damage to the circuit elements than the particle itself could produce. The mechanism of this arc buildup and decay is described in the literature and is summarized in the following paragraphs.

As a shorting particle between adjacent brush leads separates (particle ejection), the temperature of the material adjacent to the point of contact increases to the melting point due to very high local current density resulting in extremely high local temperatures. This phenomenon is common in make/break power circuits. Figure 8-7 shows the sequence of events that occurs when a shorting particle bridges opposite polarity power elements. In Figure 8-7(a), the shorting particle produces the short circuit path that in turn creates high local heating and melting. As the magnetic forces begin to eject the particle, the fluid material is drawn out by surface tension with further separation as shown in Figure 8-7(b). This is known as the bridge stage of the separation. After complete particle separation from the cathode, the cathode cools and final rupture of the liquid metal leaves a buildup of material at the cathode. The temperature of the bridge formed at the cathode can be high enough to cause thermionic emission of electrons from the cathode to the anode. This electron flow continues the short circuit path even after the initial shorting particle is no longer present. The continuation of the arc, through the thermionically emitted electron stream, continues to heat the anode which locally melts and produces an ionized gas environment providing the necessary ingredients of an arc as shown in Figure 8-7(c). The anode will then sacrificially feed the arc until the arc length becomes long enough, several inches in vacuum, to reduce the current to a threshold below which the arc cannot be sustained. The destructive nature of this mechanism is obvious.

A fully developed arc has a voltage distribution across the arc as shown in Figure 8-8. Although the exact profile is a function of the materials involved, the figure approximates the conditions which prevail. Of primary importance is the fact that the major elements of the total voltage drop are the cathode plus anode drops which remain fairly constant. This means that the ionized metal vapor path (the positive column which is visible in any arc) manifests a relatively small voltage drop and is almost constant in magnitude as the arc progresses. The characteristic voltage drop is generally in the order of 10 to 20 volts. The arc will die only when the path length is sufficiently long and the resistance sufficiently high that the current falls below a required threshold. Every material has its own minimum arc voltage and current for given arc gap lengths and ambient pressure. These minimum conditions must be exceeded in order to initiate and sustain an arc. The normal voltages and currents within the Seasat slip rings both exceed these minimums.

Some important characteristics of this arcing mechanism related to sets of adjacent slip ring brush assemblies are identified as follows:

a. The arc will not occur if separation of the particle occurs at the anode first.

b. The arc may be initiated by a small particle because it must only conduct the threshold current.

The initial series of tests were conducted by placing simulated contaminants of silver 0.006 to 0.009 inch square and 0.125 inch long across the two fixture leads and then closing a circuit switch to place a nominal 30 volts between them. Peak currents less than 8 amps ejected the particles off the simulated brush springs with no damage to the particle, and caused only superficial pot-marking to the simulated brush springs. The voltage drop across the fault was negligible.

The second series of tests was conducted in vacuum on the same apparatus with a nominal 30 VDC impressed across the simulated slip ring brush springs before the contaminant was made to cause a short between them. In these tests, the contaminant was introduced across the simulated brush springs in such a way that the particle would be ejected by the resulting short circuit cathode side first. In each of the four tests, the shorting particle was ejected without being damaged, and a severe arc was initiated. The arc characteristics were recorded on high speed strip charts, and the observed arc characteristics matched the arcing theory very closely. In every case severe damage was caused to the anode of the simulated slip ring brush spring.

Figure 8-9 is a summary of the results of the arcing tests with the Seasat flight data included for comparison. The arc initiation tests that were carried on with materials, clearances, and voltage levels similar to those of Seasat, produced test data that closely matched the Seasat flight data. These data provide strong support to the conclusion that the initial fault was an arcing short between adjacent slip ring brush assemblies of opposite polarity in the slip ring assembly.

While the failure mode which triggered the arc in the Sperry study was a relatively long debris particle, it was felt by the Board that other failure modes within the slip ring assembly could also provide the trigger. The mechanism of arc initiation requires high thermal and voltage gradients together with high local temperature sufficient to produce thermionic emission. Such high gradients together with a high rate of local heating may be caused when one or more sharp irregularities on surfaces at different voltages come in contact. This is possible if either a wire carrying current of sufficient DC voltage makes contact with a sharp edge on a brush spring assembly at a lower voltage or if adjacent brush spring assemblies having sharp edges or points make contact. Therefore, a short caused by any of the previously described failure modes which precipitated a sustained arc must be considered a prime possibility for the initial failure event.

Sequential Propagation

Included in the Sperry investigations were 4,000 frames per second films of the arcing as it progressed. One of these films taken of test no. 3 in the test summary, Figure 8-9, was viewed by the Board. The onset of the arc could clearly be seen and two important characteristics were observed. First, the arc progressed with explosive violence. Shortly after the initiation of the arc by a 0.011 inch square × 0.125 inch long silver rich simulated contaminant, relatively large particles of molten material were seen being violently ejected from the arc area. This molten material, confirmed later to be beryllium copper that was being melted due to the high local heating, was being ejected and deposited within the test chamber. A second phenomenon that was observed was the fact that as the arc continued, the film got progressively darker. This phenomenon was later found to be due to vapor deposition of the beryllium copper lead material on the view port of the vacuum test chamber. Both of the observed phenomena would contribute to the propagation of shorts within the slip ring assembly. The placing of opposite polarities on adjacent brushes enhances the possibility of progressive shorts due to vapor deposition of conductive material or by the deposition of the molten material being ejected by the arc. It can be concluded that an arcing short in the slip rings would probably progress along the assembly until a propagation barrier was reached.

The violent progress of the arc would not only propagate additional short circuits but would cause additional abnormal behavior of the spacecraft circuits in the vicinity of the arc. The vaporization of the initial arc anode could easily account for the short-term ramp rise in structure current shown in Figure 8-2. Vapor deposition of the conductive anode material could reasonably produce a circuit path to structure that would be quickly destroyed by the resulting short circuit current heating this thin conductive deposition path to its melting point.

High temperature would also be produced in the vicinity of the propagating shorts, resulting in softening or melting of the wire insulation which could cause abnormal indications if the wires shorted. The abnormal solar array temperature indications could be caused by the -Y solar array thermistor lead being shorted to a potential greater than its full scale output of 5.1 VDC which would then produce the high off-scale reading observed in the flight data. If this shorted voltage were 10 to 14 VDC, the +Y solar array temperature would also be affected through the common solar array temperature signal conditioning in the SADE that would produce the steady state +40°F offset observed in the flight data. Thus the high temperature and destructive nature of the propagating arc can account for all of the abnormal behavior observed in the UKO flight data.

Flight data acquired during the AKO and ORR passes were examined thoroughly. None of the data were inconsistent with a short within the -Y slip ring assembly that was initially progressive and massive in nature and finally reached a steady state condition. There were no data that indicated any abnormal behavior within the MPCDU and both Current Charge Controllers operated as designed by opening their respective K1 and K2 relays in response to the high battery temperatures.

The activity of the CCC's K1 and K2 relays helped identify the region of the slip ring where the short most likely occurred. As covered in Chapter V, the K1 and K2 relays disconnect certain solar array panels from the battery bus when open. In the -Y array, 1K2 controls panels 4 and 5, while 2K2 controls panels 6, 7, and 8. At the time of the initial failure event, 2K2 was open; thus, the circuits to panels 6, 7, and 8 could not have been involved in the initial fault. This relay closed very shortly after the initial fault and remained closed until the AGO pass.

During the AGO pass, 1K2 opened, disconnecting array panels 4 and 5 with no effect on any voltages or currents, indicating that the circuits to these array panels were not involved in the "final" short. When 2K2 opened, however, a reduction in fault current of 5 to 10 amps occurred. Thus, a part of the steady state short clearly involved the circuits to panels 6, 7, and 8, which had *not* been connected at the time of the initial fault. At LOS at AGO, a major short still existed with the possibility that the circuits to panels 1, 2, 3, 9, 10, and 11 might be involved.

Brush assembly connections within the slip ring assembly were such that the circuits for -Y array panels 8 through 11 were grouped on adjacent brush assemblies no. 1 through 11. The circuits for panels 1, 2, and 3 were for the most part at the other end of the slip ring assembly and were not on adjacent brush assembly pairs. Thus an initial arcing short between any adjacent brush assemblies 1 through 8, which connected panels 9, 10, 11, and the return of panel 8 is possible, and subsequent propagation to other brush assemblies is quite reasonable, and is the most probable place for the failure to have occurred. It is noted that brushes 12 and 13 were not connected and could, therefore, serve as a barrier to further fault propagation.

Summary of Failure Mode Identification

In summary, the Seasat flight failure occurred in the slip ring assembly of the -Y array. The initial fault was probably an arcing short between adjacent components within the slip ring assembly initiated either by a contaminant, by physical contact of adjacent brush and ring assemblies, or by a wire-to-brush assembly short. The explosive and destructive nature of the arc caused propagation along the slip rings adjacent to the initial fault.

The initial fault would have occurred between any two adjacent brush and ring assemblies of opposite polarity in the region of brush assemblies 1 through 11. The application of the slip ring assembly with plus power and negative return power being carried essentially on every alternate ring not only created a high potential for the failure mode that occurred, but also allowed the fault to progress along adjacent brush assemblies and rings until it reached a barrier such as unused brush assemblies. All other spacecraft systems functioned normally as long as adequate voltage levels were maintained.

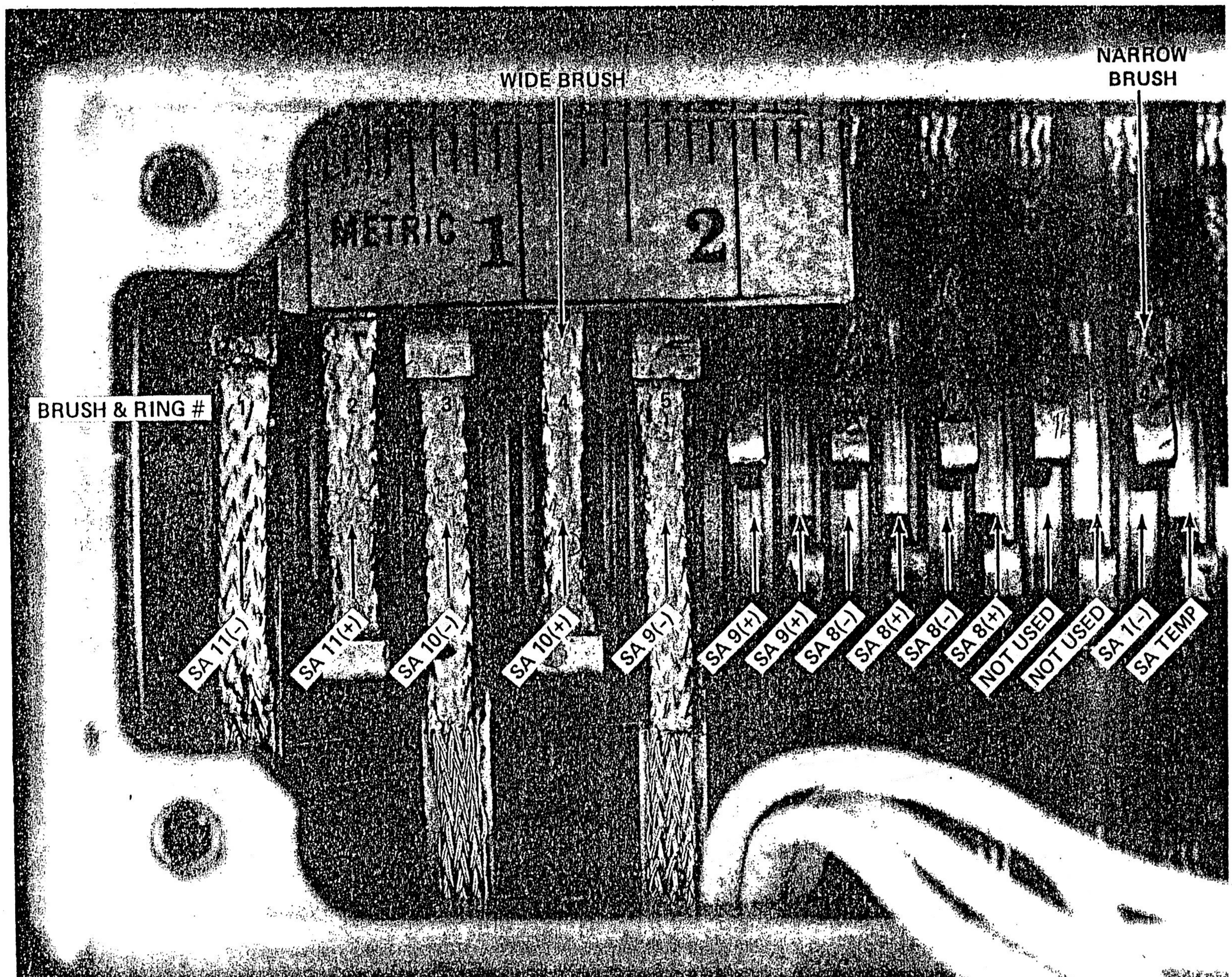

Figure 8-4. Slip Ring Assembly - Brushes 1-15 (Side Cover Removed for Viewing)

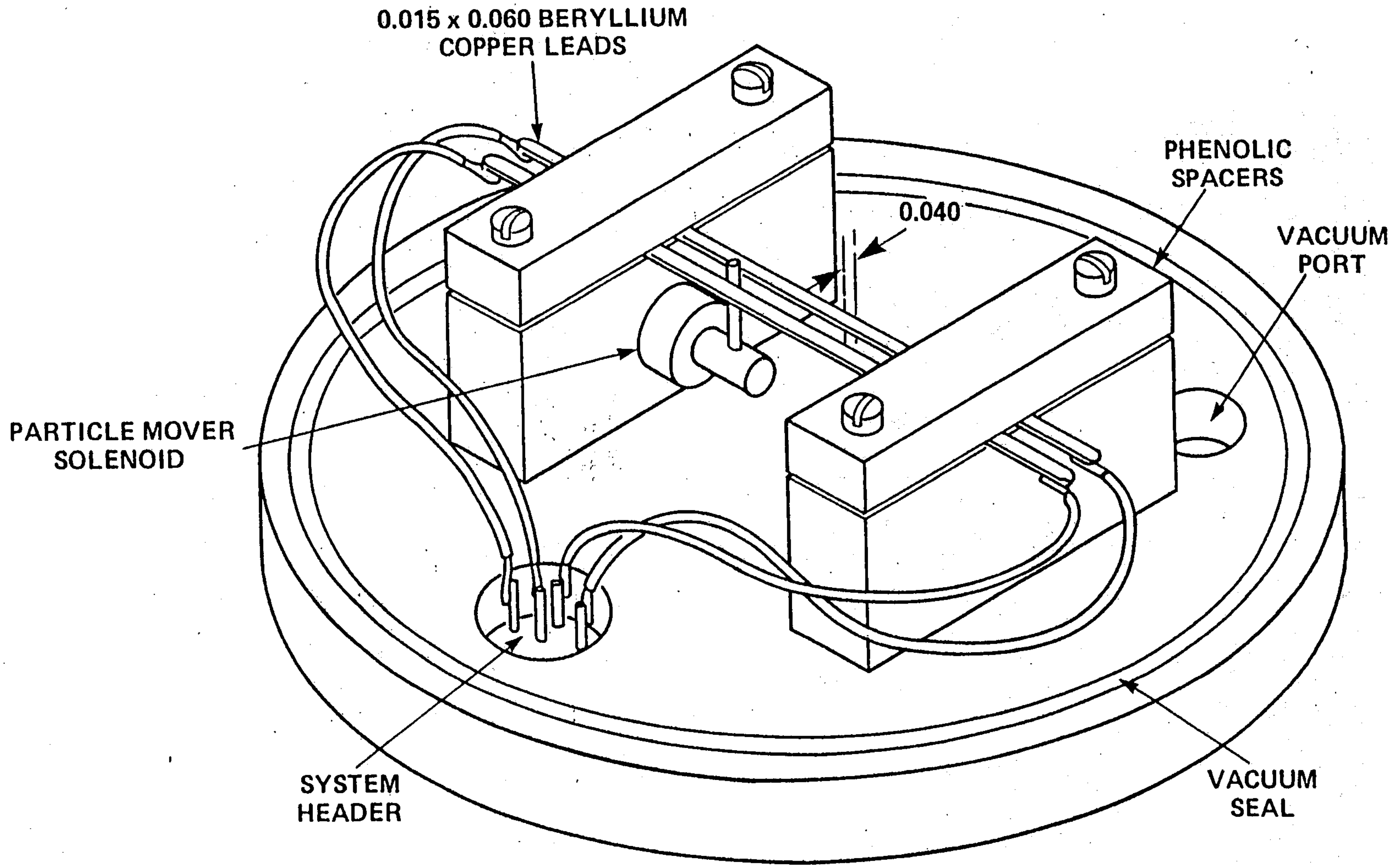

Figure 8-5. Simulation Fixture for Debris Tests

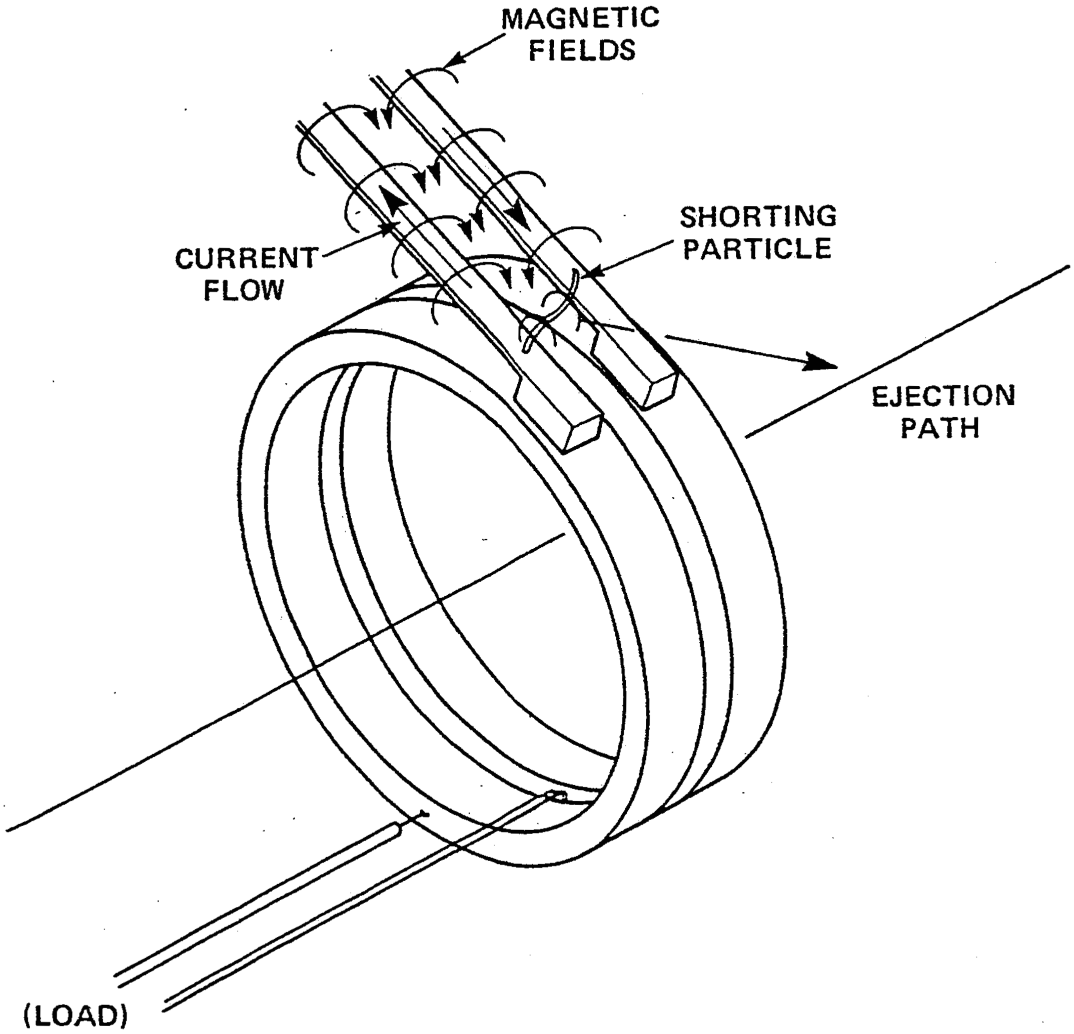

Figure 8-6. Shorting Particle Ejection

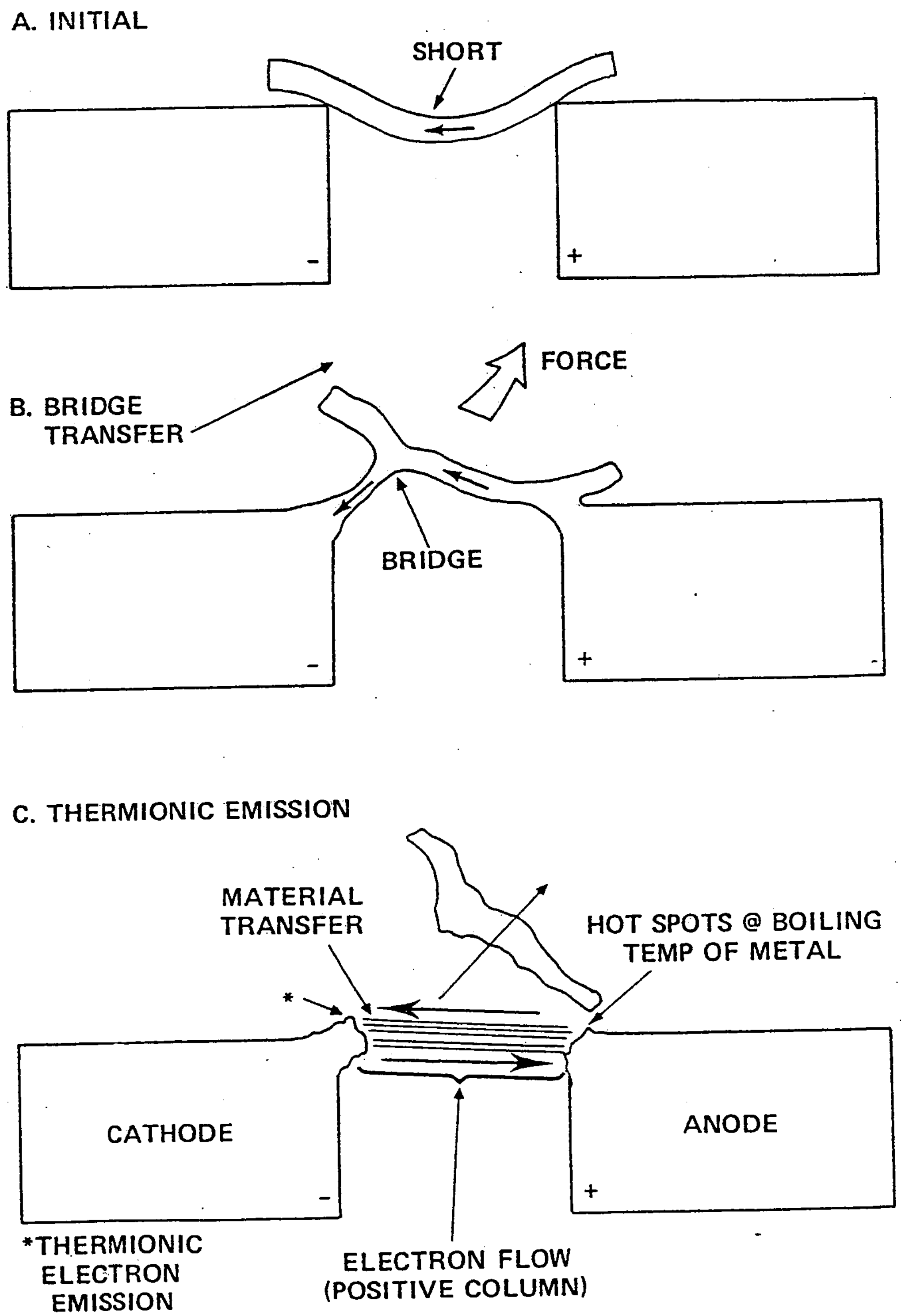

Figure 8-7. Separating Arc Sequence

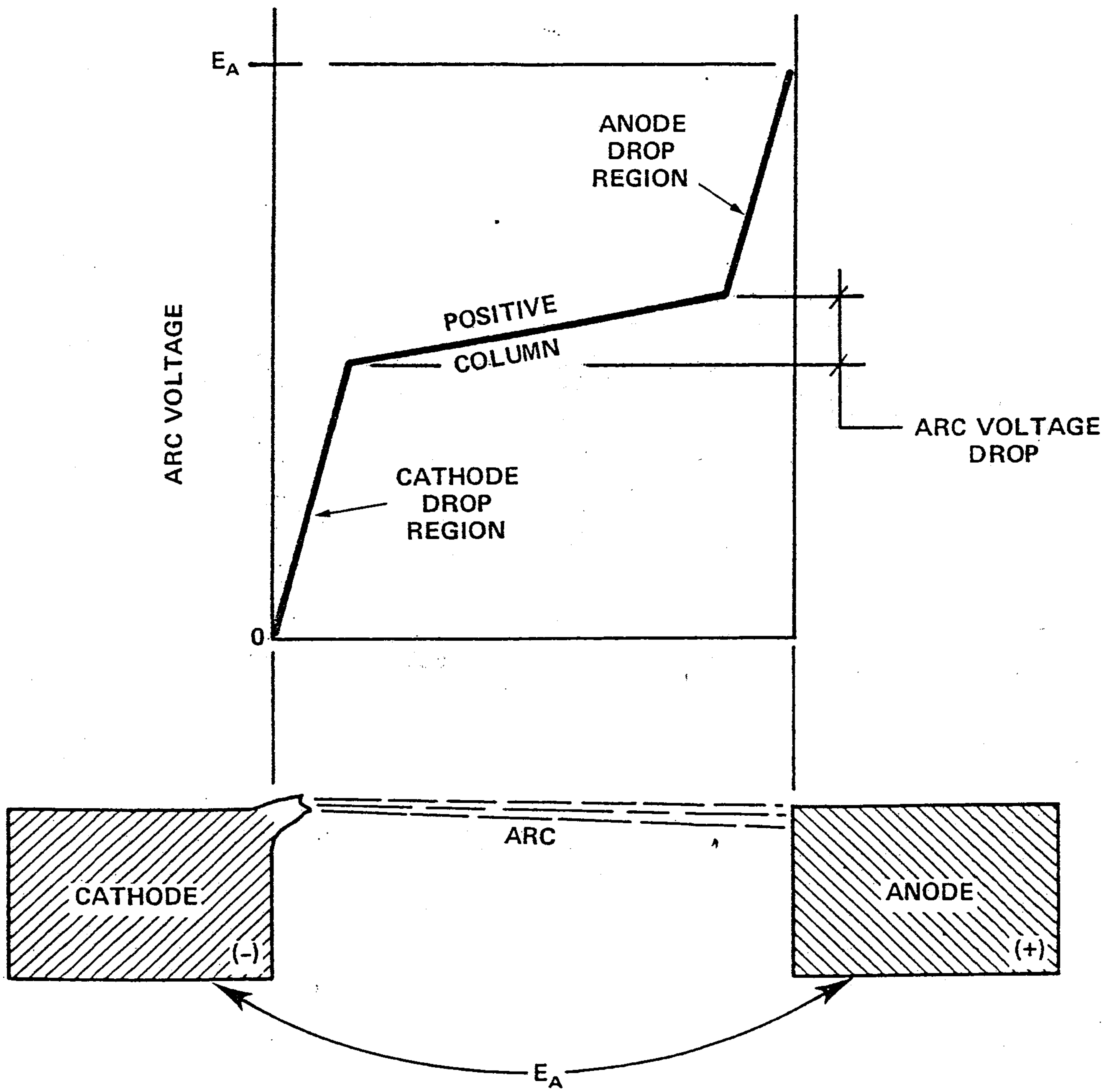

Figure 8-8. Voltage Within Cathode – Anode Arc

Test No.	Arc ΔV volts-DC	Arc Current amps	Peak Current amps	Arc Duration ms	Power Dissipation watts	Lead Material Loss inches
1	-	-	65	400	950	0.6
2	14	50	62	85	570	0.5
3	14	50	64	275	700	-
4	10	50	96	600	600	totally consumed
Seasat Flight Data	10	60	not known	600	600	not known

Figure 8-9. Sustained Arc Data in Vacuum Nominal 30 Volts-DC System

IX – SIGNIFICANT FINDINGS

1. The spacecraft failure that occurred on October 9, 1978, was due to a loss of electrical power in the Agena bus as a result of a massive and progressive electrical short within the slip ring assembly of the –Y solar array.

2. The electrical short was most probably initiated by an arc between adjacent components in the slip ring assembly. Possible triggering mechanisms for this arc are momentary shorts caused by wire-to-brush assembly contact, brush-to-brush contact, or by a contaminant.

3. The congested nature of the slip ring design, coupled with a wiring arrangement for connecting the slip rings into the power subsystem that resulted in most of the adjacent brush assemblies being of opposite polarity, made the Seasat slip ring assembly particularly prone to shorting.

4. The combination of design and wiring sequence used for the Seasat slip ring assemblies made these units unique, first-of-a-kind components.

5. The possibility of slip ring failures resulting from placing opposite electrical polarities on adjacent brush assemblies was known at least as early as the summer of 1977 to other projects within the prime contractor's organization. That the Seasat organization was not fully aware of these potential failure modes was due to a breakdown in communications within the contractor's organization.

6. The failure to recognize the potential failure modes of the slip ring assembly and to give this critical component the attention it deserved was due, in large part, to the underlying program policy and pervasive view that it was an existing component of a well-proven and extensively used standard Agena bus. This program policy further led to a concentration by project management on the sensors and sensor module of the spacecraft to the near exclusion of the bus subsystems. In actuality, many of these subsystems, including the power subsystem, contained components that were neither flight proven nor truly qualified by similarity.

7. Lack of proper attention by both LMSC and JPL Seasat program engineering to the new and unproven components on the Agena bus resulted in several instances of both noncompliance with contractual, qualification and acceptance requirements and failure to document such noncompliances.

8. The Failure Modes, Effects, and Criticality Analysis that was conducted for the electrical power subsystem did not consider shorts as a failure mode and thus did not reveal the presence of single point failure modes in the subsystem nor provide a basis for the development of a full complement of safing command sequences that could be used by the flight controllers in responding to anomalies.

9. The strong desire on the part of all concerned to initiate the project as soon as possible resulted in inadequate time for an effective Phase B study. As a result, the project office did not have the opportunity to thoughtfully plan the activity and establish the preliminary designs, component evaluations, test plans, and other Phase B project plans before becoming engaged in the actual spacecraft development.

Although unrelated to the failure of the Seasat, certain deficiencies in flight control procedures were present that are worthy of note as a lesson for the future. The flight controllers were not provided with an adequate set of safing command sequences to use in response to anomalies, were not sufficiently familiar with the system they were controlling, received insufficient anomaly training and, during the failure event itself, failed to follow the prescribed procedures in response to the flight data available to them. Compounding these difficulties were the frequent breakdowns of the ground data acquisition and processing system throughout the mission.

X – CONCLUDING OBSERVATIONS

It is ironic, and yet typical, of spacecraft failures that the termination of the Seasat flight was caused not by a malfunction of a new or sophisticated device, but by a failure in a very common component of a type that has flown in many spacecraft for many years. It is also ironic, and instructive, that the smallest of events or the slightest of communications could have prevented the failure. Better clarity in an oral communication, a brief memorandum of the right kind at the right time, a failure report coming to the right person, or an alert engineer could have made all the difference.

Basic to the Seasat mission was the concept of using an existing, flight-proven spacecraft bus for the services and housekeeping functions required by the sensors in order to minimize program costs and to permit a concentration of effort on the sensors and their integration into the spacecraft. Thus the use of a "standard Agena bus" as part of the Seasat spacecraft became an enduring tenet of the program. So firmly rooted was this principle in program philosophy that, even after the engineering staffs of both the Government and the contractor were well aware of the final uniqueness of their Agena bus, the words, and the associated way of doing business, persisted. They became deceived by their own words.

Consistent with the concept of the "standard Agena bus" was the policy decision to minimize testing and documentation, to qualify components by similarity wherever possible and to minimize the penetration into the Agena bus by the Government. As a result, a test was waived without proper approval, important component failures were not reported to project management, compliance with specifications was weak, and flight controllers were inadequately prepared for their task. Significantly, the Seasat slip ring assembly had no applicable flight history at the time of its launch and, in its application to the spacecraft, was a new device.

There can, of course, be no quarrel with the policy of using existing and well proven equipment. The use of such equipment has certainly reduced the costs and contributed to the success of many space missions. But the world of space flight is an unforgiving one and words like "standard," "existing," and "similar to" can be traps for the unwary. The technical risks of using "standard equipment" can be as high as those present in a new or untried piece of equipment, but the approach, both technical and managerial, must be different. For new equipment, one designs carefully, reviews thoroughly, and tests completely–and that we know how to do. For "standard" equipment, one should diligently and thoroughly probe the heritage that justifies the classification and identify, component by component and piece by piece, those that are truly "standard" and those that are not. One should assume that each space vehicle is unique until proven otherwise. Then, for those parts that are standard or well proven, and that are applied in the same way, one can forego design reviews, testing and extensive documentation. Conversely, components that are different should be treated as new. The policy of limited penetration into Seasat's Agena bus by the Government was appropriate, but a limited penetration must be a selective penetration and not a reduced effort everywhere.

This identification of the heritage of previously used equipment, in both design and application, need not require a large staff or a lot of money. But it does take time, both at the start of the project and at the time of the Critical Design Review. And here, responding to strong desires by all concerned to get the project on contract and underway, the Seasat Project was denied the advantage of an effective Phase B study. Had there been an effective Phase B study period, preliminary designs would have been completed, component selections better understood, test plans and qualification requirements better established, and possibly, the critical role and inherent complexities of the slip ring assembly might have been more apparent to the Seasat engineering staffs. Whether such a Phase B study period would have precluded the Seasat failure is, of course, uncertain for history does not reveal its alternatives. But such a carefully conducted planning and study period would have minimized the chances for the type of failure that did occur.

The policy of using existing, flight-proven equipment can be both valid and cost effective. But it is the main lesson of Seasat that an uncritical acceptance of such classifications as "standard" can submerge important differences from previously used equipment in both design and in application. It is important, therefore, that thorough planning be conducted at the start of a project to fully evaluate the heritage of such equipment, to identify those that are standard and those that are not, and to establish project plans and procedures that enable the system to be penetrated in a selective manner.

APPENDIX A

- CHARTER OF SEASAT MISSION FAILURE INVESTIGATION BOARD
- LETTER OF AUTHORIZATION TO THE BOARD CHAIRMAN
- MEMBERSHIP OF THE BOARD

NASA Notice
National Aeronautics and Space Administration

NN 1150
Date October 16, 1978

Responsible Office: E/Office of Space and Terrestrial Applications

Subject: SEASAT MISSION FAILURE INVESTIGATION BOARD

1. PURPOSE

This Notice establishes the Seasat Mission Failure Investigation Board and sets forth its responsibilities and membership.

2. ESTABLISHMENT

a. The Seasat Mission Failure Investigation Board is hereby established because it is in the public interest to determine the actual or probable cause(s) of the Seasat mission failure.

b. The Board is a "project-oriented technical team" as defined in paragraph 4d of NMI 1150.1B.

c. The Chairman of the Board will report to the Deputy Administrator.

3. AUTHORITIES AND RESPONSIBILITIES

a. The Board will:

(1) Determine the actual or probable cause(s) of the Seasat mission failure and will document the technical and management history of such cause or causes.

(2) Provide a final written report to the Deputy Administrator.

(3) Carry out any other responsibilities that may be requested by the Deputy Administrator.

b. The Board may:

(1) Obtain and analyze whatever evidence, facts, and opinions it considers relevant, using reports of studies, findings, recommendations, and other actions

by NASA officials and contractors and by conducting inquiries, hearings, tests, and other actions de novo--in doing so, it may take testimony and receive statements from witnesses.

(2) Impound property, equipment and records to the extent that it considers necessary.

c. The Chairman will:

(1) Conduct Board activities in accordance with the provisions of this Notice and any other instructions that the Deputy Administrator may issue.

(2) Establish and document, to the extent considered necessary, rules and procedures for the organization and operation of the Board, including any subgroups, and for the format and content of oral or written reports to the Board and by it.

(3) Designate any representatives, consultants, experts, liaison officers, or other individuals who may be required to support the activities of the Board and define their duties and responsibilities.

(4) Establish and announce a target data for submitting a final report and keep the Deputy Administrator informed of the Board's plans, progress, and findings. All public release of information on ongoing Board activities will be made by the Deputy Administrator or, with his approval, by the Chairman of the Board.

(5) Designate another member of the Board to act as Chairman in his absence.

4. MEMBERSHIP

The Chairman, members of the Board, observers and supporting staff are designated in Attachment A.

5. MEETINGS

The Chairman will arrange for the conduct of all meetings and for such records or minutes of meetings as he considers necessary.

6. ADMINISTRATIVE AND OTHER SUPPORT

   a. The Board will operate from NASA Headquarters, Washington, DC. The Director of the Jet Propulsion Laboratory will arrange for providing office space and other facilities and services that may be requested by the Chairman or his designee. Support will also be provided by the Director of the Goddard Space Flight Center as requested.

   b. All elements of NASA will cooperate fully with the Board and provide any records, data, and other administrative or technical support and services that may be requested. The Associate Administrator for Space and Terrestrial Applications will provide such funds as are necessary for the conduct of the Board's business.

   c. A specific time or milestone shall be selected after which all activity physically affecting the hardware shall cease, unless specifically directed by the Board.

7. DURATION

   The Deputy Administrator will terminate the Board when it has fulfilled his requirements.

8. CANCELLATION

   This Notice is automatically cancelled one year from its effective date.

A. M. Lovelace

Deputy Administrator

DISTRIBUTION:
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Attachment:
A - Members and Affiliates of Seasat Mission Failure Investigation Board.

NASA

National Aeronautics and
Space Administration

Washington, D.C.
20546

Office of the Administrator

OCT 1? 1978

Dr. Bruce T. Lundin
5859 Columbia Road
North Olmsted, OH 44070

Dear Bruce:

I appreciate very much your willingness to accept the Chairmanship of the Seasat Mission Failure Investigation Board. As we discussed during our meetings last Tuesday, it is important that the Board move as expeditiously as possible, consistent with a thorough analysis of the facts, to determine the actual or probable causes of this mission failure. All necessary steps are being taken to ensure that appropriate Agency resources and all background information are made available to your Committee.

Enclosed is a copy of the NASA Notice formally establishing the Seasat Mission Failure Investigation Board which appoints you Chairman and names the other board members. I believe the charter of this group, as set forth in the Notice, is consistent with the discussions you and I have had. I will appreciate your keeping me directly informed of the Committee's progress and of any problems you may encounter in pursuing this assignment.

Both the Administrator and I appreciate your willingness to respond on very short notice and accept this important assignment without regard to any personal inconvenience it may cause you. Your continued dedication to excellence in both space and aeronautics constitutes a valuable asset to this Agency.

Sincerely,

A. M. Lovelace
Deputy Administrator

Enclosure
As Stated

BOARD MEMBERSHIP

Chairman

Bruce T. Lundin	Director, Lewis Research Center (Retired)

Members

Parker V. Counts	Assistant to the Director, Reliability and Quality Assurance Office, NASA Marshall Space Flight Center
James E. Hannigan	Assistant Chief for Systems, Flight Control Division, NASA Johnson Space Center
T. Bland Norris	Director, Astrophysics Division, Office of Space Science NASA Headquarters
Daniel J. Shramo	Director, Space Systems and Technology, NASA Lewis Research Center
James E. Stitt	Director for Electronics, NASA Langley Research Center

Counsel to the Board

Robert Kinberg	Office of General Counsel, NASA Headquarters

Executive Secretary

Dell P. Williams	Acting Director, Space Systems Technology Division, Office of Aeronautics and Space Technology, NASA Headquarters

USAF Liaison

Major James T. Mannen	Program Manager, Space and Missiles Systems Organization, U.S. Air Force

GSFC Liaison

Merland L. Moseson	Director of Flight Assurance, NASA Goddard Space Flight Center

JPL Liaison

Lt. General Charles H. Terhune	Deputy Director, Jet Propulsion Laboratory

Special Assistant to the Board Chairman

Vincent L. Johnson	Deputy Associate Administrator for Space Science, NASA Headquarters (Retired)

Technical Consultant

Arthur F. Obenschain	Head, Power System Design and Analysis Section, NASA Goddard Space Flight Center

APPENDIX B

SEASAT

SEQUENCE OF EVENTS – MISSION OPERATIONS ROOM

REVOLUTION 1503

APPENDIX B

SEASAT
SEQUENCE OF EVENTS – MISSION OPERATIONS ROOM
REVOLUTION 1503

GMT	ACTIVITY/ EVENT
23:30 (DOY282)	Team 3 on console (7:30 p.m. local time)
00:54 (DOY283)	Begin Rev 1502
01:32	Start Tape Recorder #2 Recording (SPC)
01:36	End of SAR pass at OAK HANGER (UKO)
01:37	Tape Recorder #1 Full (ready to dump)
02:18	**AOS ORRORAL (ORR)**
	All Systems Normal
02:20:00	Start Tape Recorder #1 Dump (RTC)
02:28:00	Complete Tape Recorder #1 Dump (RTC)
02:30:41	**LOS ORRORAL (ORR)**
02:35:13	Begin Revolution 1503
03:00:10	Start Tape Recorder #1 Recording (SPC)
03:05:30	Tape Recorder #2 Full (ready to dump)
03:06:00	Enter Earth Shadow
03:29:43	**AOS SANTIAGO (AGO)**
	First indication of satellite problem. Battery currents off-scale high (approximately 51 amps). Battery voltage down to approximately 24.7 V. –Y Solar Array Temperature off-scale high (approximately 869°F), +Y Solar Array Temperature off-set 40°F, unregulated bus voltage down to approximately 24.1 V. Other system parameters read normally. LMA informed ACMO of these abnormal indications. ACMO's RF and command system displays looked normal. All sensor data appeared normal. Vehicle attitudes looked normal. All on-board command sequencing was executing properly and solar arrays tracking normally. Team suspected data systems data processing quality. ACMO changed pass plan to forego ranging mode in order to allow commanding as required.
03:31:00	ACMO requested POCC computer reloading.

GMT	ACTIVITY/EVENT
03:31:37	Lost live data due to reloading.
03:32:06	Reloading complete, regained live data. No change in data indications.
03:33:00	Team still questioned data validity. ACMO requested Santiago STDN station to reload real-time telemetry decom program and computer telemetry processing program.
03:33:08	Lost live data due to reloads.
03:33:40	Station reloads complete, regained live data. No change in data indications. Still could see no sensor problems or obvious system problems from real-time telemetry.
	ACMO requested that ORRORAL be scheduled upon this revolution. ACMO discussed with LMA the possibility of a POCC computer convert table problem.
03:41:00	Exit Earth Shadow. Battery currents back on-scale, but high.
03:42:53	LOS SANTIAGO (AGO)
	No RTC's sent during pass. Problem was not understood, and no clear-cut corrective action was apparent. All on-board command sequencing still executing properly.
03:43 to 03:59	Team discussed data indications and possible problems, both on-board and on the ground. Could not identify any on-board system failure that could cause data indications. A "snapshot" of the POCC computer convert tables was made, since problems in this area were suspected, but insufficient time existed for in-depth analysis. Team still felt no clear-cut actions were to be taken. ACMO confirmed that ORRORAL would be up for this revolution at 03:59 GMT.
03:59:09	AOS ORRORAL (ORR)
	Previous indications of problem still existed. Battery voltages were down to 22 V. Unregulated bus current up to 34 amps. Otherwise, subsystems and sensors still appeared normal. Convert tables still suspect, therefore, ACMO requested the POCC computer be switched to backup disk pack to resolve processing problem.
04:03:08	Lost live data due to POCC computer disk switching. System would not run on backup disk, so ACMO requested return to original configuration.
04:06:58	POCC computer backup, regained live data. Data indications of problem still present. Team still suspecting data problem. ACMO did not have time to reconfigure station and command a tape recorder playback, although it was planned.

GMT	ACTIVITY/EVENT
04:08:28	LOS ORRORAL (ORR)
	No RTC's sent during pass. Problem was not understood, ACMO requested ORR analog tape of pass be replayed into POCC computer. No further acquisitions at any stations.
04:09	ACMO notified CMO of problem.
04:20	LMA notified LMSC team leader of problem.
04:15	Begin Rev 1504.
04:20:00	ORR Rev 1503 playback indicates data quality good, no problem with ground system, and that the satellite had suffered a true electrical problem.
04:55:00	Team declared spacecraft emergency. ACMO requested command sequences to remove sensor power from unregulated bus (Sequence 2) be transmitted in the blind (during predicted MERRITT ISLAND acquisition).
05:10:55	Last part of safing Sequence 2 transmitted in the blind (during predicted QUITO acquisition).
08:14:20	ACMO requested "no downlink" procedure be transmitted in the blind (during predicted GOLDSTONE acquisition).
	Safing Sequences 1 and 2, plus "No Downlink" procedure, transmitted in the blind repeatedly, from this point on.
03:30	ACMO requested ORR Revolution 1502 tape recorder dump "quicklook" data tape from IPD.
11:30	Team 2 on console (7:30 a.m. local time).
18:30	Team received request from LMSC to transmit RTC's which disconnect solar array panels 9 and 10 from diode bus, which was added to other uplink sequences transmitted in the blind.
20:30	Received ORR Rev 1502 "quicklook" data tape from IPD, and processed through POCC computer.

Mission support, with repeated acquisition attempts and safing sequence transmissions, continued until November 10, 1978.

APPENDIX C

ABBREVIATIONS AND ACRONYMS

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ABBREVIATIONS AND ACRONYMS

ACMO	Assistant Chief of Mission Operations
AFPRO	Air Force Plant Representative Office
AGO	Santiago Tracking Station
ALT	altimeter (Seasat sensor)
AMA	Assistant Monitor Analyst
AOS	acquisition of signal
APL	Applied Physics Laboratory
CCC	Charge Current Controller
CEI	Contract End Item
CMG	control moment gyro
CMO	Chief of Mission Operations
CPAF	cost-plus-award-fee
CRT	cathode ray tube
CSE	Chief Systems Engineer
CTU	command timing unit
DC	direct current
DOY	Day of the Year
EDT	Eastern Daylight Time
EM	Engineering Memorandum
EPS	electrical power subsystem
ESA	European Space Agency
FMECA	Failure Modes, Effects, and Criticality Analysis
FPI	fixed price incentive
GFE	Government furnished equipment
GMT	Greenwich Mean Time
GSFC	Goddard Space Flight Center
IPD	Information Processing Division (at GSFC)
JPL	Jet Propulsion Laboratory
LMA	Lead Monitor Analyst
LMSC	Lockheed Missiles and Space Company
LOS	loss of signal
M&O	Maintenance and Operations
MCT	Mission Control Team

MOR	Mission Operations Room
MPCDU	Main Power Control and Distribution Unit
NASA	National Aeronautics and Space Administration
NASCOM	NASA Communications
NCR	Nonconformance Report
ODT	Operational Demonstration Test
ORCH	Operations Requirements and Constraints Handbook
ORR	Orroral Tracking Station
ORT	Operational Readiness Test
PAD	Program Approval Document
P/N	part number
POCC	Project Operations Control Center
POS	Project Operations System
POST	POCC Operations Support Team
PSS	Portable Simulation System
RCS	reaction control subsystem
REE	Responsible Equipment Engineer
Rev	Revolution
RF	radio frequency
RFP	Request for Proposal
RFQ	Request for Quote
RPM	Revolutions Per Minute
RTC	Real Time Command
S/A	solar array
SADE	solar array drive electronics
SAM	solar array module
SAR	Synthetic Aperture Radar (Seasat sensor)
SASS	Seasat-A Scatterometer System (Seasat sensor)
SFOP	Space Flight Operations Plan
SMMR	Scanning Multichannel Microwave Radiometer (Seasat sensor)
S/N	serial number
SOC	state-of-charge
SPAT	Satellite Performance and Analysis Team
S/R	slip ring
STDN	Spaceflight Tracking and Data Network
TSU	telemetry and sensor interface unit
UKO	Oak Hanger Tracking Station
VDC	volts-DC
VGP	vehicle ground point
VIRR	Visual and Infrared Radiometer (Seasat sensor)
WFC	Wallops Flight Center
WTR	Western Test Range