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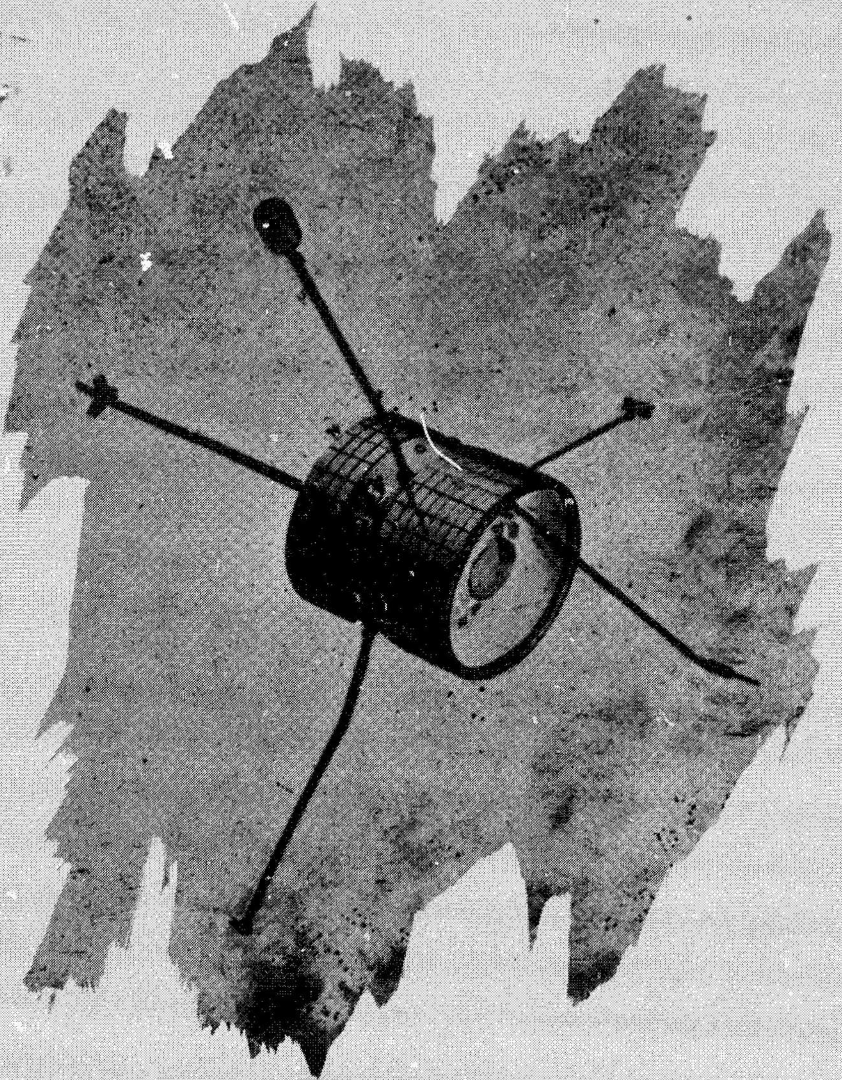

PIONEER VI ORIENTATION CONTROL SYSTEM DESIGN SURVEY

NASA/ERC
Design Criteria Program, Guidance and Control

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Electronics Research Center
Cambridge, Massachusetts

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TRW SYSTEMS GROUP

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1. SUMMARY AND CONCLUSIONS

The Pioneer VI orientation control system design survey covers only the current series of Pioneer Interplanetary Spacecraft (Pioneers VI through X). All of these spacecraft are essentially of the same configuration. To date, Pioneers VI through VIII have been launched. Pioneer IX is scheduled for a 1968 launch and Pioneer X for a 1969 launch. The generally successful and trouble-free design and development of the Pioneer VI orientation control system is credited to:

- A simple and basically sound design concept
- Use of proven hardware
- Efficient and technically competent management
- Early determination of most of the system requirements

The entire development was performed with no overruns in cost or schedule.

The control system equipment, described in Section 2, was relatively unsophisticated and required no advances in state-of-the-art technology. The functional performance of the orientation control system met or exceeded the specified requirements with a few minor exceptions.

Section 3 describes the historical evolution of the control system equipment from the first feasibility studies until the completion of the design phase of the program.

Despite the general success of the development, a number of serious problems and "near misses" did occur. Section 4 describes the problems which were encountered. Subsection 4.13 summarizes and categorizes the problems and lists some general techniques for preventing problems of the type which occurred.

The sun sensors were the most problematic assembly of the orientation control system. Sun sensor problems included fabrication and testing difficulties, thermal compensation problems, component procurement and reordering problems, and an unexpectedly short life in orbit due to extreme sensitivity to radiation. It is believed that the chief single cause of these problems was the use of a new and relatively unknown device for the sensor detectors. In 1963, the use of proven hardware for every component was a severe design constraint, and the use of some new devices was unavoidable. Nevertheless, the experience on Pioneer supports the well-known principle that the use of proven components reduces risks and problems.

Section 5 describes the innovations which were produced in developing the orientation control system. Due largely to the effort to use proven techniques, no significant advances in technology were produced.

The changes which would be made "if the development could be done over again" are described in Section 6. The chief changes which would be made are listed below:

1) Proven detectors would be used in the sun sensors.
2) A modified redundancy technique would be used in the electronics to permit checkout of all redundant components at the black-box level.

3) The sun sensor design would be modified to improve the ease of fabrication.

4) The sun sensor test equipment would be modified to improve the accuracy and repeatability of tests.

5) The pneumatic valve and regulator would be changed to a leakage-redundant type.

6) The pneumatic tube fittings and fill valve would be modified to improve the ease of fabrication.

7) AC input signals for the sun sensors would be generated by the electronics assembly rather than by the spacecraft equipment converter.

8) The sun sensor thermal analysis would be expedited.

9) Interface compatibility would be more carefully documented and tested.

10) A backup design for the sun sensors would be developed in parallel with the baseline design until the completion of detector evaluation testing.

11) NASA would be invited to participate more actively in some of the design tradeoffs.

The recommended changes would not have materially impacted the functional performance of the control system except to avoid the sun sensor radiation damage problem. The chief benefits of these changes would have been lower schedule and cost risks, greater design margins, fewer problems in fabrication, integration, and testing, and greater confidence in the flight performance of the system. Predicting the effect of these changes on the total program project cost, if they had been adopted, has not been attempted.

2. SYSTEM DESCRIPTION

2.1 GENERAL REQUIREMENTS AND GOALS

The Pioneer orientation control system (OCS) was specifically designed to meet the requirements of an interplanetary probe mission for a spacecraft launched into the ecliptic plane. The design was intended to optimize the achievement of the following spacecraft goals:

- Provide a communication capability of at least 50 million nmi
- Require minimum weight and power
- Provide lowest possible magnetic field
- Provide a thermal design capable of operating over a range of solar distances between 0.8 and 1.2 AU
- Use proven hardware
- Provide a probability not less than 0.8 of 6 months life in orbit
- Achieve injection from a Thor/Delta boost vehicle
- Meet a schedule of 21 months from go-ahead to first launch
- Minimize costs

The design chosen to meet these goals is illustrated in Figure 1. The spacecraft is spin oriented to maximize stability and control simplicity and to minimize thermal problems. The mast along the spin axis contains a high gain fan beam antenna producing a "pancake-shaped" radiation pattern normal to the spin axis and symmetrical about the mast. The function of the OCS is to align the spin axis normal to the sun line for maximum solar array power and normal to the plane of the ecliptic so that the major antenna beam lobe coincides with the ecliptic plane. The orientation is accomplished by releasing pulses of cold gas from a pneumatic nozzle as described in Subsection 1.2. The chief requirements identified for the OCS, in addition to the general spacecraft goals listed previously, are to:

1) Ultimately orient the spacecraft spin axis perpendicular to the sun-spacecraft line to within ±1.5 deg and perpendicular to the ecliptic plane to within ±1 deg.

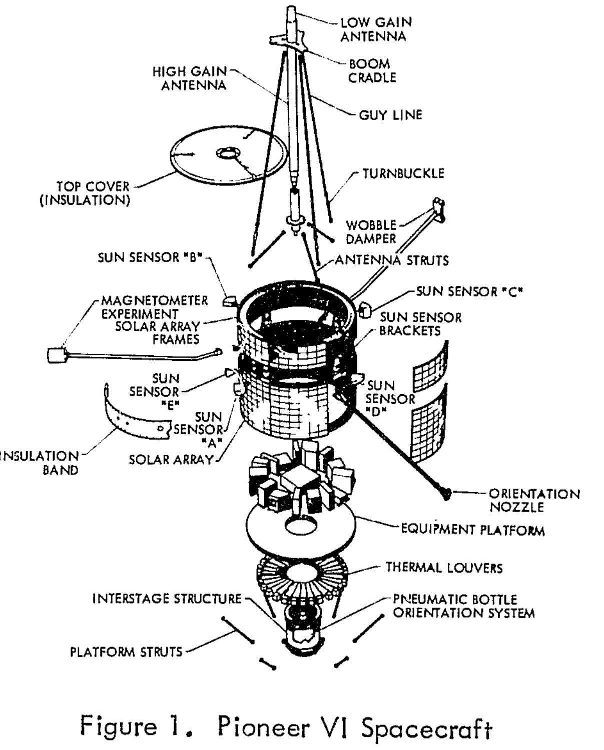

Figure 1. Pioneer VI Spacecraft

2) Orient the spacecraft perpendicular to the sun-spacecraft line as soon after launch as is possible.

3) Orient the spacecraft perpendicular to the ecliptic plane as soon as is possible after the near-earth geometry permits, but before the bit rate would be lowered below that specified due to the reduction of gain with distance of the omnidirectional low-gain antenna.

4) Provide information on the angles that have been traversed so that the pointing of the spacecraft may be reconstructed as a function of time of the mission and to hold the errors of this reconstruction to the minimum possible.

5) Provide enough pneumatic gas to perform all the necessary maneuvers adequately under the worst conditions of temperature, errors, leakage, etc., that may be reasonably expected.

6) Provide, through instrumentation, knowledge of the quantity of gas remaining.

7) Provide a system wherein any single electronic part failure will not prevent the successful performance of the requirements.

8) Provide a system that is not dependent on the timing of the receipt of commands such that the failure to receive a command at the proper time will not cause any unplanned firings or loss of gas.

9) Limit the wobble resulting from orientation maneuvers to less than 1 deg maximum.

10) Provide a sun reference pulse to the experiments once per spacecraft revolution.

2.2 ORIENTATION CONCEPT

The Pioneer OCS consists of five sun sensors, an electronics assembly, and a pneumatic assembly as shown schematically in Figure 2. Four of the sun

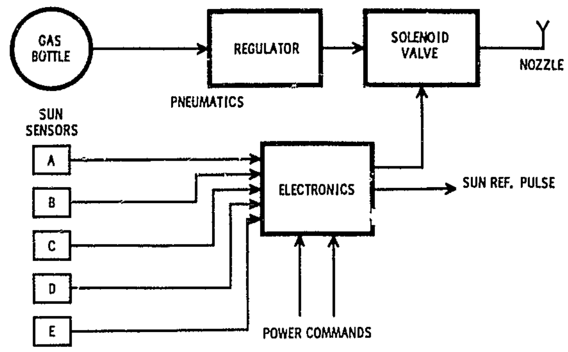

Figure 2. Orientation Control Subsystem Block Diagram

sensors, designated A, B, C, and D, are spaced at 90-deg intervals about the spacecraft and are used to control the timing and duration of gas firings which apply torque increments to the spacecraft. The fifth sun sensor, E, supplies a reference timing pulse once per revolution of the spacecraft for use in the control system logic and the experiments. The electronics assembly provides the logic and signal conditioning circuitry to process the sun sensor signals and drive the pneumatic valve. The pneumatics assembly stores gas under high pressure, regulates it to a lower pressure, and releases it through a solenoid valve and nozzle at the end of a boom in response to signals from the electronics.

An important design feature is the use of a single gas jet and solenoid valve controlled by several sun sensors. Each sun sensor is a simple, binary-output photodetector with a specific field-of-view. The proper sun sensor is enabled to fire the valve when the command status is correct, as determined by the time history of commands and sun sensor illuminations. The axis and direction of precession are determined by the orientation of the nozzle boom relative to the sun line at the effective center of the torque impulse. Sensors A and C are used to orient the spin axis normal to the sun line; sensors B and D are used to orient the spin axis normal to the ecliptic. Since the two pairs of sensors are at right angles about the spacecraft, there is ideally no cross coupling between the two orientation maneuvers.

2.3 ORIENTATION CONTROL EQUIPMENT

2.3.1 Sun Sensors

Each of the sun sensors has the function of sensing the presence of the sun whenever the sun is within the sensor field-of-view, and abruptly changing its electrical impedance between the illuminated and nonilluminated conditions. The sensing is performed by photosensitive SCR's (PSCR's) connected in a redundant quad for each sensor. The fields-of-view are established by aluminum shade structures in front of the detectors. Thermal compensation and light threshold adjustment are provided by a resistor and thermistor selected at assembly and connected in parallel between the gate and cathode of each PSCR. The sun sensors are mounted on brackets outside the spacecraft circumference with passive thermal control provided by deposited aluminum thermal coatings. The approximate locations and fields-of-view of the sun sensors are shown in Figure 3.

2.3.2 Pneumatics Assembly

The pneumatics assembly stores gas under high pressure, regulates the gas to a lower pressure, and provides a valve and nozzle for expulsion of gas in response to commands from the electronics assembly. The components of the pneumatics system are shown schematically in Figure 4.

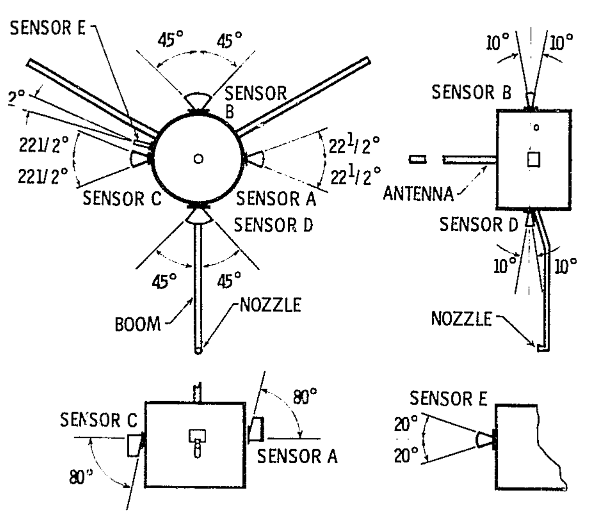

Figure 3. Sun Sensor Locations and Fields-of-View

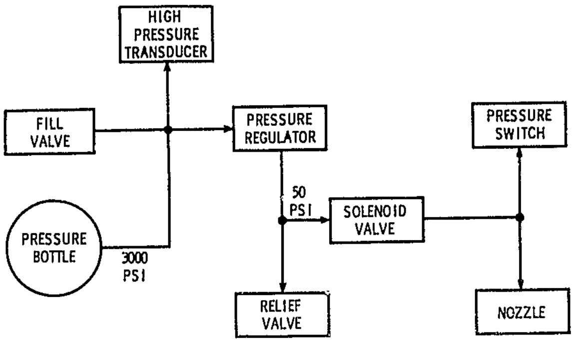

Figure 4. Pneumatics Assembly Block Diagram

The gas bottle is a pressure sphere containing nitrogen gas under 3,250 psi of pressure. This bottle is mounted on the central axis of the spacecraft near the center of gravity to minimize the effects of gas usage. A combination pressure regulator and relief valve mount directly to the bottle, regulating the pressure to 50 psi. A fill line is connected at the regulator inlet port, with a fill valve at the other end of the line near the spacecraft periphery. A high pressure transducer mounted on the fill valve provides an analog signal indicating the pressure and thus the quantity of gas remaining in the vessel.

The output of the pressure regulator is connected by aluminum tubing to the solenoid valve at the end of the orientation boom. Flexible teflon tubing is used in the vicinity of the boom hinge. The nozzle is located adjacent to the solenoid valve to minimize thrust buildup and decay times. A pressure switch is provided between the valve and nozzle to monitor gas firings. The pneumatic components are nonredundant. A complete description and analysis of the pneumatics assembly is given in Reference 1.

2.3.3 Electronics Assembly

The electronics assembly contains all the logic and amplification functions of the OCS. The circuits are digital in nature and are completely redundant so that no single part failure will affect the functional performance. The assembly consists of 17 modules of cordwood welded construction. No integrated circuits are used. A single connector is used for all connections which interface with the assembly.

The majority of circuits are made redundant by tripling the circuits and using voting logic to determine the majority decision. The remaining circuits, including the voting circuits, use quad redundancy applied at the part level.

Figure 5 shows a simplified block diagram of the electronics without redundancy; that is, this block diagram would apply if no redundancy were present. The sun sensors, which act as diodes when illuminated and as open circuits when dark, are included in the diagram because they are intimately associated with the logic. The blocks labled AC-1 and AC-2 represent windings on the spacecraft static converter which supply a half-wave rectified voltage to the orientation electronics. The illumination of sensors A, B, C, or D in conjunction with an appropriate ground command produces a rectified current to the valve driver circuit which causes a gas firing. A timing signal from sensor E to enable these firings is also used to ensure that a firing of proper duration takes place regardless of the timing of ground commands received. Thus, is the sun is in the center of sensor B's field-of-view when the enabling command is received, no firing takes place until the next revolution of the spacecraft when the sun enters sensor B's field-of-view. This feature assures the desired center of thrust as well as the correct gas firing duration.

The one-shot multivibrator (OSMV) indicated in Figure 5 operates in conjunction with sun sensor E to produce a reference pulse once per revolution to the experiments and telemetry unit.

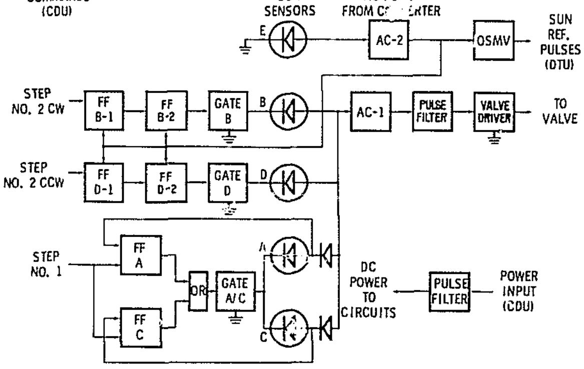

Figure 5. Electronics Assembly Block Diagram

2.4 ORIENTATION MANEUVERS

At the end of the third boost stage the spacecraft is spun up and separated from the booster. After boom deployment, which is initiated automatically upon separation, the spin rate is nominally 60 rpm.

The deployment of the booms initiates orientation step No. 1 by enabling sun sensors A and C. Sensor A looks up (toward the antenna end) and sensor C looks down, with their fields-of-view just meeting at the spacecraft equatorial plane. The elevation field-of-view of either sensor is 80 deg, so that the sun must not be within 10 deg of the spin axis in order for step No. 1 orientation to begin. This constraint is achieved by arranging the launch trajectory to be compatible with this requirement.

Whichever sensor (A or C) is illuminated by sunlight initiates gas firings which precess the spacecraft equatorial plane closer to the sun line in the shortest direction. This precession continues until the remaining sensor is illuminated, producing a pulse which is utilized by the electronics logic circuitry to turn off step No 1 orientation. Since some wobble will generally remain at this time, it may be necessary to repeat step No. 1 by ground command after the wobble has been damped out. The final accuracy of step No. 1 orientation is ±0.5 deg. The time required, not including damping time, is less than 15 min.

Step No. 2 is performed using the spacecraft low gain omnidirectional antenna for communications. The spacecraft should be at least 250,000 mi from the earth before the start of step No. 2 to provide stable geometry between the spacecraft orbital plane and the ground tracking station. At the completion of step No. 2, the low gain antenna is switched off; the high gain antenna is switched on; and the orientation control system is switched off except for the reference timing pulses from sen sensor E. The accuracy of step No. 2 is estimated to be ±1.0 deg. The time to perform step No. 2 is largely a function of command rate and is not critical.

2.5 SUMMARY OF OCS DESIGN SPECIFICATIONS

The following parameters summarize the chief specifications of the orientation control system:

Accuracy:	±0.5 deg relative to sun ±1.0 deg relative to earth
Orientation Capability:	225 deg minimum
Weight:	<6.5 lb including gas
Power:	0.6 w when valve not firing 6.3 w when valve is firing
Magnetic Field:	<0.1 gamma at magnetometer
Assessed Reliability:	>0.98 for 6 mo in orbit
Unique Features:	100 percent redundancy of all electronic and sun sensor parts

A more complete description of the Pioneer orientation control system is contained in Reference 2. Further details on the errors and tolerances of the control system are given in the general system error analysis of Reference 3.

3. HISTORICAL SUMMARY OF DESIGN DEVELOPMENT

3.1 EARLY HISTORY

The essential concept of the Pioneer orientation control system was first described by T. G. Windeknecht in 1961 (Reference 4). The system originated by Windeknecht utilized two gas jets, each controlled by a separate sun sensor, to orient the body-fixed solar panels of a spinning spacecraft perpendicular to the sun line. Solar panels were to be located at one end of the cylindrical spacecraft. Brief gas firing impulses, two per revolution, were to be used to precess the spacecraft spin axis to a direction parallel to the sun line. Only one axis of orientation control was considered in the system described by Windeknecht.

In early 1962, TRW proposed to NASA a spacecraft conceived to extend the capabilities of the earlier Pioneer series. The orientation control system proposed at that time was conceptually similar to Windeknecht's attitude control method. The TRW proposal resulted in a 2-1/2 month study contract (NAS2-884) to examine the feasibility of an interplanetary probe for the international quiet sun year. These studies produced a refined concept for the spacecraft and the orientation control system including the following features:

- Spin axis alignment perpendicular to the sun line, with solar cells on the outer surface of the cylinder.
- The capability to align a high gain antenna beam with the ecliptic plane by means of a second axis of attitude control.
- A single gas jet for all orientation pulses, with relatively long pulse durations (90 deg of spacecraft rotation per firing).
- A total of six sun sensors: four for controlling gas firings; one to indicate sun line perpendicularity to the spin axis; and one for timing reference pulses.
- All orientation steps to be initiated by ground command.

Following the 2-1/2 month study contract, an RFP was issued by NASA. TRW submitted a proposal on 4 March 1963 with relatively few significant changes in the OCS concept. The number of sun sensors was reduced from six to five in the proposed design; and electronic component redundancy was proposed in response to RFP requirements.

3.2 DESIGN CONCEPTS AT PROGRAM START

The conceptual design of the sun sensor assemblies did not change significantly between the contract go-ahead in September 1963 and the final design described in Paragraph 2.3.1. SCR detectors in a redundant quad were selected to perform the sensing function and view angles were established by simple sun shades. The details of level setting, thermal control and compensation, radiation protection, alignment control, etc., were not determined until later in the program.

The electronics assembly also changed very little between the initial concept and the final design described in Paragraph 2.3.3. The method of implementing redundancy was not firmly established at the program start; but the voting circuit approach was chosen soon after the beginning of work.

The pneumatics assembly concept at the program start utilized a pneumatic nozzle at the end of the antenna mast rather than on a boom. The solenoid valve was to be located inside the spacecraft compartment, a configuration which would have produced long rise and fall times for the pneumatic pulses. The reason for this design choice was an uncertainty concerning what type of booms, if any, would be included on the spacecraft. The TRW proposal called for short rigid booms; a later NASA directive eliminated the booms entirely; and a final directive following further studies called for the use of relatively long semiflexible booms. When the boom configuration was settled, the pneumatic valve and nozzle were relocated close together at the end of one of the booms (thereafter designated the orientation boom). The remainder of the pneumatics assembly remainded basically unchanged in concept throughout the program.

3.3 CHRONOLOGICAL DEVELOPMENT

The development of the Pioneer orientation control subsystem started officially on 21 September 1963. A preliminary system specification was written by 1 October 1963, and a PERT chart for control system tasks was prepared shortly afterward. Control system sizing studies and dynamic motion analyses were started at once. A conceptual design review of the OCS was conducted on 4 October 1963, with no significant changes recommended. The control system analysis continued, and an analog simulation of the orientation maneuvers was started on 2 January 1964. The simulation revealed a stability problem which is described in Subsection 4.1; a change in the orientation logic was made to eliminate this instability. The analog simulation was repeated with completely normal behavior. The simulation, which included the effects of flexible booms and a wobble damper, was completed on 28 January 1964. Detailed results of this study are reported in Reference 5. A final error analysis of the OCS was completed and published in April 1965 (Reference 3).

The overall time from go-ahead to the release of drawings for the OCS equipment was approximately 8 months, very close to the planned schedule. A major factor in maintaining this schedule is believed to be a relatively strict adherence to a detailed program plan.

The chronological development of the three assemblies which comprise the orientation control subsystem is described in Paragraphs 3.3.1 through 3.3.3.

3.3.1 Sun Sensor Chronological Development

Initial work on the sun sensor assemblies started a few weeks before the official go-ahead on 21 September 1963. A preliminary unit specification was produced by 1 October. A specification for the PSCR detectors was prepared and negotiated, and the engineering model detectors were ordered by mid-November. In the meantime development tests were performed using sample PSCR's. These tests revealed electrical and thermal problems which are described in Subsection 4.5. Corrective steps were taken; and the engineering model sun sensors were completed in early February 1964. Test equipment for the sun sensors was designed and fabricated by the time the sensor engineering models were available. The sun sensor engineering model tests were completed during the first week of March, with acceptable functional and environmental performance. The final design review was conducted on 20 March, and all of the drawings were released to manufacturing on 7 April 1964. Test procedures and final design specifications were released by May 1964.

The preliminary thermal design and analysis of the sun sensors was completed near the end of January 1964. A major error in the original thermal analysis was discovered in September 1964, changing the equilibrium temperatures of sensors A and C by more than 60°F. The final thermal data for both transient and steady-state sensor operation was not available until October 1964, 6 months after drawing release. One result of the transient analysis was a predicted 30-min delay in the start of step No. 1 orientation following a worst-case coast and eclipse of the spacecraft. The final results of the thermal analysis had relatively little impact upon the sensor design margins. Thermal vacuum tests in April 1965 supported the predicted thermal behavior of the sensors.

The first production model sun sensors were completed in January and February of 1965. Considerable difficulty in fabrication and testing was experienced as described in Subsections 4.6 and 4.12. Qualification tests were passed without incident. The first sun sensors were integrated in March 1965 revealing the electronics assembly interface problem described in Subsection 4.8. This problem was corrected by a minor change in the electronics assembly. The remainder of the sun sensor production, testing, integration, and flight performance was uneventful except for the Pioneer VII orientation anomaly discussed in Subsection 4.11.

3.3.2 Electronics Assembly Chronological Development

The development of the electronics assembly closely paralleled that of the sun sensors. Work officially commenced on 21 September 1963; a preliminary unit specification and a schematic block diagram were prepared within a few days' time. Magnetically acceptable components were chosen from the Pioneer approved parts list, and circuit design and breadboard testing was begun within 2 weeks of the program go-ahead.

The electronics assembly breadboards were completed during November 1963, and were modified and completely tested for functional performance and adequacy of redundancy by mid-December. A design review was conducted on 19 December 1963, but a partial redesign of the electronics to eliminate the control system stability problem (see Subsection 4.1) resulted in a repetition of the design review on 7 February 1964. Progress on the electronics assembly engineering model proceeded in the meantime. The model was completed on 12 February 1964 and was completely tested by early March. The test equipment was fabricated and tested concurrently with the engineering model fabrication and testing.

Amendment 10 to the Pioneer spacecraft contract required a conditioned sun reference pulse from sun sensor E; the signal conditioning circuitry necessitated the addition of another module to the electronics assembly. The changes to implement this requirement were carried out during March and early April of 1964, including changes to the test equipment consoles. A final design review of the electronics was held on 14 April, with design approval being granted. All drawings and test procedures were relased to manufacturing in early May 1964, and the test equipment consoles were transferred to manufacturing at that time.

Fabrication, testing, and integration of the flight model electronics assemblies were uneventful except for two problems which are described in Subsection 4.8. One of these problems necessitated the addition of a small circuit board to the electronics in April 1965. Subsequent to that change, no significant problems involving the electronics assembly have occurred.

3.3.3 Pneumatics Assembly Chronological Development

The development of the Pioneer pneumatics assembly was started in late August 1963. The design and development of the pneumatic ground support equipment was carried out in parallel with the development of the flight pneumatic equipment. Long lead items for the ground equipment were ordered by the end of September. By early October a preliminary unit specification and a preliminary solenoid valve specification were issued. Specifications for the major remaining pneumatic components (the pressure regulator and relief valve, the pressure switch, and the high pressure transducer) were completed in October 1963 and were released for bids the following month. A vibration test plan for the pneumatic components was completed around 1 November 1963; and vibration testing of most of the pneumatics components was performed during December.

Development tests were performed between November 1963 and February 1964 to measure solenoid valve delay, line pressure drops, shape and magnitude of pulse thrust, and residual magnetic fields. A reliability test plan to evaluate the effects of acceleration and long term vacuum on 20 solenoid valves and five pressure regulators was formulated during March and was revised in April and July of 1964. Reliability and life testing were begun with the receipt of the engineering model components in August 1964; the first phase of this testing was completed in September 1964; and the second phase was completed approximately 1 yr later.

The first pneumatic test cart was completed in March 1964; and fabrication of the second (deliverable) test unit was started. The second test cart was completed in May 1964 but was not delivered to integration testing until January 1965.

The pneumatics assembly specification and test procedures were issued in June and July 1964. Engineering model tests of the pneumatics assembly were performed from August through October 1964, with entirely acceptable results. A combined first and second pneumatics design review was held on 6 November 1964. Drawings for TRW-built components were released in June 1964, 5 months before the design review. No significant changes to the pneumatics assembly have been required during the remainder of Pioneer program activities up to the present time.

3.4 IMPACT OF CUSTOMER REDIRECTION

Three amendments to the Pioneer spacecraft affected the cost and schedule of the orientation control system development. The TRW proposal called for short, rigid booms for inertial control with the pneumatic valve and nozzle at the end of one of the booms. Amendment 1 deleted the booms from the spacecraft at the beginning of the program. The pneumatic nozzle was relocated on the antenna mast; and the solenoid valve was relocated near the center of the spacecraft. This configuration produced thrust rise and trailoff problems which were studied at considerable length. Amendment 3, issued in October 1963, reinstated booms in the design; but these booms were considerably longer and hence more flexible than those of the TRW proposal. The valve solenoid and nozzle were relocated on the end of a boom, solving the trailoff problem. The addition of flexible booms caused the analog simulation of the orientation control system to be more complicated than the simulation which was planned and costed in the proposal. The total cost increment of Amendments 1 and 3 was estimated to be approximately $23, 500 (including overhead and administrative costs) for the effort expended in iterating the pneumatic design and for the added complexity of the analog simulation. The schedule delay caused by these amendments was approximately 4 weeks. These cost and schedule changes are estimated in Reference 6.

Amendment 10 was received in February 1964, and it requested a conditioned sun reference pulse from the electronics. This change required the development of some new circuits, contruction of new breadboards, modification of the engineering model, engineering model retesting, documentation, drawing changes, and changes to the test equipment. The total unburdened cost increase was an estimated $5400, and the schedule delay for the electronics was an estimated 3 to 4 weeks (Reference 7). It should be noted that the relatively small magnitude of the costs and delays for this change were the result of the fortunate existence of some extra space in the electronics assembly which was not needed before the changes were made to implement Amendment 10.

4. DISCUSSION OF MAJOR PROBLEMS

The development of the Pioneer OCS equipment was generally straightforward and successful. All tasks were performed within the alloted cost and schedule; and all performance requirements were met or exceeded with one or two minor exceptions. A number of significant problems did occur, however, and are described in Subsections 4.1 thru 4.12. Most of the conclusions to be drawn from these problems are given in Subsection 4.13.

4.1 CONTROL SYSTEM STABILITY PROBLEM

The orientation control system as first conceived was found to be unstable. This instability, which was completely unexpected, was uncovered during an analog computer simulation of the step No. 1 orientation maneuvers. The simulation showed that a divergent unbounded wobble oscillation would occur when the spin axis approached the normal to the sun line.

At the time of the instability problem, sun sensors A and C were designed to have a small deadband between their fields-of-view. It was intended that whichever sensor viewed the sun would initiate gas pulses to precess the spin axis until the sun was in the deadband at the spacecraft equatorial plane. At this point neither sensor would see the sun and the gas firings would cease. The problem was due to the relatively large wobble amplitude introduced by each pulse. The wobble was not considered a likely problem source because analysis had shown that gas firings once per revolution would cancel wobble contributions rather than allowing wobble to accumulate. The analog simulation showed that wobble was well-controlled until the sun entered the deadband. At that point, however, firings would occur on every other cycle rather than once per cycle, with wobble adding in-phase. Eventually, the sun sensor on the other side of the deadband would also begin to view the sun on alternate revolutions, adding to the instability. The wobble would increase without bound until the gas supply was exhausted.

Once this problem was recognized, it was easily corrected by small changes in the sun sensor viewing angles and the electronics logic. The viewing angle deadband was reduced to zero and the control system logic was revised to terminate step No. 1 as soon as both sensors (A and C) have been illuminated by the sun.

The control stability problem was recognized early enough in the program that it could be corrected without significant schedule or cost impact. It is doubtful that the problem could have been avoided by any presently known design practices. The occurrence of the instability problem is now well known by TRW control system engineers and may help to avert such problems in future analogous situations.

4.2 MAGNETIC CLEANLINESS PROBLEMS

Despite the stringency of magnetic cleanliness requirements for Pioneer, only minor problems were encountered in designing the OCS to meet these requirements. The electronics assembly utilized standard components identified on the Pioneer parts approval program with no significant difficulty. The pneumatic assembly required a specially designed solenoid valve to minimize the residual magnetic field. This design proved to be relatively straightforward.

The sun sensor development was impacted by magnetic requirements more severely than the other two assemblies. The prime supplier of the sun sensor detector quads, Solid State Products, Inc., was unable to devise a hermetic detector window package which would meet the magnetic requirement. The material normally used for a window style detector header is Invar, a highly ferromagnetic alloy with a low expansion coefficient. Nonferromagnetic metals which could meet the magnetic requirements have relatively high expansion coefficients making a glass-to-metal seal difficult to achieve.

After some preliminary testing and analysis it was decided that nonhermetic detectors could probably be used, subject to additional test verification. As a backup, however, an alternate detector procurement was initiated. The alternate detectors were SCR's manufactured by Western Semiconductors, Inc., in a window package which exceeded the magnetic field allotment. When further tests established the acceptability of nonhermetic detectors, the procurement of an alternate device was abandoned. Western Semiconductors had encountered serious difficulty in manufacturing the alternate devices and was therefore entirely agreeable to a cancellation of the purchase order.

The failure to develop a sealed package for the sun sensor detectors caused the devices to be extremely vulnerable to damage during sun sensor assembly. The manufacturing problems which arose from this cause are discussed in Subsection 4.6. The costs incurred in the process of taking steps to meet the magnetic cleanliness goal are believed to be in excess of $10, 000.

4.3 WEIGHT PROBLEMS

The primary weight consideration in developing the Pioneer OCS was the strong weight incentive provided in the contract. Considerable ingenuity was used to save tenths or even hundredths of a pound wherever possible.

The electronics assembly utilized an innovation to provide RF shielding and minimum weight simultaneously. The modules at the top of the electronic package were plated with metal rather than using a cover on the assembly. The result was a thin RF-proof enclosure around the package that minimized size as well as weight.

Weight was strongly considered also in the choice of designs and materials for the pneumatic components. The pressure vessel was designed to have a shape and material which would minimize weight; this resulted in a spherical titanium alloy vessel. Weight was also a factor in the choice of designs for the pressure regulator, tubing, and the solenoid valve.

The sun sensor shade structures were designed to have brazed light-gauge aluminum construction. This manufacturing process proved to be quite expensive but the weight savings justified the expense. Another weight-saving feature of the sun sensors was the connector chosen. In order to perform complete functional and diagnostic testing of the redundant detectors, a nine-pin connector was desired for each sun sensor. Standard nine-pin connectors were both bulky and heavy in relation to the overall sensor weight of approximately 0.2 lb each. A new lightweight connector manufactured by Cannon had an all nylon construction except for the pins and sockets. After evaluation, this connector was added to the approved parts list and has been used with failure-free service throughout the program.

In summary, weight was not a problem for the Pioneer OCS in the sense that the basic requirement could have been met by straightforward design. Considerable attention was devoted to weight economy, however, becuase of the contract incentive, and a significant percentage of the small OCS weight was trimmed away by this attention to weight control.

4.4 PROBLEMS IN IMPLEMENTING REDUNDANCY

The implementation of redundancy in the Pioneer OCS was undoubtedly the chief source of design difficulties for the electronics and sun sensor assemblies. The pneumatics assembly was not impacted by redundancy since it was established early in the program that redundancy would not be required for pneumatic components.

It was anticipated at the start of the electronics assembly design that redundancy would greatly increase the size, weight, complexity, and power consumption of the unit. These problems proved to be somewhat more severe than expected, however, particularly in the areas of weight and power consumption. Other problem areas not anticipated at the design start were:

- Testing problems and limitations
- Increase in parts count
- Performance penalties

Circuit redundancy with voting logic was used in the timing and logic electronic modules. Component quad redundancy was used for the voting circuits and the valve driver circuits. This implementation permitted complete testing of all components at the module level, but did not permit the testing of redundant parts after final assembly of the unit. It was found that a general reduction of performance design margins was necessary to provide redundancy without using an excessive amount of power. The reduction of performance margins was a factor in the sun sensor interface problem discussed in Subsection 4. 8.

Difficulty was also encountered in the implementation of sun sensor redundancy, despite the fact that the PSCR's were chosen to facilitate redundant sensing. The major problem areas were:

- Providing the required field-of-view for four distinct detectors
- Negotiating a detector specification with a suitable packaging configuration
- Packaging problems for auxiliary components

The field-of-view tolerances ranged between 0. 5 and 2. 5 deg for the critical sensing angles of the sun sensors. Close spacing and alignment of the detectors and relatively large shade structures were required to meet the view angle tolerances. The large shades added significantly to the weight and fabrication cost of the sun sensors. The close spacing of detectors was achieved with some difficulty, adding considerably to the detector cost. The auxiliary resistors and thermistors used for level setting and thermal compensation were difficult to package in the small space available for sun sensor electronics; the need for redundancy aggravated fabrication problems and necessitated the use of a less-than-optimum thermal compensation circuit because additional components would not fit in the cavity.

Redundancy in the pneumatics assembly was considered at the beginning of the Pioneer program. Leakage of the solenoid valve was determined to be the most likely failure mode of the pneumatics. Valve redundancy was abandoned because it would cause increases in size, weight, power consumption, and magnetic field.

Several alternatives to the selected redundancy implementations are described in Subsections 3. 3 and 6. 1. The methods chosen were adequate to meet the system requirements although numerous difficulties stemmed from the use of redundancy. The actual reliability increase produced by the use of redundancy would be difficult to estimate for the following reasons:

1) The reduction of performance margins caused by the use of redundancy re not reflected in the assessed reliability.

2) The assessed reliability figures are, to some extent, mathematical fictions. A design for which no single part failure will cause a system failure partially avoids this fiction since a single component with an atypical failure probability is less significant.

The experience on the Pioneer control system equipment substantiates the well-known fact that redundancy imposes heavy penalties upon equipment cost and performance (i. e., weight, size, and power) in exchange for the reliability increase and should be used only when such a tradeoff is warranted.

4.5 PROBLEMS UNCOVERED IN DEVELOPMENT TESTS

Development tests of the pneumatics equipment revealed two problems worth noting. The first problem was a leakage rate for the solenoid valve which exceeded the leakage budget. The teflon valve seat was found to be inadequate for the required tight seal. This problem was solved by replacing the valve seat with a special type of rubber containing no volatile plasticizer. The same material has been used on subsequent space applications as a result of its successful use in the Pioneer program.

The second problem encountered in the pneumatics development tests was the formation of ice in the regulator that prevented the valve from seating properly during low-temperature operation. This problem was solved by using very dry nitrogen gas to ensure a dew point well below the lowest regulator temperature experienced during the throttling operation.

Sun sensor development tests revealed that the PSCR detectors had relatively high sensitivity to temperature, becoming less sensitive at low temperatures. Sample detectors with low intrinsic sensitivity were found to be totally inoperable at expected temperatures for 1.2 AU operation (approximately -50°F). Two corrective steps were taken to reduce this problem.

1) The subcontractor was directed to maximize the intrinsic detector sensitivity.

2) Temperature compensation was provided in the sun sensor design by incorporating a thermistor in the detector bias circuit.

These steps reduced the problem of temperature dependence to acceptable limits, although some problems remained and contributed to the high rejection rate of sun sensors during production tests.

Another sun sensor problem identified during development tests was an interaction between the light threshold of the PSCR's and the waveform of the voltage applied across the device. Ideally, the voltage at the PSCR gate terminal would be determined only by the photo current across the gate/cathode junction and the value of the resistor connected between the gate and cathode terminals. When a sharp voltage pulse is applied across the device, however, the capacitance between the anode and gate of the SCR conducts a voltage spike to the gate. The electrical excitation at the SCR gate adds to the photo excitation to trigger the SCR. This problem was controlled by adding an RC filter to the interfacing electronics to reduce the voltage rise rate of the applied square wave voltage and by specifying a maximum voltage rise rate in the sun sensor design specification.

No unique or noteworthy problems were uncovered in the development testing of the orientation control electronics, except as discussed in Subsection 4.8.

4.6 FABRICATION PROBLEMS

The sun sensor assemblies proved to be quite problematic in fabrication. The chief factors which contributed to this difficulty were:

1) The PSCR quads were supplied in a packaging configuration which afforded no protection for the detector faces and small delicate leads. Numerous devices were damaged before fabrication personnel developed suitable methods for handling the detectors during assembly and testing.

2) The parameter spread of the detectors supplied for the production phase of the program exceeded that of the prototype samples supplied earlier. This spread led to problems in achieving adequate temperature compensation. Detectors which failed to meet specification limits were not rejected because detectors were in short supply throughout the program. A prepotting functional test was added to the fabrication procedure to

identify problems at a stage of production where corrective action was possible. This procedure was quite effective in reducing the incidence of failure in final test.

3) The small cavity for sun sensor electronics made the wiring and connector installation difficult to perform.

4) The exteriors of the sun sensors were coated with deposited aluminum for thermal control and gloves were required for handling the sensors during tests.

5) Electrical pickup in the sun sensor test cable was found to be the cause of some errors in light threshold measurements on the first group of sensors tested. This problem was reduced to an acceptable level by fabricating a new test cable with shielded wires.

6) The repeatability of sensor threshold measurements was somewhat poorer than expected. The main cause is believed to be the sun gun lamps used in the test. When first switched on, the lamp brightness was not stable; if left on for a longer time, the lamp heating would affect the sensor temperature, especially during functional tests at -40°F. Corrective action was not taken because the magnitude of the test errors appeared to be within acceptable limits.

The assembly and testing of the sun sensors would have been considerably more efficient if a single crew had been able to build all of the sensors in one production run. Unfortunately, schedule constraints dictated that only the units for a single launch be manufactured at one time. Considerable time was expended in reassembling the test equipment for each manufacturing run, and in retraining a new crew of assembly and test personnel.

Minor leakage problems were encountered in manufacturing the pneumatic fill valves since the valve seal required stringent fabrication tolerances. No corrective action was deemed necessary. The fill valve design, which was modeled after a similar valve in OGO, has now been redesigned for future applications to avoid the problem. Minor problems also occurred in manufacturing the flared aluminum tubing of the pneumatics assembly. Once again, no corrective action was required. Tubing with brazed fittings rather than flared tubing is now normally used for this type of application.

4.7 PROBLEMS DURING QUALIFICATION AND ACCEPTANCE TESTS

No problems were uncovered during the qualification or acceptance testing of the OCS equipment.

4.8 PROBLEMS IN INTEGRATED SYSTEM TESTS

The major problem discovered in integrated system tests was an incompatibility in the electrical interface between the sun sensors and the orientation control electronics. The sun sensors were assumed to switch abruptly when illuminated, so that the voltage across the sensor would drop sharply. The voltage drop was to be differentiated by a capacitor producing a negative pulse to terminate step No. 1 orientation. The voltage supply was designed with very high impedance (300 K ohms) to minimize power consumption. When a sun sensor was slowly illuminated, it was found that the leakage current through the PSCR's increased sharply just before switching. This leakage was sufficient to drop the supply voltage to a low value and the actual switching did not produce a pulse adequate to actuate the logic circuits. In actual use the sun sensor would change from zero illumination to full illumination in less than 2 msec. For such operation, the performance of the sensors and electronics would probably have been normal. In test, the illumination was introduced much more slowly so that the problem was emphasized.

The compatibility problem discussed above was discovered relatively late in the program. For this reason it was decided not to change the existing design of the electronics; instead, a small circuit board was added to the electronics assembly. The circuit board contained a capacitor to maintain the voltage across the sun sensors for brief periods of high leakage and a series resistor to prevent excessive capacitive discharge current through the sensors after switching. These additional components were quad redundant to comply with the redundancy specification. The adequacy of this fix was checked extensively in ground tests and has functioned normally in all Pioneer flights.

Another serious problem occurred during the thermal vacuum testing of the flight two Pioneer spacecraft. On the third day of the test, when conditions for 1.2 AU operation were simulated, the orientation control system failed to operate normally. The dominant characteristics of the failure were:

- Once the gas valve had been turned on by an illumination of sensor A or C during a step No. 1 maneuver, the valve would remain on until power was removed from the orientation control electronics.

- The control system did not respond to step No. 2 commands in either the clockwise or counterclockwise direction.

During the course of diagnostic testing to isolate the cause, the problem went away and could not be made to reappear. Subsequent testing and data analysis could only isolate the problem to the spacecraft equipment converter, the control electronics assembly, or the sun sensors. A number of steps were taken to correct this problem:

1) The equipment in question was replaced.

2) A 10,000-ohm resistor was placed across the new equipment converter output terminals to reduce noise in the sun sensor excitation voltage.

3) The method of illuminating sun sensors in the test chamber was modified.

4) The removed equipment was extensively retested. No further evidence of malfunction was uncovered, however, in this retesting.

The abnormal behavior discussed above has never been repeated either in test or in flight. A more complete description of this problem is given in Reference 8.

Testing the integrated control system for pneumatic leakage presented a problem in instrumentation. The specified limit for leakage was too low to be measured by the storage vessel pressure transducer during a short-term test. The method first tried consisted of measuring the differential pressure between the pneumatic storage vessel and a reference vessel while monitoring the temperature of the gas in each vessel. This setup proved to be a better thermometer than a leakage tester because the effects of gas temperature uncertainties swamped out the desired test data. An alternate technique used with some success consisted of measuring the leakage of argon gas from the pneumatics by means of a mass spectrometer. This method had some problems with gas circulation and sensitivity to the skill of the test conductor. The method presently in use is to make long-term (30 days or longer) measurements of the storage vessel pressure and temperature, with a sufficient number of tests to average out random errors. Leakage rates below 2 cc/hr can be measured by this technique.

4.9 PROBLEMS RESULTING FROM QUALITY ASSURANCE REQUIREMENTS

The quality assurance (QA) requirements on the Pioneer program were quite stringent. The procurement of components was performed in accordance with the guidelines of NPC 200-2. It was first thought that the combination of magnetic requirements and QA requirements might alarm prospective vendors and cause factor-of-two increases in the cost of components. The fears proved to be an exaggeration of the actual problem. Some vendors who had not supplied high reliability parts previously showed considerable alarm initially at the requirements for traceability, process control, and test record maintenance. Once the requirements were thoroughly understood, however, it was found that the cost increment for QA documentation was on the order of 10 to 15 percent. The quality requirements did increase schedule problems, however, due to increases in the time for negotiating purchase specifications and increases in delivery schedules.

4.10 INTERFACE PROBLEMS

Two major interface problems occurred in the design of the Pioneer OCS equipment. The first was the compatibility problem in the electrical interface between the sun sensors and the electronics, described in Subsection 4.8. The second was a mechanical interface problem between the spacecraft structure and the pneumatic pressure regulator. Both the structure and the regulator had vibration resonances near 500 cps. The regulator resonance was discovered during engineering model tests of the pneumatics; the damage threshold was determined to be between 20 and 30 g's. A potential problem was recognized at this time; but it was ultimately decided that test data indicated a small but positive design margin. No damage was experienced during the spacecraft qualification tests. On the first Pioneer flight, however, a high pressure leak occurred after launch. The most probable source of leakage was the main seat of the regulator. Fortunately, the leakage rate was small and decreased as the vessel pressure became lower. After completion of the orientation maneuvers and 6 mo time in orbit, the pneumatic system still contained a pressure of 150 psi. On subsequent flights the pressure vessel cradle design was modified to provide an isolation mounting for the pressure vessel and regulator.

4.11 PROBLEMS IN FLIGHT OPERATIONS

The first Pioneer flight (Pioneer VI) revealed the mechanical interface problem discussed in Subsection 4.10 above. No other problems were identified until Pioneer VII was launched, oriented, and in orbit for approximately 6 mo. At this time a step No. 1 orientation command was sent for the purpose of determining the spin axis orientation error after one-half orbital revolution. When no gas firings occurred, a pair of step No. 2 commands in each direction were sent. These commands also failed to produce the normal response. Subsequent to this failure, orientation commands were sent to Pioneer VI (which was out of gas by that time). Some of the sun sensors produced intermittent signals when enabled, while others produced no response. Pioneer VII was retested after approximately a 3-month time. At this time some pulses were generated, but it appeared that the sun sensors were operating marginally.

Sufficient data was obtained in the tests of Pioneer VI and VII to indicate with high probability that sun sensor degradation was the cause of the orientation anomalies. Radiation damage appeared to be an obvious explanation but was initially viewed with skepticism for the following reasons:

1) The sun sensors had 20 mils of protective radiation resistant glass over the detectors as compared to only 8 mils of glass over the solar cells (which had shown negligible degradation).

2) TRW radiation effects experts doubted that any silicon junction semiconductor device would be more sensitive to radiation than a solar cell.

The latter premise was examined by subjecting a few sample PSCR's to 1 MEV electron bombardment. It was discovered that the PSCR's were incapacitated by radiation flux levels several orders of magnitude below the damage level for silicon solar cells. After additional tests to recheck the initial calibration and threshold adjustment of the sun sensors, it was concluded that radiation damage was almost certainly the cause of the orientation anomalies which occurred on Pioneers VI and VII.

The first corrective step taken after the recognition of the sun sensor radiation susceptibility was to replace the protective glass covers on the Pioneer C sun sensors with much thicker (100 mil) covers. Pioneer C was la nched, thereby becoming Pioneer VIII, and successfully achieved initial orientation as did Pioneers VI and VII. The adequacy of the design change will not be known from flight data until early 1969. In the meanwhile, a program is being carried out at TRW to find means for improving the sun sensor design and extending the range of solar distances for sun sensor operation.

4.12 MISCELLANEOUS DESIGN PROBLEMS

A few notable problems were encountered in the sun sensor design which have not been covered in the preceding sections. These problems are outlined below:

1) The thermal design and analysis of the sun sensors was completed quite late in the design evolution. First estimates of sensor equilibrium temperatures proved to be grossly in error. Thermal behavior during launch transient periods was not available until after the end of the sensor design phase. Fortunately, the thermal behavior indicated by the final analysis was compatible with the sensor operating characteristics.

2) Earthshine at the time of spacecraft injection into orbit placed a serious constraint upon the lower sensing threshold of sun sensors A and C. This problem was reduced by narrowing the view angles of the sensors so that less diffuse light could be incident on the detectors. The only penalty for this change was an unimportant increase in the time required to complete step No. 1 orientation.

3) Testing the view angles and light sensing thresholds of the sun sensors appeared to require a well-collimated high-intensity light source which was not available among existing capital equipment in the fabrication area. This problem was avoided by using a low-level collimated light source to test the view angles and by sensing the detector responses by means of analog photovoltaic signals at the PSCR gates. The sensor switching thresholds were measured separately using gun lamps with no provisions for collimation control.

4) Solid State Products, Inc., the only source for the PSCR's, was unable to furnish additional units after the first purchase order had been filled. The unexpectedly high attrition rate of these detectors in manufacturing resulted in a shortage of flight-quality parts. The vendor claims to have lost the capability to produce more of these parts. Parts from test lots which would normally have been scrapped were ultimately used on Pioneer VIII. A design change to utilize a different detector will almost certainly be required for followon Pioneer spacecraft.

5) The design of sun sensor test equipment was complicated by the fact that some of the equipment was to be supplied by the engineering division and some equipment (a test bench, rotary table, optical rail, and a thermal control chamber) was to be supplied by the manufacturing division. This arrangement caused problems in designing test fixtures and preparing test procedures.

4.13 PROBLEM SUMMARY

The problems outlined in Subsections 4.1 thru 4.12 may be categorized into three groups:

1) Problems which could not have been foreseen or avoided by reasonable precautions

2) Problems which could have been avoided by major changes in equipment design concepts

3) Problems which could have been avoided or significantly reduced without changes in the design concept.

The problems which fall into each category are listed and discussed in Paragraphs 4.13.1, 4.13.2, and 4.13.3 which follow.

4.13.1 Unavoidable Problems

The following problems appear, in retrospect to be have been unavoidable and/or were identified and corrected as well as may reasonably be expected:

1) The spacecraft control system stability problem

2) Minor magnetic cleanliness problems in the design of the pneumatics assembly

3) Problems in meeting the electronics assembly weight and power budget, caused chiefly by the requirement for redundancy

4) Problems in achieving the desired sun sensor view angle accuracy with multiple redundant detectors

5) Minor problems in the pneumatics assembly development tests

6) Problems in obtaining hermetically sealed detectors for the sun sensors.

4.13.2 Problems Avoidable by Design Concept Changes

The following problems could reasonably have been anticipated, and could have been avoided by changing the design concept of one or more of the assemblies:

1) Problems in developing sun sensors with PSCR detectors. Problems similar to those which occurred should be expected when utilizing a new device of a rare or exotic type which has not been thoroughly tested and is not a "standard line" item for several vendors. Development problems, procurement problems, cost and schedule problems, reordering problems, and unique problems (such as the PSCR radiation sensitivity) are high-risk factors in a design based upon an unfamiliar nonstandard component.

2) Limitations to the testing of the electronics assembly. A somewhat different design concept would have permitted testing of redundant parts at the black-box level rather than the module level.

Recommendations for design concept changes which would have avoided some of the problems described above are presented in Section 6. The alternate concepts would undoubtedly have encountered some problems, but it is believed that the alternates would have been preferable design approaches.

4.13.3 Problems Avoidable Without Design Concept Changes

Problems which could have been reduced or avoided without conceptual changes in the design of the OCS equipment are summarized below, with brief outlines of the means by which improvements could have been achieved.

1) Sun sensor fabrication problems could have been significantly reduced by small design changes to improve producibility. Detectors with protective packages should have been specified; and the sensor electronics cavity should have been enlarged.

2) The interface problem of the sun sensors and electronics could have been discovered before drawing release by testing their compatibility during development tests.

3) The sun sensor radiation damage problem could have been avoided by radiation tests to examine the assumption that the PSCR's have a radiation tolerance similar to solar cells. Also, a thicker cover glass could have been used initially to increase protection since this apparent overdesign would have imposed no penalties.

4) The regulator valve damage on the first launch could have been avoided by (a) the use of a more realistic vibration qualification test level since the flight levels apparently exceeded the qualification test levels furnished by NASA;

and (b) a more conservative design of the pneumatic vessel and regulator support structure in view of the known valve resonance.

5) The anomalous behavior of the orientation control system in the thermal vacuum test described in Subsection 4.8 could probably have been avoided by generating the sun sensor excitation voltage in the electronics assembly rather than using the equipment converter output to drive the sun sensors, or by specifying the converter wave shapes and noise limits more carefully. This interface was not adequately controlled.

6) The sun sensor thermal interface was not pursued in a sufficiently aggressive manner. Although no actual problem occurred, the sensors should have been designed to have more effective temperature compensation; and the thermal requirements should have been more carefully specified to the spacecraft heat transfer department which was performing the thermal design and analysis.

7) Fabrication testing problems for the sun sensors could have been reduced by greater emphasis upon test equipment design.

8) The shortage of PSCR spares could have been avoided by a more conservative allowance for attrition. It must be acknowleged, however, that the actual attrition rate could scarcely have been anticipated. Also, it appeared at the time that the availability of such devices would improve with advancing semiconductor technology, although the opposite actually occurred.

9) Problems in designing sun sensor test equipment and preparing test procedures could have been reduced by more careful planning of the test phase of the program.

4.13.4 General Techniques for Problem Reduction

The avoidable problems illustrate the need for careful attention to a number of topics. Several general techniques for problem elimination which could have reduced problems in the Pioneer OCS development, if properly implemented, are listed below:

1) Use known and proven components whenever possible.

2) Avoid the use of single-source parts, especially if (a) the supplier is a small company which might go out of business, or (b) if the part is a nonstandard item which may be out of production when reordering is necessary.

3) Make conservative spare parts allowances for items which may be difficult to reorder.

4) Take extreme precautions in controlling electrical, mechanical, thermal, and other interfaces.

5) Carefully document the control of interfaces.

6) Insist upon quantitative test data to support engineering judgment on critical desicions (as in the case of the sun sensor radiation damage problem).

7) Place heavy emphasis upon the producibility and human engineering of designs.

8) Use integrated equipment tests rather than separate assembly tests whenever possible to verify interface compatibility and end-to-end performance.

9) Be certain that the test equipment design is adequate to ensure accurate, repeatable results, and be extremely careful in simulating interfaces for the equipment under test.

10) Maintain complete records of development test data in an engineering notebook.

11) Document all analyses, data inputs, recommendations, and other significant understandings which impact the course of design evolution to prevent misunderstandings, clarify responsibilities, and facilitate review of the design.

12) Emphasize test planning, and thoroughly check and validate test equipment and procedures before the beginning of production tests.

Although most of the items above are familiar cliches, the problems encountered in the Pioneer program could have been lessened by a more effective application of these concepts.

5. NEW CONCEPTS AND HARDWARE

One of the goals of the Pioneer program was to use proven hardware. It was generally found that the program requirements could be met without developing new equipment. Consequently, the innovations in the OCS equipment were relatively few.

5.1 OCS CONCEPT INNOVATIONS

The orientation concept used on Pioneer, with four sun sensors and one gas jet to provide two-axis attitude control for a spinning spacecraft, was a new and important advance in attitude control system technology. Although the Pioneer OCS was based in part upon the ideas developed by Windeknecht (Reference 4), important improvements were conceived and developed by TRW as described in Subsection 3.1. The Pioneer spacecraft was the first space vehicle to put such a concept into practice. The implementation which was developed contained minor innovations in the logic system to permit nonsynchronous commands to be used without risk of incomplete gas firings, and to terminate step No. 1 orientation accurately and automatically.

5.2 NEW COMPONENTS

The redundant quads of PSCR's used in the sun sensors were developed especially for the Pioneer program. The application of a PSCR as the sensing element of a digital sun sensor was first conceived and reduced to practice on Pioneer. The knowledge gained from this application was used advantageously on the Vela III attitude reorientation system and on the sun interference sensor for the Reliable Earth Sensor program at TRW. In each of these programs, a PSCR sun sensor was successfully developed with unprecedented speed and economy as a result of the technology acquired on the Pioneer program. Vendor contacts, part specification knowledge, test data, thermal compensation techniques, and methods for testing were all clearly understood after the development of the Pioneer sun sensors. Thus, at least one Air Force program and one NASA program (other than Pioneer) have benefited from the sun sensor experience on Pioneer at the time of this writing.

The nonvolatile rubber valve seat developed for the Pioneer pneumatics assembly was another new component. The low-leakage properties of this valve seat material for space applications have been utilized on several subsequent programs at TRW.

5.3 RELIABILITY FEATURES

The type of redundancy used in the Pioneer OCS was generally not new or unique. The choice of PSCR sun sensor detectors was a rather original idea which greatly simplified the implementation of sun sensor redundancy. This choice made it unnecessary to use any active components other than the detectors themselves to produce a quad redundant sensor output. Another novel feature of the sun sensors was a fail-safe temperature compensation circuit which caused the sensors to become undersensitive at temperatures either above or below the useable operating range. This feature was intended to prevent false-triggering from the earth's albedo after various coast and eclipse conditions which could produce extreme thermal transients.

5.4 PACKAGING INNOVATIONS

The orientation electronics assembly had a unique packaging design described in Subsection 2.3. The electronics package allowed half of the modules to extend above the top of the metal case; these modules were covered only with thin metallic plating to seal and shield the electronics.

5.5 TEST EQUIPMENT INNOVATIONS

There were few, if any, innovations in the design of the test equipment for the Pioneer OCS. The electronics test console was straightforward in design. The pneumatic test console was modeled after that of OGO with no significant nuances. A low-intensity solar simulator (collimated light source) was designed and built for testing the sun sensors; this equipment is noteworthy only because it was duplicated for testing the Vela III sun sensors, avoiding the need for developing a new design and a set of drawings for the latter program. A two-axis test fixture was also developed for testing the sun sensors. This fixture consisted of two single-axis rotary tables held at right angles to each other by a machined support. The horizontal table was a standard relatively heavy model. The vertical table was a very small light-weight model, only 2 in. in thickness. During test, the sun sensors were mounted to the smaller table in such a manner that the rotational axes of both tables intersected at the center of the sensor detectors. The test fixture provided a means for testing the sensor field-of-view without translating the detectors through the collimated light beam, an advantage over a conventional two-axis rotary table, and was fabricated for a fraction of the cost of available two-axis tables.

6. IMPROVEMENTS BASED ON RETROSPECTION

6.1 EQUIPMENT DESIGN MODIFICATIONS

If the development of the Pioneer orientation control system were to be repeated, the following equipment changes would be made:

1) The sun sensors would not utilize PSCR detectors, since PSCR's have been found to be very sensitive to radiation and are difficult to procure. Miniature solar cells would be used instead. A possible technique which could be used for the implementation of redundancy is illustrated in Figure 6. A more straightforward redundancy mechanization would be to provide a separate preamplifier and threshold circuit for each of three solar cell detectors and use majority voting logic to provide the output signal. Before the days of integrated circuits, the latter method was unattractive because a large number of components were required. With today's technology, however, this implementation would probably be preferred.

2) The sun sensor test equipment would be redesigned to have a more stable light source with a shutter system to prevent heating of the test article. Better electrical shielding would be provided in the test panel and test cable to reduce electrical pickup.

3) More volume would be provided in the sun sensor electronics cavity to facilitate fabrication.

4) The electronics assembly would be changed to utilize a modified redundancy technique to permit testing of all components after final assembly. The recommended redundancy implementation is illustrated in Figure 7. In the original design, voting was performed after each circuit function (such as a gate or flip-flop). The alternative shown in Figure 7 would defer voting until all logic and timing functions were performed by the three independent logic and timing units.

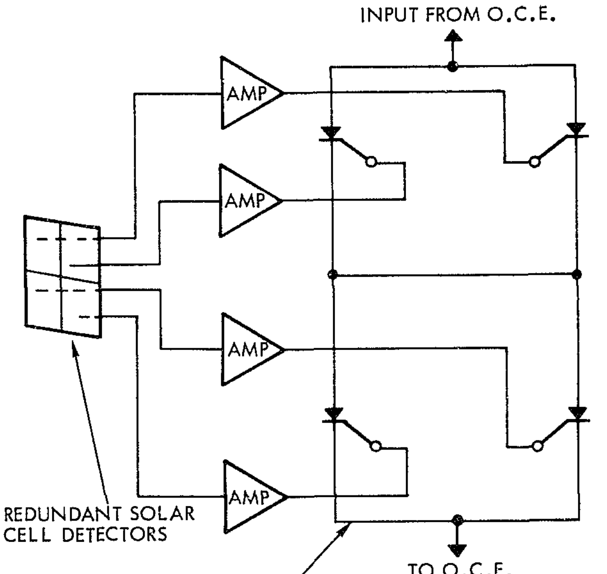

Figure 6. Simple Configuration for Redundant Sun Sensor Using Solar Cell Detectors

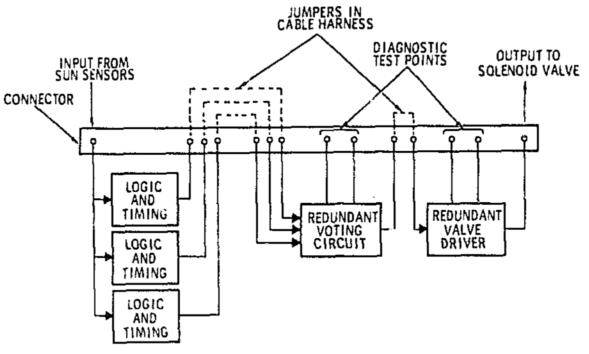

Figure 7. Alternate Redundancy Technique for Orientation Control Electronics

Jumpers in the cable harness would connect the voting circuit to the other electronic units during normal operation. During test, the connector would provide access to a sufficient number of test points to permit complete checkout of the assembly. (Note: Switching by ground command between two nonredundant systems was rejected because step No. 1 orientation must be performed automatically without ground control.)

5) The electronics assembly would be modified to supply all necessary signals to the sun sensors rather than using the spacecraft equipment converter for that purpose.

6) If PSCR's were to be used in the sun sensors (although this is not recommended), the electronics assembly would be modified to allow for higher leakage currents when the detectors are partially illuminated.

7) The electronics assembly test consoles would be changed to provide a more exact simulation of the sun sensors.

8) In the pneumatics assembly, leakage redundant valves would be used for the solenoid and pressure regulator valves. Since the design of leakage redundant valves was not developed at the beginning of the Pioneer program, it is understandable that such valves were not considered. However, these components have now been developed and could be used, with minor modifications, on Pioneer.

9) Brazed fittings would be used on the pneumatic tubing in most places to facilitate fabrication and reduce the risk of developing leaks during launch.

10) The pneumatic fill valve would be modified to reduce problems.

11) If the design were to be repeated with present day technology, integrated circuits would be used extensively in the electronics.

The desired effect of the suggested changes would be to improve the reliability, design margins, ease of fabrication, and ease and accuracy of testing for the OCS equipment. The functional performance of the control system would be essentially unchanged.

6.2 DESIGN APPROACH MODIFICATIONS

The following changes in general design approach would probably have reduced the problems and/or near misses encountered in developing the OCS equipment.

1) It is felt that a backup design for the sun sensors should have been partially developed to guard against unfavorable results in development testing of the PSCR's. A concept decision tradeoff should have been performed, with NASA participation, to choose between the two design alternatives. This dual approach would have cost more money, but it would have reduced risks substantially.

2) The control of interfaces should have been more completely documented.

3) Development test data, particularly for the sun sensors, should have been recorded and documented more fully.

4) More effort should have been exerted to complete the sun sensor thermal analysis before the criti cal design review of the sensors.

6.3 TEST PROGRAM IMPROVEMENTS

It is felt that two improvements in the test program could have been made. First, a more detailed development test plan could have been prepared and followed, with test reports being issued to indicate the achievement of test plan milestones. Second, tests should have been performed at the development testing stage and the engineering model testing stage to examine end-to-end functional performance and compatibility of the sensors, orientation electronics, and the pneumatic assembly.

7. REFERENCES

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