

NASA SP-57

FACILITY FORM 602

N66 29482

(ACCESSION NUMBER)

(THRU)

308

(PAGES)

1

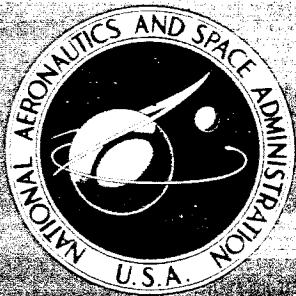
(CODE)

31

(CATEGORY)

(NASA CR OR TMX OR AD NUMBER)

ORBITING SOLAR OBSERVATORY SATELLITE



OSO

GPO PRICE \$ 3.00

CFSTI PRICE(S) \$ _____

Hard copy (HC) _____

Microfiche (MF) 1.75

ff 653 July 65

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

ORBITING SOLAR OBSERVATORY SATELLITE OSO I

The Project Summary

Prepared by
Goddard Space Flight Center,
Greenbelt, Maryland



Scientific and Technical Information Division 1965
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Washington, D.C.

For Sale by the Superintendent of Documents, U.S. Government Printing Office, Washington, D.C. 20402 • Price \$2

FOREWORD

This document describes the work performed in connection with the Orbiting Solar Observatory and the scientific data and results obtained therefrom. Supplementary information has also been added to provide the reader with a background in solar-oriented equipment.

Basically, this project was an effort to develop a solar observatory capable of operation in space for approximately six months. Ball Brothers Research Corporation conducted a research and development program to resolve methods for enabling the solar observatory to perform as required. Acting as subcontractors, Electronic Specialties Company designed the antennas, AVCO designed and manufactured the receiver and decoders, while Vector Manufacturing designed and manufactured the transmitter and diplexer. All of these items were prepared in accordance with BBRC specifications.

The Orbiting Solar Observatory, designated OSO I, was launched March 7, 1962 from AFMTC at Cape Canaveral, Florida. The successful observatory flight provided a wealth of scientific data concerning both the sun and the portion of space through which the observatory orbits.

The NASA Project Manager was Dr. J. Lindsay. The BBRC project directors were Mr. F. P. Dolder for spacecraft systems and Mr. R. H. Gablehouse for communications systems.

CONTENTS

FOREWORD	Page iii
<i>Chapter 1</i>	
INTRODUCTION AND PROGRAM DESCRIPTION	
Introduction	1
General Description of the Orbiting Solar Observatory	1
Description of the Observatory Experiments	7
Launch Vehicle and Orbit	9
Payload Envelope and Weight	9
Design of the Observatory	10
Development of the Observatory	10
<i>Chapter 2</i>	
SPACECRAFT DYNAMICS	
Analysis of Spacecraft Rigid-Body System	13
Nutation Damper Development	22
Effects of External Torques on Spacecraft	47
Control of Torque Effects	67
<i>Chapter 3</i>	
STRUCTURAL DESIGN AND FABRICATION	
Design Criteria	69
Wheel Structure	70
Upper Structure	76
Component Mounting	82
Arms and Arm Dampers	84
<i>Chapter 4</i>	
CONTROL SYSTEMS	
Introduction	89
Electrical Control Systems	91
Gas Control Systems	141
Performance	151
<i>Chapter 5</i>	
DATA ACQUISITION AND COMMAND SYSTEM	
Introduction	161
Systems Design	161
Subassembly Design and Qualification	169
Operations	181
Data Reduction	184
Overall Performance	185

CONTENTS

Chapter 6

POWER SUPPLY

	Page
Introduction	189
Power and Temperature Estimates	189
Solar Cell Array	189
Battery	190
Power Margin	190
Power Supply Performance	191

Chapter 7

THERMAL CONTROL

Introduction	193
Analysis of Simplified Mathematical Model	193
Analysis of Shielded Model of Spacecraft Wheel	206
Evaluation of the Shape Factor Integrals	211

Chapter 8

ORBITING SOLAR OBSERVATORY SCIENTIFIC EXPERIMENTS

General Description	219
Solar-Oriented Experiments	219
Wheel Experiments	228

Chapter 9

TEST PROGRAM

Introduction	243
Test Equipment	243
Developmental Testing	247
Prototype Testing	252
Flight Model Testing	267
Prelaunch Testing	273
Miscellaneous Tests	275

APPENDIX A—ANTENNA PATTERNS	279
--	-----

APPENDIX B—PASSIVE TEMPERATURE CONTROL	300
---	-----

Chapter 1

INTRODUCTION AND PROGRAM DESCRIPTION

This chapter presents a description of the Orbiting Solar Observatory program, a brief description of the scientific experiments aboard, orbit criteria, spacecraft design history, and the development sequence of the spacecraft.

INTRODUCTION

It has become evident that solar radiation and particle emissions, which have a great effect upon our atmosphere, must be measured *before* their character is changed by the atmosphere. Until recently, many solar emissions could be studied only through the effects they have on the atmosphere. However, to study both cause and effect, it is essential that these phenomena be measured both from above and below the atmosphere.

The studies decided upon for the first attempt at continuous measurement of solar phenomena above the atmosphere were spectrometric studies in the ultraviolet and in the short wavelength x-ray regions of the solar electromagnetic radiation spectrum. The value of the data gathered from above the atmosphere is two-fold. In the first place, measurements derived directly from the sun with no atmospheric attenuation may be used to arrive at explanations for purely solar phenomena and give greater insight into the mechanics of the production of the measured emissions in the sun. Secondly, the direct measurements from the sun when correlated with observed effects produced in the earth's atmosphere may be used to explain the mechanisms by which these atmospheric phenomena occur.

There can be no doubt that the ability to measure unattenuated solar emissions is a giant stride in the advancement of knowledge of the sun and its influence on the earth. The Orbiting Solar Observatory gave us this ability.

GENERAL DESCRIPTION OF THE ORBITING SOLAR OBSERVATORY

The Orbiting Solar Observatory (Figure 1-1) was primarily a stabilized platform for solar-oriented scientific instruments. In addition, experiments which did not require solar orientation were contained in the observatory.

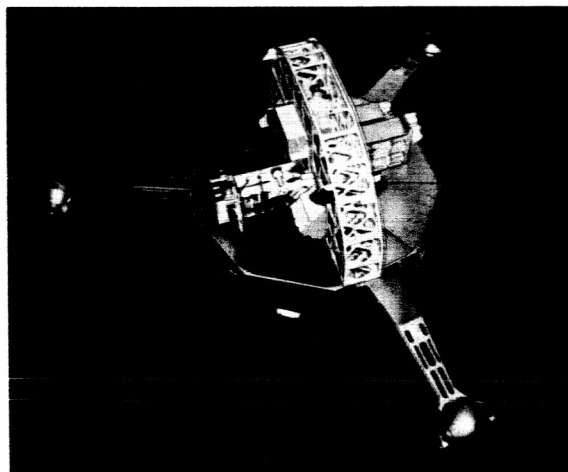


FIGURE 1-1.—Orbiting Solar Observatory (OSO I).

Electrical power required for observatory operation was supplied by an array of solar cells mounted on the upper section. A complete telemetry and command system was provided to transmit information back to earth, and to receive commands from a ground station.

Basic Spacecraft

The observatory consisted of two main sections: the wheel structure and the upper structure. The lower wheel structure was composed of nine wedge-shaped compartments. Five of these compartments contained scientific

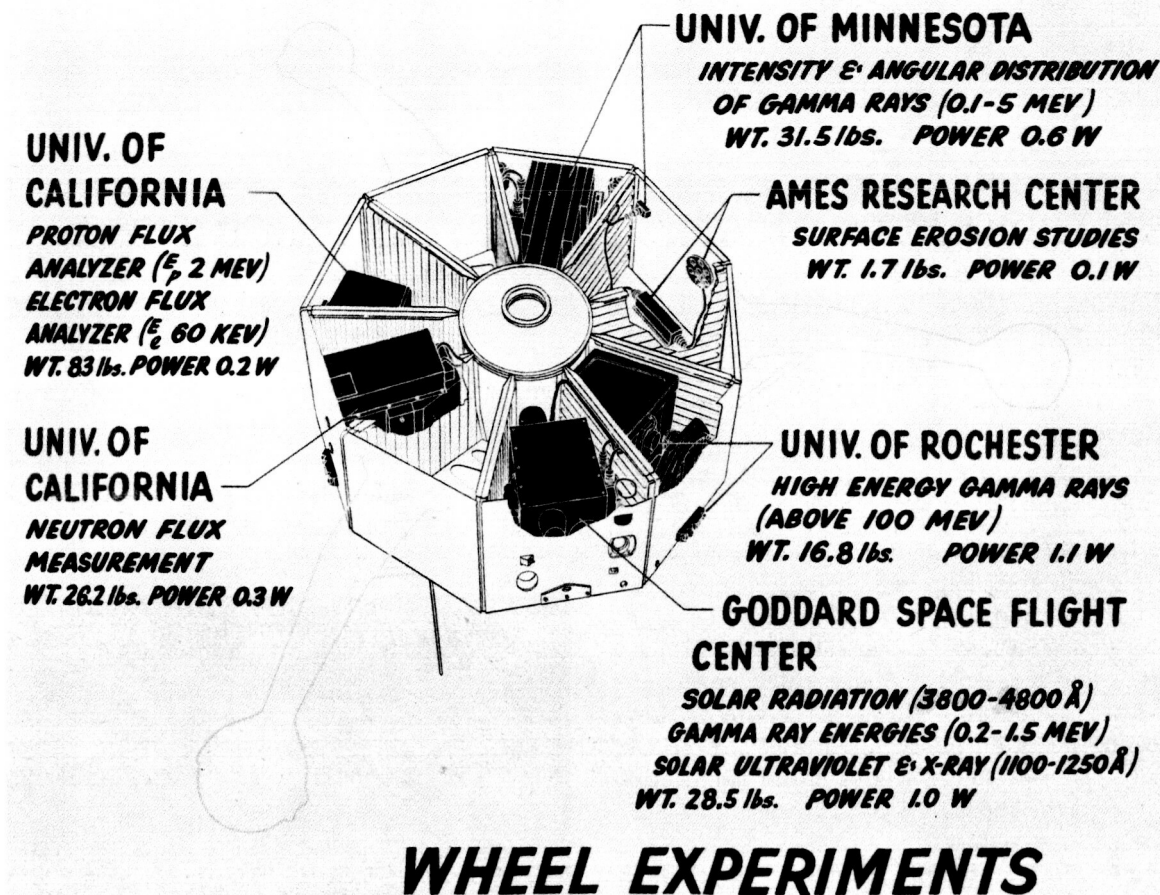


FIGURE 1-2.—Wheel experiments.

experiments (Figure 1-2); the other four contained functional hardware such as the wheel electronic controls, batteries, telemetry system, radio command system, and the in-flight data storage system (Figure 1-3). Three fiberglass spheres on extended arms contained pressurized nitrogen gas for the spin control system.

The upper structure of the observatory contained the solar oriented experiments (Figure 1-4), and was mounted on the wheel. The fan-shaped array to which silicon solar cells were attached made up the larger part of the upper structure.

The two main structures were connected by an aluminum shaft (Figure 1-5) extending from the base of the casting containing the pointed instruments, through the center of the wheel, and terminating in the support ring structure on the underside of the wheel. This shaft was

held in position by two bearings, one at the top of the wheel and one at the bottom. Mounted on the shaft between the bearings was a high-pressure nitrogen gas tank which carried the gas supply for precession jets. These jets were mounted atop the solar array structure. A torque motor mounted on the base of the shaft drove the shaft and upper structure relative to the wheel. This motor actively controlled the azimuth orientation of the stabilized upper structure by driving it at an equal rate but opposite rotational direction with respect to the wheel. Also mounted on the base of the shaft was a slip-ring assembly that transferred power, telemetry signals, and control signals from the upper section into the wheel.

The observatory was designed for maximum utilization of the payload section of the Thor-Delta launch vehicle. The maximum wheel

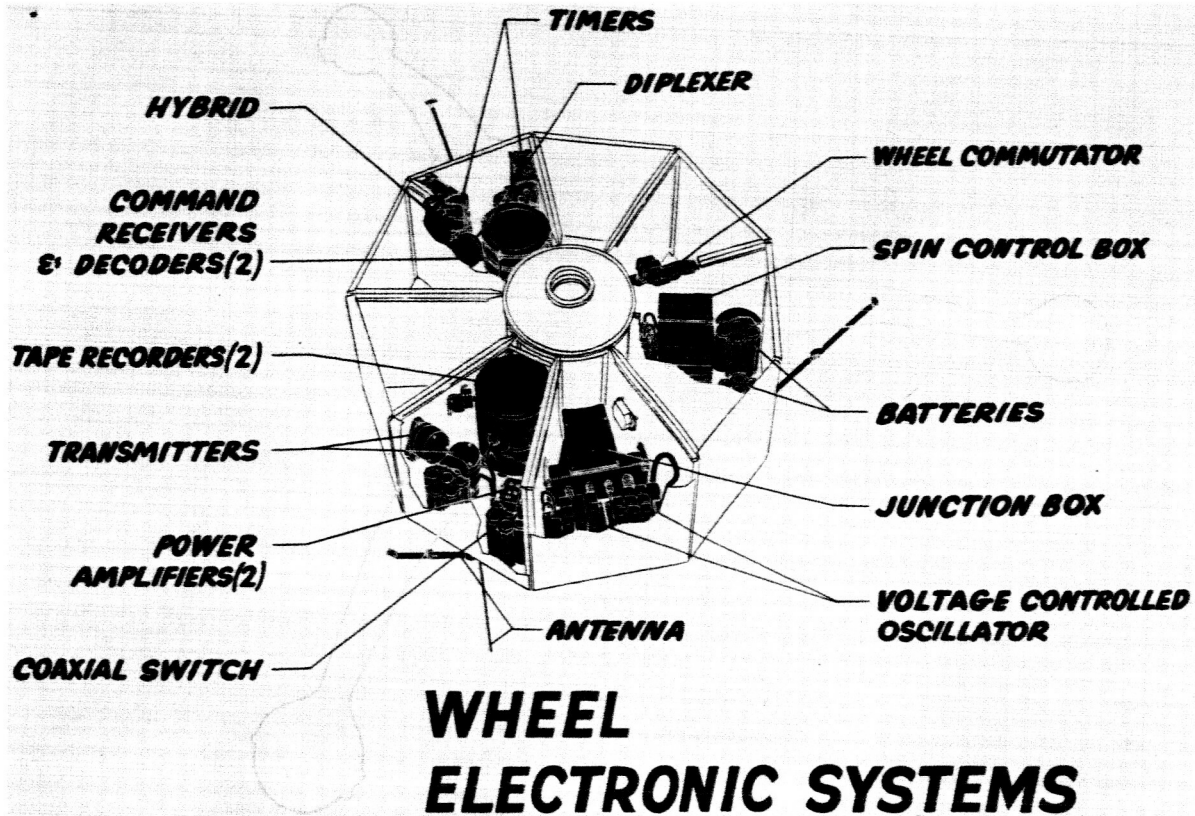


FIGURE 1-3.—Wheel electronics systems.

diameter allowed by the Thor-Delta shroud was 44 inches. During launch, the three nitrogen gas containers on the extended arms were folded down along the sides of the third-stage motor. When the nitrogen tanks were extended outward after third-stage burnout, the diameter of the payload was increased to 92 inches. Since these spheres were carried at the ends of the extended arms, the axial moment of inertia was larger than either transverse moment of inertia. The overall height of the observatory was 37 inches, and total weight was about 460 pounds.

The Orbiting Solar Observatory utilized the gyroscopic properties of a spinning body for stability. Prior to third-stage firing, the third stage and the spacecraft were spun up to approximately 100 rpm by a system of small rocket motors (Figure 1-6). After third-stage burnout, the arms supporting the three gas containers were extended. The spacecraft was separated from the rocket, and the spin rate

of the wheel was reduced to approximately 30 rpm by jet action. This spin rate was maintained within ± 5 percent of nominal value by gas jets attached to each of the spherical gas containers. These jets were actuated by a signal from photodetectors and an electronic control system which computed the instantaneous period of rotation of the wheel with reference to the sun. The three gas-filled spheres were interconnected to assure that unbalance of the rotating body would not occur due to an unequal gas flow rate through the jets.

Control System

The biaxial pointing control system of the observatory utilized the entire vehicle as the controlled platform. Coarse elevation positioning of the stabilized section was accomplished by determining pitch error with pitch control photodetectors and controlling pitch attitude with on-off jets (Figure 1-7). This was pos-

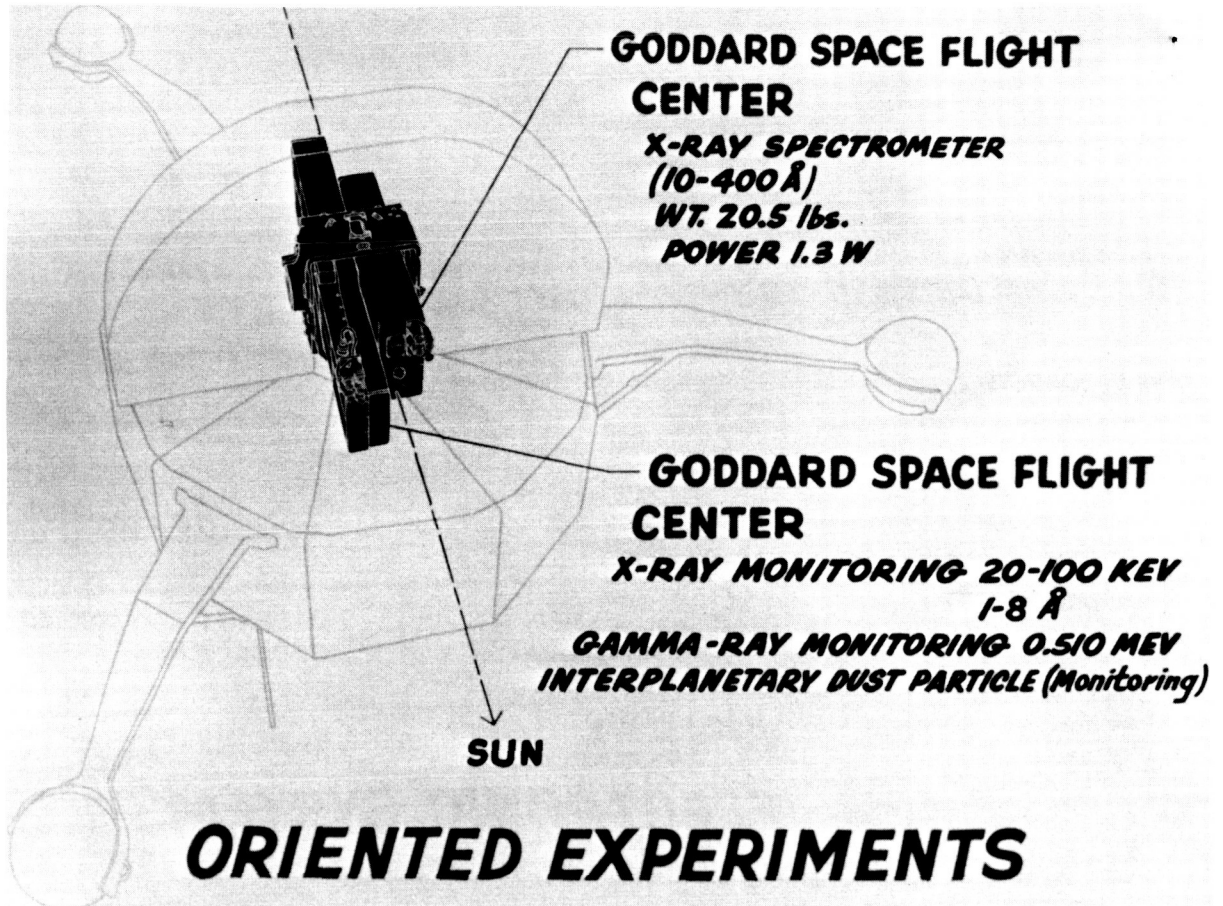


FIGURE 1-4.—Solar oriented experiments.

sible because of the gyroscopic properties of the spinning body. By exhausting nitrogen gas (after stabilization) through nozzles

mounted on the stabilized section, torques of either sense, normal to the plane of the solar array, could be produced to precess the spacecraft in elevation. The spin axis was thus positioned perpendicular to the solar vector within about three degrees. There was no control about the sun vector (roll), but rates around this axis were very slow due to the rigidity of the gyro.

Fine elevation and azimuth positioning of the instruments was accomplished by electrical servo motor control. As described earlier, the elevation servo motor was mounted on the casting which supported the pointed instruments. This motor drove the instruments relative to the spin axis for fine positioning in elevation. The azimuth servo motor, mounted to the shaft connecting the stabilized upper section to the spinning wheel, is mentioned in an earlier paragraph.

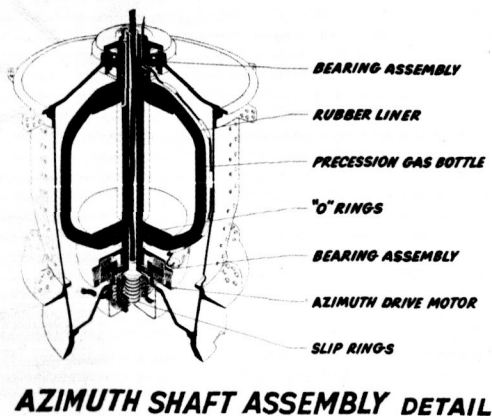


FIGURE 1-5.—Azimuth shaft assembly.

OSO I FLIGHT SEQUENCE

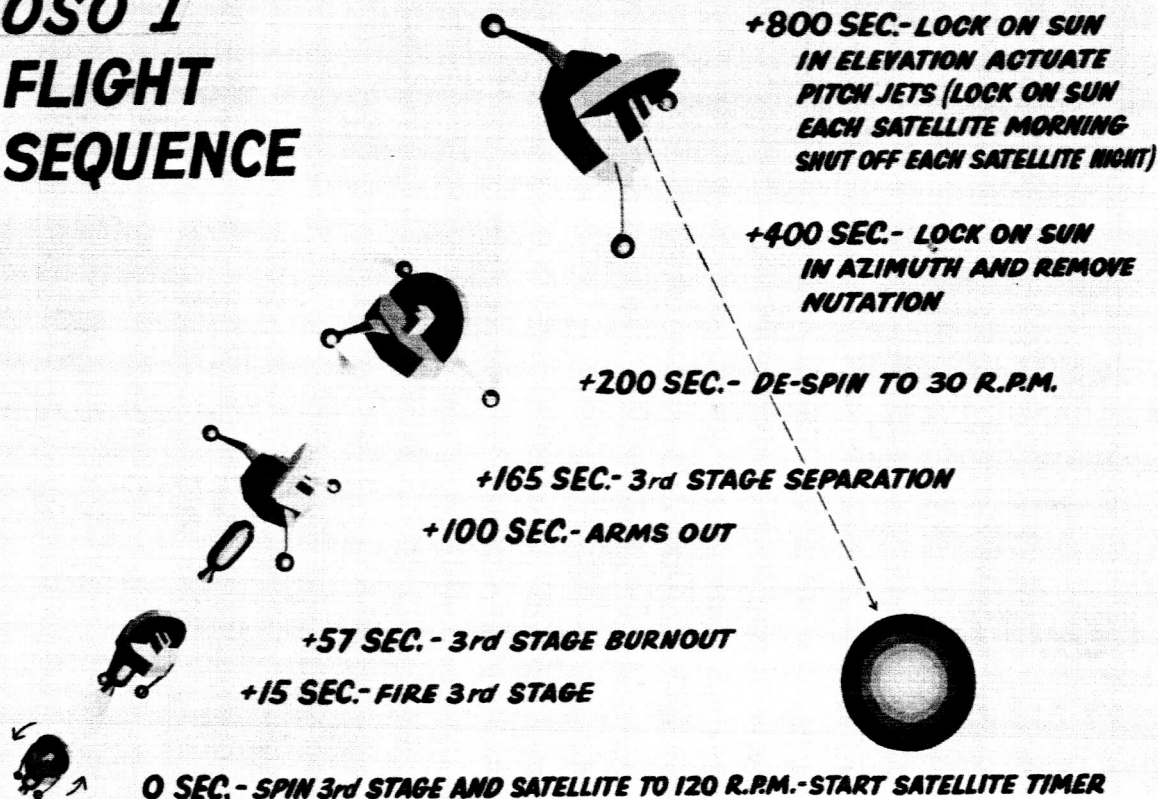


FIGURE 1-6.—OSO I flight sequence.

The azimuth and elevation servo motors were actuated by signals from two types of photo-detectors mounted on the stabilized upper section of the spacecraft. These were: (1) coarse detectors mounted on the sail structure (Figure 1-7) and (2) fine detectors mounted directly to the pointed instruments.

There were four coarse detectors, each having a 90-degree field of view, which provided a full 360 degrees of position control. There were two fine detectors which provided a differential signal over about 10 azimuth degrees in each direction from the solar vector. The elevation servo position error signals originated from the 10-degree fine detectors only.

When the spacecraft was in sunlight and the upper section was spinning, the azimuth

coarse detectors provided a signal to the azimuth servo drive system which would despin the upper section. As the upper section spin rate was reduced to zero, the coarse detectors pointed the solar instruments to within two to three degrees of the solar vector. At that time, a disabling detector mounted on the pointed instruments actuated a relay which turned the coarse detector control off, leaving the azimuth positioning servo with fine detector control only. When the azimuth servo had positioned the upper section normal to the solar vector, the elevation servo then positioned the pointed instruments in elevation. As explained earlier, the fine detectors for both elevation and azimuth position control were mounted directly to the pointed instruments.

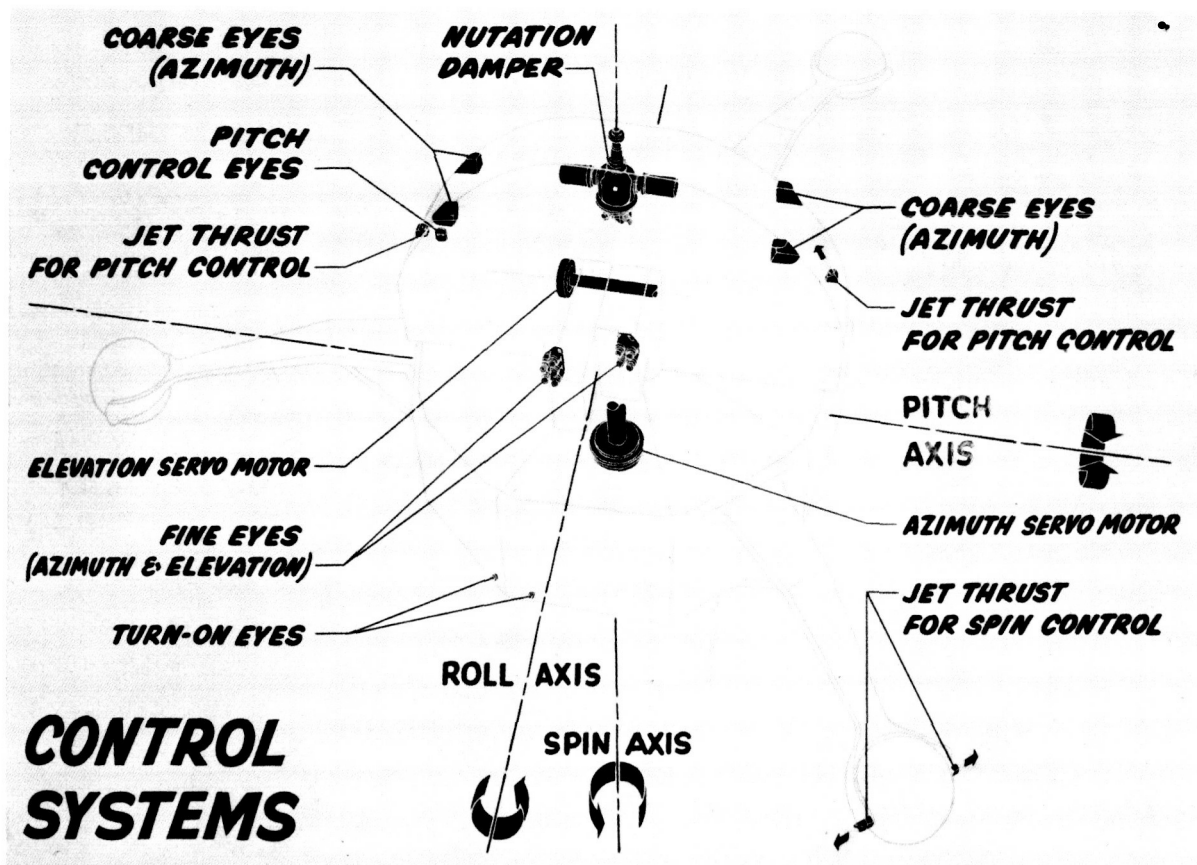


FIGURE 1-7.—Control systems.

Two of these detectors were used for each control loop. Operating as pairs, the detectors provided a differential signal which was the error signal for the particular servo. Short-term pointing accuracy of the fine control system was better than one minute of arc in elevation and azimuth. Long-term accuracy was better than two minutes of arc.

The pitch control detectors were detection units for the pitch jet control system. This block of four detectors was mounted facing the sun on the front of the stabilized structure. Whenever the spin axis of the spacecraft changed pitch more than three degrees from the normal to the solar vector, one of these detectors turned on the appropriate jet to precess the spacecraft back toward the desired attitude. Two of the detectors were required for this function, one of either sense. The other two detectors turned off the jets whenever

the spacecraft attitude had been returned to within one degree of the desired position.

Other photodetectors used in the pointing control system were: (1) turn-on detectors and (2) spin control detectors. The turn-on detectors, as the name implies, were used to actuate electrical equipment (turned off in the dark) each time the satellite emerged from the earth's shadow. A set of six detectors was distributed around the outer surface of the spinning wheel to turn on equipment both in the wheel and on the pointed section. The detectors observed the sun once per revolution of the wheel, providing the spin controller with the necessary information for the determination of wheel spin rate.

Power Supply

Solar energy was the sole source of power for operating electrical equipment and experiments

on the spacecraft. The solar-cell array mounted on the stabilized section converted solar energy into electrical energy whenever the spacecraft was in sunlight. This energy was stored in nickel-cadmium batteries contained in the spinning wheel. During daylight, when the spacecraft was pointing, the solar-cell array was normal to the sun vector within three degrees.

The total surface area of the solar-cell array was 3.72 square feet. This array was composed of thirty-one, 18-volt, 60-cell modules which produced a power output of approximately 27 watts. The average day-night power available for use from storage batteries was approximately 16 watts. Since the telemetry, data system, and control system required approximately seven watts, nine watts were available for the various scientific experiments.

Thermal Control

Spacecraft temperature was controlled by making use of carefully selected external surfaces. Special paints were developed for the upper structure back surfaces and the wheel rim surfaces. Polished aluminum covers were used for the flat top and bottom wheel compartment covers.

Data Acquisition and Command System

The observatory telemetry was accomplished with an FM-FM system. There were two independent parallel multiplexing systems. Each system had a tape recorder, a set of subcarrier oscillators, and a transmitter.

The tape recorder, running at approximately 0.65 ips, recorded the complex signal (eight non-standard low frequency sub-carriers) for 90 minutes of the 95-minute orbit. During a five-minute interval, when the spacecraft was within receiving range, the tape recorder would play back the recorded signal at 18.35 times the record speed. At this playback speed, the subcarrier oscillator frequencies became standard IRIG frequencies and modulated the spacecraft transmitter.

One subcarrier oscillator was a fixed frequency, highly stable oscillator. This oscillator

frequency was used in the tape speed compensation networks of the receiving station to compensate for permanent changes in tape recorder speed and for transient changes, such as wow and flutter, up to 300 cps.

The subcarrier oscillator output signals were recorded and played back at equal amplitudes. A pre-emphasis filter preceded the transmitter input to provide an amplitude taper commensurate with transmitter noise level and a bandwidth of 100 kc.

The transmitter consisted of two parts: the modulated driver, and the RF power amplifier. Both units used solid-state active elements. During the 90-minute record phase of the orbit, the driving signal to the RF amplifier was reduced to permit an output of 300 mw. By ground control initiation of playback, the driving signal to the power amplifier was increased to permit an output of 1.75 watts during the five-minute tape recorder playback interval. Since only one transmitter operated at a time, both were able to operate at the same frequency.

The antenna system utilized two of the spacecraft arm supporting structures as radiating elements. Antenna polarization required diversity combining of vertically and horizontally polarized signals at the receiving station. Due to a two-db loss in the antenna and RF multiplexing system, the radiated power was nominally 1.0 watt.

The command system for the observatory was a 7-tone AM system. It was capable of actuating 10 distinct command functions.

DESCRIPTION OF THE OBSERVATORY EXPERIMENTS

Solar X-Ray Spectrometer

A solar x-ray spectrometer was employed for the primary study of short wavelength x-rays (See Figure 1-4). It was designed with a sensitivity adequate to yield a spectral distribution over the 60A-400A wavelength band. With the experiment continuously pointed at the sun, sunlight entered the instrument through a slit. The sunlight was dispersed at the 2-degree grazing angle grating into the x-ray spectrum, and was focused spectrally on the "Rowland"

circle. The detector was mounted on a motor-driven carriage which moved along a track coincident with the Rowland circle and thus could continuously scan and measure the spectrum presented. The spectrometer was developed at Goddard Space Flight Center by W. Behring and W. Neupert.

20-100 Kev X-Ray Monitor

An experiment for monitoring the 20-100 kev x-ray region was devised utilizing a thin NaI (Tl) crystal scintillator. This emission band is believed to be associated with Type III radio events, and correlation will be sought between the radio observations and variations in the transient x-ray phenomena. This instrument was developed at Goddard Space Flight Center by K. Frost and W. White.

1.8 Angstrom X-Ray Monitor

An experiment for monitoring the 1.8A x-ray region was developed at Goddard Space Flight Center by K. Frost, W. White, and R. Young. It utilized two Be(Xe) ion chambers connected in parallel and collimated to reduce low-energy electron effects.

0.2-1.5 Mev Gamma Ray Monitor

This gamma ray monitor used two NaI (Tl) scintillators: one was shielded for a penumbral detection angle of 20° , and the other was unshielded and nearly isotropic in detection ability so that it could be used as a background control detector for the shielded detector. This experiment was developed at Goddard Space Flight Center by K. Frost, W. White, and K. Hallam.

50 Kev-3 Mev Gamma Ray Monitor

An experiment was designed to monitor gamma rays between 50 kev and 3 Mev. For the 50-150 kev range, a NaI (Tl) crystal scintillator monitored radiation through a lead shield. The detector operating in the 0.3-1.0 Mev and 1.0-3.0 Mev energy regions used two scintillators connected as a Compton coincidence telescope. Data were accumulated in a

storage register which was read into a shift register and presented as digital words for output data. This gamma ray monitor experiment was developed at the University of Minnesota by J. Winkler and L. Peterson.

100-500 Mev Gamma Ray Monitor

An experiment was devised to monitor gamma rays in the range from 100 Mev to 500 Mev using coincidence detection of scintillation and Cerenkov light. Gamma radiation creates an electron-positron pair in a lead sheet, generating a scintillator pulse in coincidence with Cerenkov light produced in a plastic cylinder. An anti-coincidence scintillator was placed ahead of the lead sheet as protection against primary relativistic charged particles. This experiment was developed at the University of Rochester by G. Fazio and E. Hafner.

0.510 Mev Gamma Ray Monitor

This experiment was designed to monitor the 0.510 Mev electron-positron annihilation line using a NaI (Tl) scintillator. It was developed by K. Frost and W. White at Goddard Space Flight Center.

3800A-4800A Monitor

An experiment for monitoring the 3800A-4800A band was built using a special photodiode with a filter. This experiment was capable of detecting variations of solar flux in a bandwidth as small as 0.1 percent. This band of wavelengths is important in the total energy balance studies of the sun. The experiment was developed at Goddard Space Flight Center by K. Hallam, W. White, and H. Murphy.

1100A-1250A Monitor

An experiment for measuring the intensity of the 1100A-1250A solar ultraviolet band was developed at Goddard Space Flight Center by K. Hallam, W. White, and R. Young. It used an LiF-CS₂ ion chamber, which was stopped down to retard deterioration. The region included the chromospheric Lyman-Alpha line which was strongly affected by solar activity.

Interplanetary Dust Monitor

An interplanetary dust monitor experiment was built to measure the number, momentum, and kinetic energy of dust particles by the use of a microphone and a photomultiplier with a two-micron coating of aluminum. The photomultiplier measured the luminous energy in the plasma cloud formed by impacts, and the microphone measured the mechanical impulse of the impact. This experiment was developed at Goddard Space Flight Center by M. Alexander and C. McCracken.

Neutron Monitor

A neutron detection experiment was built to determine whether or not the lower portion of the earth's radiation belt is caused by the decay of earth-emitted neutrons and the subsequent trapping of resulting electrons and protons in the earth's field. The detector was a moderated BF_3 counter. This experiment was developed at the University of California by W. Hess.

Proton-Electron Identifier

A proton-electron experiment was built to distinguish between the electron and proton ionization events. This was accomplished using a scintillator which can electronically distinguish between the two types of ionization. This experiment was developed at the University of California by Dr. C. Schroeder.

Emissivity Experiment

An emissivity experiment was built to study the stability of the emissivity of various surfaces in a space environment. Six surfaces and a black body reference were used, and equilibrium temperature was the parameter measured by thermistor detectors. This experiment was developed at the Ames Research Center by G. Robinson.

LAUNCH VEHICLE AND ORBIT

The standard Thor-Delta vehicle with the bulbous nose cone (Figure 1-8) was used to place OSO I into orbit. The vehicle could

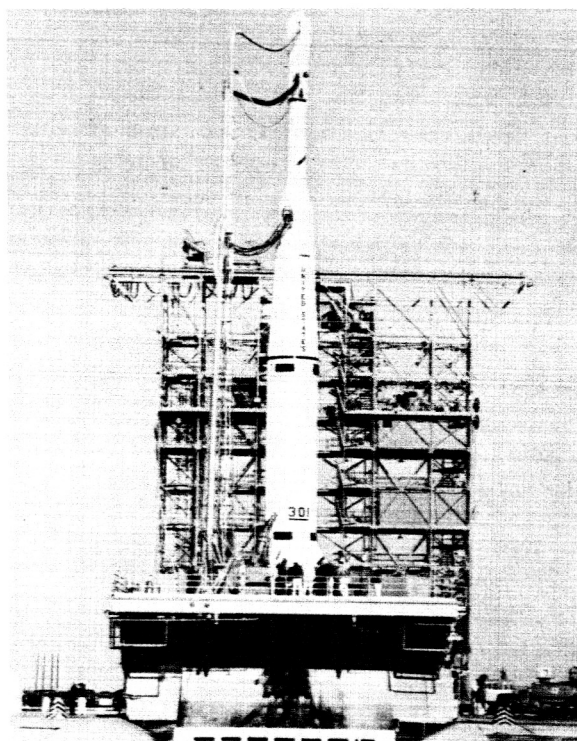


FIGURE 1-8.—Thor-Delta launch vehicle.

reliably place a 500-lb payload into the desired orbit. This orbit had to be high enough to free the spacecraft from most of the atmospheric drag and attenuation, but low enough to avoid the radiation belt which would affect some of the experiments. These requirements indicated that a 300-nautical-mile orbit, as near circular as possible, would be most satisfactory. An orbit inclination of 33 degrees was chosen for the following reasons: (1) the geometry of the launch area, (2) the required injection point, (3) the resulting trajectory, and (4) the requirement for a maximum sunlight orbit.

PAYLOAD ENVELOPE AND WEIGHT

The Thor-Delta vehicle payload shroud dictated the size of the payload envelope. The observatory was required to fit within a certain enclosure with sufficient clearance to allow for shroud buffeting motion. Having determined the payload size and shape, the next step was to keep the weight within the capability of the launch vehicle. Certain dynamics considerations dictated that the relative weights, or more

precisely, momentums, were such that considerably more scientific instrument weight could be handled than was originally expected. The design weight was to be approximately 400 lb, 100 lb less than the maximum capability of the vehicle system.

DESIGN OF THE OBSERVATORY

The Orbiting Solar Observatory was originally conceived as a gyroscope. The spinning portion was to be an open mechanical structure carrying gas bottles and batteries at the extremes for spin-up impulses and energy storage, and a stationary structure (with respect to the sun) holding the experiments to be pointed at the sun. This stationary structure was to be shaft-mounted through the open spinning wheel structure, with a bearing supported shaft through the center of a large gas bottle containing the pitch control gas supply. As the design evolved, it became evident that a higher wheel spin momentum was required than could be made available with the structure in mind, and that a life longer than a few days would be desirable. At this point, it was realized that the additional wheel weight required to increase the spin momentum could be used for payload, thus scientific experiments were added to the wheel structure. The batteries which were required to provide the power had to be charged, so a solar-cell array was added.

These developments led to the design of a two-section structure: a wheel structure that consisted of compartments to provide space for the placement of the wheel controls, batteries, telemetry equipment, and experiments; and a large flat "sail" structure to carry solar cells. From this point, the design was not changed basically, but instead of putting the spin-up gas bottles in the wheel structure, the bottles were put on extendable arms, such that the arms could be stowed for launch and extended after orbit was achieved. The reason for this was to increase the spin momentum. The most obvious problem that this configuration presented was the limit of the life of the bearings required to support the solar oriented portion upon the spinning portion. The success of the entire venture depended upon the ability of

bearings to operate reliably in the near absolute vacuum conditions of space. Early investigations indicated that bearings had been tested in laboratory vacuums before, but with unsatisfactory results. This indicated that a new kind of lubricant and lubricating technique was required. Practically all of the well known lubricants, lubrication concepts, and lubrication philosophy were of no help in the solution of the problem. A new approach was used, and a dry lubricant technique was developed and successfully tested in the laboratory. The same problem existed for slip rings and brushes. Again, a new technique was developed and tested with success.

DEVELOPMENT OF THE OBSERVATORY

The observatory development was begun by conducting three simultaneous programs. These programs were the development of the electrical control system, the basic spacecraft, and the data acquisition and command system.

Electrical Control System

The basic approach in the development of the electrical control system was to design a mechanized system which could be rapidly assembled and would perform in the same manner as the proposed orbit system. The initial effort culminated in a mock-up of the entire system on an air-bearing-mounted test fixture. The air-bearing fixture was built to simulate the spacecraft dynamically. The moments of inertia of the fixture were the same as the final payload. This fixture consisted of a large-diameter flat plate mounted horizontally on an air bearing by means of a conical center section that allowed the center of gravity of the flat plate wheel to be lower than the mounting sphere of the air bearing. The wheel then represented the spinning portion of the spacecraft and provided a mounting surface for the control equipment that would actually be mounted in the observatory wheel structure. An upper structure was then mounted on the air-bearing-supported wheel. This upper structure was shaft-mounted on the wheel structure to enable it to rotate with respect to the wheel about an axis (spin axis)

normal to the plane of the wheel. A bearing-mounted shaft, perpendicular to the spin axis of the wheel, was attached to the upper structure for mounting simulated solar oriented instruments. This method provided a fixture which had a two-axis gimbal mounting for instruments similar to the actual spacecraft. The center of gravity of the fixture was adjusted to coincide with the center of the spherical air bearing suspension ball. When balanced, this fixture essentially provided a free body that would react to external torques as would the actual observatory in orbit.

The most efficient method for providing the simulated spacecraft with control system electronics was to adapt an existing pointing control to the simulated spacecraft. The system electronics were constructed from the drawings of a balloon-borne pointing control system previously flown, enabling the use of readily available components. Gains and limits were adjusted to suit the new application. A test fixture was built to test the electronics before the controls were applied to the air-bearing fixture. This test fixture was essentially identical to the air-bearing fixture, except that it was fixed to the floor. The test fixture could rotate and had biaxial gimbaling and could wobble about the rotational axis by means of an eccentric adjustment at the drive axis. This *wobble table* fixture was used to check out the electronic and gas controls. The table fixture could be rotated, given specified wobbles, and could otherwise recreate the conditions of the air-bearing fixture with the exception of the pitch and roll capabilities of the air-bearing fixture.

At this point, a complete simulated spacecraft was available for control system development. The control system was then installed on the air-bearing fixture. The fixture was suspended on its bearing, and the wheel structure was spun up to the expected spin rate of the actual observatory. An indoor lamp was directed at the device from a distance to simulate the sun for control system acquisition and positioning. The test on the air-bearing fixture provided the necessary confidence in the method of control, the feasibility of the gas systems, and the nutation damper. Several modifications in the original

concepts were made as a result of the air-bearing fixture work.

After the operation of the control system had been proven successful, the next step in the electronics development was the prototype effort. The breadboard design, in progress as the air-bearing work was being done, was finalized using experience gained in the air-bearing work. Prototype electronics were prepared in flight configuration, and flight electronic components were requisitioned.

Prototype electronic subassemblies were installed in the prototype spacecraft after they had successfully passed qualifying vibration and thermal-vacuum tests. After the prototype system was installed and integrated, the entire prototype spacecraft was subjected to a series of environmental tests. Problems which were encountered during prototype testing were solved and the solutions applied to the flight model control system. The environmental testing of flight subassemblies was completed and the subassemblies were integrated into the flight spacecraft which was then environmentally tested.

Basic Spacecraft

As decisions were made with respect to providing a vibration-resistant structure, the basic spacecraft design evolved from the anticipated shape and size concepts into actual hardware. The design to survive the expected launch thrust was straightforward; however, a design to survive the expected launch vibration was not as simple. A structure which was designed as a flight structure was assembled and designated the vibration structural model. Simulated instruments and control equipment were mounted on this structure to simulate the eventual flight model configuration. Extensive vibration testing was accomplished on the structural model. Only a minimum of useable numerical data were derived from this testing but the structure proved to be adequate for the vibration loading imposed. From this point of the development, only minor changes were made as the work progressed to the prototype structure. The prototype structure was assembled, and prototype instruments and controls were installed.

Vibration testing revealed that the only modifications required were minor changes in the mounting bracketry hardware for some of the equipment in the wheel structure. Testing of the arms in the erection sequence indicated that the arm structure required strengthening. Following these modifications, the prototype structure survived vibration testing far in excess of that required to prove structural integrity. The flight model structure, with modifications made as the prototype testing exposed problems, successfully completed the flight environmental test with no problems.

Data Acquisition and Command System

The data acquisition system for the observatory was designed to perform the following functions:

1. Accept the analog outputs of seven data channels* and one reference channel and combine them into a frequency multiplex of eight subcarrier oscillators.
2. Demodulate and select the particular tone command channel via the receiver and decoder respectively.
3. Actuate ten spacecraft command functions.

Launch Sequence

The launch and orbital acquisition phase most severely tested the spacecraft system. The orbital acquisition sequence was programmed by digital timers which were started by centrifugal switches. These switches sensed the centrifugal force produced when the third stage and payload were spun up by the solid-propellant rockets mounted on the spin table between the second and third stages.

The switches closed properly at spin-up and the timers started properly. One hundred seconds later the timed release of the stowed arms by means of explosive squib-actuated pin pullers was properly accomplished, and the arms extended and locked as anticipated.

Shortly after arm release, third-stage engine separation occurred. The launch vehicle timed this separation.

Two hundred seconds after spin-up, the timers turned on orbit power and turned off the

*In some cases, several experimental measurements were combined into a single channel.

redundant tape recorder. The tape recorders were both running during the launch acceleration and vibration to prevent brinnelling of the bearings. With orbit power turned on, data recording began on the main tape recorder.

Also, at 200 seconds after timer start the despin gas jets were turned on. The spin rate for third-stage thrust stability was a nominal 120 rpm, but the spacecraft operational spin rate was required to be 30 rpm. The extension of the arms reduced the spin rate somewhat, but it was necessary to despin from this rate to 30 rpm. The despin gas jets slowed the spin rate until the automatic spin-up controller sensed that the rate was at 30 rpm. At this time, the automatic spin-up controller turned on the spin-up gas jets to produce a known incremental spin-up impulse and permanently disconnect the despin circuitry so that no accidental despin was possible. This sequence left the spacecraft spinning at the nominal control rate.

After 400 seconds, the upper structure and nutation damper *bob*, which had been latched during the powered flight phase of the launch to prevent mechanical damage, were released by explosive squib-actuated pin pullers. At the same time, a latching relay applied the error voltage to the azimuth torque motor drive electronics so that the azimuth positioning control system could begin to orient the upper structure to an attitude fixed relative to the sun. The upper structure, which was spinning with the wheel when latched, slowed and then locked on the sun. The unlatched nutation damper quickly absorbed the energy of nutation. No detectable nutation was present after the azimuth pointing controller had positioned the upper structure.

At 800 seconds after spin-up (the last timed function of the timers), the elevation gimbal was unlatched by means of an explosive squib-actuated pin puller. The elevation positioning servo mechanism pointed the upper structure experiments to the center of the sun at this time.

At 800 seconds, the timers also repeated all of the previous functions to give a higher probability that all functions had successfully occurred. At the end of this sequence, the timers turned themselves off.

Chapter 2

SPACECRAFT DYNAMICS

ANALYSIS OF SPACECRAFT RIGID-BODY SYSTEM

The unique biaxial pointing control of the Orbiting Solar Observatory utilized the entire vehicle as part of the controlled platform. Coarse elevation control of the stabilized upper section was accomplished by controlling the vehicle attitude with on-off jets. This was possible because of the gyroscopic property of the spinning wheel. Fine elevation and azimuth positioning of the instruments were accomplished by electrical servo motor control. In addition to the above, it was necessary to use a nutation damper to conserve servo power by eliminating the undersirable motions of the spacecraft.

Due to the complexity of the system, it seemed desirable to simulate the complete control system of the OSO on a computer before proceeding too far into the hardware phase.

The equations of motion for the four-body problem were derived and then run on an analog computer to check the parameters of the system and to observe, if possible, any unexpected effects.

Coordinate System

The coordinate system used is shown in Figure 2-1. The x, y, z axis system is an inertial system, and the M, N, S system is

produced by a rotation, α , about the z -axis, followed by a rotation, θ , about the N -axis.

A principal axis system, x_1, y_1, z_1 , is attached to the wheel and is formed by a rotation, ϕ' , about the S -axis.

Axes x_2, y_2, z_2 form a principal axis system attached to the upper structure which is formed by a rotation, ϕ'' , about the S -axis. The x_2 -axis is along the long dimension of the solar array.

Axes x_3, y_3, z_3 form a principal axis system attached to the pointed instruments which is formed from the x_2, y_2, z_2 system by a rotation, ϵ , about the x_2 axis.

The symbols d_1, d_2 , and d_3 denote the distance from the spacecraft center of gravity to the center of gravity of the wheel, upper structure, and pointed instrument package, respectively. The distance from the spacecraft center of gravity to the point of suspension of the nutation damper is d_4 .

The equation of transformation between the inertial coordinate system and the M, N, S system is given by

$$\begin{pmatrix} x \\ y \\ z \end{pmatrix} = \begin{pmatrix} \cos \alpha \cos \theta & -\sin \alpha & \cos \alpha \sin \theta \\ \sin \alpha \cos \theta & \cos \alpha & \sin \alpha \sin \theta \\ -\sin \theta & 0 & \cos \theta \end{pmatrix} \begin{pmatrix} M \\ N \\ S \end{pmatrix} \quad (2.1)$$

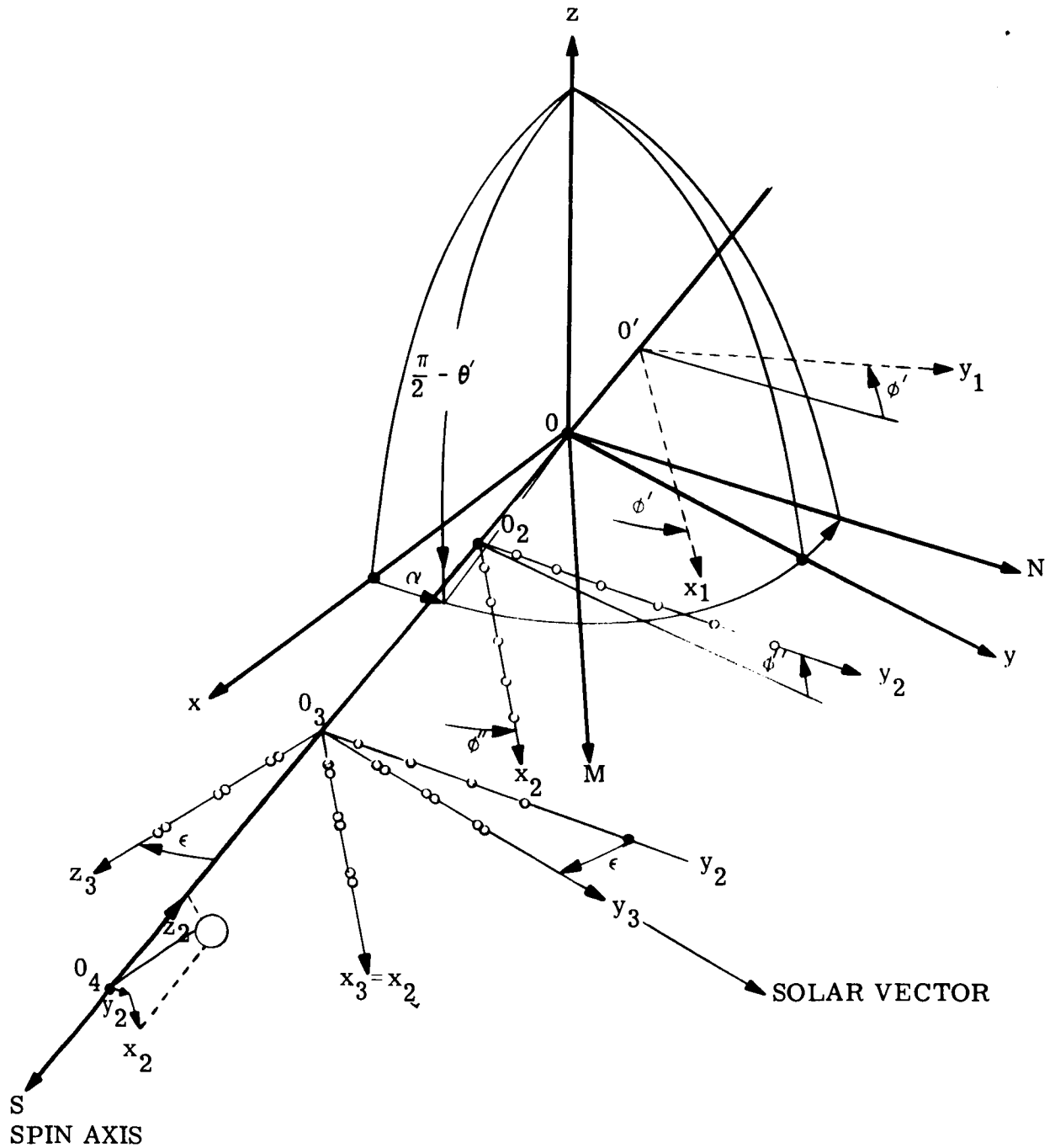
and the transformation between the M, N, S system and the x_3, y_3, z_3 system is given by

$$\begin{pmatrix} x_3 \\ y_3 \\ z_3 \end{pmatrix} = \begin{pmatrix} \cos \phi'' & \sin \phi'' & 0 \\ -\cos \epsilon \sin \phi'' & \cos \epsilon \cos \phi'' & \sin \epsilon \\ \sin \epsilon \sin \phi'' & -\sin \epsilon \cos \phi'' & \cos \epsilon \end{pmatrix} \begin{pmatrix} M \\ N \\ S \end{pmatrix} \quad (2.2)$$

The equations of transformation between the M, N, S system and the x_2, y_2, z_2 system can be obtained by setting $\epsilon=0$. The equations of

transformation between the M, N, S system and the x_1, y_1, z_1 system can be obtained by setting $\epsilon=0$ and by replacing ϕ'' by ϕ' .

ORBITING SOLAR OBSERVATORY



S
SPIN AXIS

α AND $\frac{\pi}{2} - \theta'$ - SPIN AXIS ORIENTATION

ϕ' - WHEEL POSITION

ϕ'' - UPPER STRUCTURE POSITION

ϵ - INSTRUMENT POSITION

(x_2, y_2, z_2) - DAMPING BODY POSITION

$$0 \ 0_1 = d_1$$

$$0 \ 0_2 = d_2$$

$$0 \ 0_3 = d_3$$

$$0 \ 0_4 = d_4$$

FIGURE 2-1.—Coordinate system.

Derivation of Equations of Motion

The angular velocities of the M, N, S system relative to inertial space are given by

$$\omega_M = -\dot{\alpha} \sin \theta, \omega_N = \dot{\theta} \text{ and } \omega_S = \dot{\alpha} \cos \theta \quad (2.3)$$

Therefore, the angular velocities in the x_3, y_3, z_3 system are given by

$$\begin{aligned} \omega_{x_3} &= -\dot{\alpha} \sin \theta \cos \phi'' + \dot{\theta} \sin \phi'' + \dot{\epsilon} \\ \omega_{y_3} &= (\dot{\alpha} \sin \theta \sin \phi'' + \dot{\theta} \cos \phi'') \cos \epsilon \\ &\quad + (\dot{\phi}'' + \dot{\alpha} \cos \theta) \sin \epsilon \end{aligned}$$

and

$$\begin{aligned} \omega_{z_3} &= -(\dot{\alpha} \sin \theta \sin \phi'' + \dot{\theta} \cos \phi'') \sin \epsilon \\ &\quad + (\dot{\phi}'' + \dot{\alpha} \cos \theta) \cos \epsilon \quad (2.4) \end{aligned}$$

The kinetic energy of the pointed instrument is

$$T_3 = \frac{1}{2} [m_3 d_3^2 (\omega_M^2 + \omega_N^2) + I_{x_3} \omega_{x_3}^2 + I_{y_3} \omega_{y_3}^2 + I_{z_3} \omega_{z_3}^2] \quad (2.5)$$

where m_3 is the mass of the pointed instruments and $I_{x_3}, I_{y_3}, I_{z_3}$ are the moments of inertia of the instruments about the indicated axes.

The partial derivatives of the kinetic energy with respect to the generalized coordinates, the derivative of the generalized coordinate, and the total derivative of the latter are as follows:

$$\begin{aligned} \frac{\partial T_3}{\partial \dot{\alpha}} &= m_3 d_3^2 \dot{\alpha} \sin^2 \theta + \omega_{x_3} I_{x_3} (-\sin \theta \cos \phi'') + \omega_{y_3} I_{y_3} (\sin \theta \sin \phi'' \cos \epsilon \\ &\quad + \cos \theta \sin \epsilon) + \omega_{z_3} I_{z_3} (-\sin \theta \sin \phi'' \sin \epsilon + \cos \theta \cos \epsilon) \end{aligned}$$

$$\begin{aligned} \frac{d}{dt} \left(\frac{\partial T_3}{\partial \dot{\alpha}} \right) &= m_3 d_3^2 (\ddot{\alpha} \sin^2 \theta + \dot{\alpha} \dot{\theta} \sin 2\theta) + I_{x_3} [\dot{\omega}_{x_3} (-\sin \theta \cos \phi'') + \omega_{x_3} (-\dot{\theta} \cos \theta \cos \phi'' + \dot{\phi}'' \sin \theta \sin \phi'')] \\ &\quad + I_{y_3} [\dot{\omega}_{y_3} (\sin \theta \sin \phi'' \cos \epsilon + \cos \theta \sin \epsilon) + \omega_{y_3} (\cos \theta \sin \phi'' \cos \epsilon - \sin \theta \sin \epsilon) \\ &\quad + \dot{\phi}'' \omega_{y_3} \sin \theta \cos \phi'' \cos \epsilon + \dot{\epsilon} \omega_{y_3} (-\sin \theta \sin \phi'' \sin \epsilon + \cos \theta \cos \epsilon)] \\ &\quad + I_{z_3} [\dot{\omega}_{z_3} (-\sin \theta \sin \phi'' \sin \epsilon + \cos \theta \cos \epsilon) - \dot{\theta} \omega_{z_3} (\cos \theta \sin \phi'' \sin \epsilon + \sin \theta \cos \epsilon) \\ &\quad - \dot{\phi}'' \omega_{z_3} \sin \theta \cos \phi'' \sin \epsilon - \dot{\epsilon} \omega_{z_3} (\sin \theta \sin \phi'' \cos \epsilon + \cos \theta \sin \epsilon)] \quad (2.6) \end{aligned}$$

$$\frac{\partial T_3}{\partial \alpha} = 0$$

$$\frac{\partial T_3}{\partial \dot{\theta}} = m_3 d_3^2 \dot{\theta} + \omega_{x_3} I_{x_3} \sin \phi'' + \omega_{y_3} I_{y_3} \cos \phi'' \cos \epsilon - \omega_{z_3} I_{z_3} \cos \phi'' \sin \epsilon \quad (2.7)$$

$$\begin{aligned} \frac{d}{dt} \left(\frac{\partial T_3}{\partial \dot{\theta}} \right) &= m_3 d_3^2 \ddot{\theta} + I_{x_3} (\dot{\omega}_{x_3} \sin \phi'' + \dot{\phi}'' \omega_{x_3} \cos \phi'') + I_{y_3} (\dot{\omega}_{y_3} \cos \phi'' \cos \epsilon \\ &\quad - \dot{\phi}'' \omega_{y_3} \sin \phi'' \cos \epsilon - \dot{\epsilon} \omega_{y_3} \cos \phi'' \sin \epsilon) \\ &\quad - I_{z_3} (\dot{\omega}_{z_3} \cos \phi'' \sin \epsilon - \dot{\phi}'' \omega_{z_3} \sin \phi'' \sin \epsilon + \dot{\epsilon} \omega_{z_3} \cos \phi'' \cos \epsilon) \quad (2.8) \end{aligned}$$

$$\begin{aligned} \frac{\partial T_3}{\partial \dot{\theta}} &= \frac{1}{2} m_3 d_3^2 \dot{\alpha}^2 \sin 2\theta - \omega_{x_3} I_{x_3} \cos \theta \cos \phi'' + \dot{\alpha} \omega_{y_3} I_{y_3} (\cos \theta \sin \phi'' \cos \epsilon \\ &\quad - \sin \theta \sin \epsilon) - \dot{\alpha} \omega_{z_3} I_{z_3} (\cos \theta \sin \phi'' \sin \epsilon + \sin \theta \cos \epsilon) \quad (2.9) \end{aligned}$$

$$\frac{\partial T_3}{\partial \dot{\phi}''} = \omega_{y_3} I_{y_3} \sin \epsilon + \omega_{z_3} I_{z_3} \cos \epsilon$$

$$\frac{d}{dt} \left(\frac{\partial T_3}{\partial \dot{\phi}''} \right) = I_{y_3} (\dot{\omega}_{y_3} \sin \epsilon + \dot{\epsilon} \omega_{y_3} \cos \epsilon) + I_{z_3} (\dot{\omega}_{z_3} \cos \epsilon - \dot{\epsilon} \omega_{z_3} \sin \epsilon) \quad (2.10)$$

$$\frac{\partial T_3}{\partial \dot{\phi}''} = I_{x_3} (\dot{\alpha} \omega_{x_3} \sin \theta \sin \phi'' + \dot{\theta} \omega_{x_3} \cos \phi'') + (\dot{\alpha} \sin \theta \cos \phi'' - \dot{\theta} \sin \phi'') (\omega_{y_3} I_{y_3} \cos \epsilon - \omega_{z_3} I_{z_3} \sin \epsilon) \quad (2.11)$$

$$\frac{\partial T_3}{\partial \dot{\epsilon}} = \omega_{x_3} I_{x_3} \quad \frac{d}{dt} \left(\frac{\partial T_3}{\partial \dot{\epsilon}} \right) = \dot{\omega}_{x_3} I_{x_3} \quad (2.12)$$

$$\begin{aligned} \frac{\partial T_3}{\partial \epsilon} = & -(\dot{\alpha} \sin \theta \sin \phi'' + \theta \cos \phi'')(\omega_{y_3} I_{y_3} \sin \epsilon + \omega_{z_3} I_{z_3} \cos \epsilon) \\ & + (\dot{\phi}'' + \dot{\alpha} \cos \theta)(\omega_{y_3} I_{y_3} \cos \epsilon - \omega_{z_3} I_{z_3} \sin \epsilon) \end{aligned} \quad (2.13)$$

Equations (2.3) through (2.13) may be simplified by assuming $\alpha, \phi'', \dot{\alpha}, \epsilon$ small and $\theta = \frac{\pi}{2} - \theta'$ where θ' is small. Then

$$\omega_{x_3} \cong -\dot{\alpha} + \dot{\epsilon} \quad \omega_{y_3} \cong \dot{\theta} \quad \omega_{z_3} \cong \dot{\phi}'' \quad (2.14)$$

and

$$\left. \begin{aligned} \dot{\omega}_{x_3} &= \ddot{\epsilon} - \ddot{\alpha} + \dot{\theta} \dot{\phi}'' \\ \dot{\omega}_{y_3} &= \ddot{\theta} + \dot{\phi}''(\dot{\epsilon} + \dot{\alpha}) \\ \dot{\omega}_{z_3} &= \ddot{\phi} - \dot{\theta}(\dot{\epsilon} + \dot{\alpha}) \end{aligned} \right\} \quad (2.15)$$

For the pointed instruments the Lagrangians become:

$$\left. \begin{aligned} \frac{d}{dt} \left(\frac{\partial T_3}{\partial \dot{\alpha}} \right) - \left(\frac{\partial T_3}{\partial \alpha} \right) &= m_3 d_3^2 \ddot{\alpha} + I_{x_3}(\ddot{\alpha} - \ddot{\epsilon} - \dot{\theta} \dot{\phi}'') + (I_{y_3} - I_{z_3}) \dot{\theta} \dot{\phi}'' \\ \frac{d}{dt} \left(\frac{\partial T_3}{\partial \dot{\theta}} \right) - \left(\frac{\partial T_3}{\partial \theta} \right) &= m_3 d_3^2 \ddot{\theta} + I_{y_3}[\ddot{\theta} + \dot{\phi}''(\dot{\epsilon} + \dot{\alpha})] + \dot{\phi}''(\dot{\epsilon} - \dot{\alpha})(I_{x_3} - I_{z_3}) \\ \frac{d}{dt} \left(\frac{\partial T_3}{\partial \dot{\phi}''} \right) - \left(\frac{\partial T_3}{\partial \phi''} \right) &= I_{z_3}[\ddot{\phi}'' - \dot{\theta}(\dot{\epsilon} + \dot{\alpha})] + \dot{\theta}(\dot{\alpha} - \dot{\epsilon})(I_{x_3} - I_{y_3}) \\ \frac{d}{dt} \left(\frac{\partial T_3}{\partial \dot{\epsilon}} \right) - \left(\frac{\partial T_3}{\partial \epsilon} \right) &= -I_{x_3}(\ddot{\alpha} - \ddot{\epsilon} - \dot{\theta} \dot{\phi}'') - (I_{y_3} - I_{z_3}) \dot{\theta} \dot{\phi}'' \end{aligned} \right\} \quad (2.16)$$

For the upper structure, the Lagrangians become:

$$\left. \begin{aligned} \frac{d}{dt} \left(\frac{\partial T_2}{\partial \dot{\alpha}} \right) - \left(\frac{\partial T_2}{\partial \alpha} \right) &= m_2 d_2^2 \ddot{\alpha} + I_{x_2}(\ddot{\alpha} - \dot{\theta} \dot{\phi}'') + (I_{y_2} - I_{z_2}) \dot{\theta} \dot{\phi}'' \\ \frac{d}{dt} \left(\frac{\partial T_2}{\partial \dot{\theta}} \right) - \left(\frac{\partial T_2}{\partial \theta} \right) &= m_2 d_2^2 \ddot{\theta} + I_{y_2}(\ddot{\theta} + \dot{\alpha} \dot{\phi}'') - \dot{\alpha} \dot{\phi}''(I_{x_2} - I_{z_2}) \\ \frac{d}{dt} \left(\frac{\partial T_2}{\partial \dot{\phi}''} \right) - \left(\frac{\partial T_2}{\partial \phi''} \right) &= I_{z_2}(\ddot{\phi}'') + \dot{\alpha} \dot{\theta}(I_{x_2} - I_{y_2} - I_{z_2}) \end{aligned} \right\} \quad (2.17)$$

And for the wheel, the Lagrangians become:

$$\left. \begin{aligned} \frac{d}{dt} \frac{\partial T_1}{\partial \dot{\alpha}} - \frac{\partial T_1}{\partial \alpha} &= m_1 d_1^2 \ddot{\alpha} + I_T \ddot{\alpha} - I_S \dot{\theta} \dot{\phi}' \\ \frac{d}{dt} \frac{\partial T_1}{\partial \dot{\theta}} - \frac{\partial T_1}{\partial \theta} &= m_1 d_1^2 \ddot{\theta} + I_T \ddot{\theta} + I_S \dot{\alpha} \dot{\phi}' \\ \frac{d}{dt} \frac{\partial T_1}{\partial \dot{\phi}'} - \frac{\partial T_1}{\partial \phi'} &= I_S(\dot{\phi}' - \dot{\alpha} \dot{\theta}) \end{aligned} \right\} \quad (2.18)$$

since $I_{x_1} = I_{y_1} = I_T$. The moment of inertia of the wheel about its spin axis is I_S .

The kinetic energy of the nutation damper (T_4) is given by

$$T_4 = \frac{1}{2} m_4 (\dot{x}^2 + \dot{y}^2 + \dot{z}^2) \quad (2.19)$$

Its potential energy, V_4 , is expressed by

$$V_4 = \frac{1}{2} k (x_2^2 + y_2^2 + z_2^2) \quad (2.20)$$

and its dissipation function, F_4 , may be written

$$F_4 = \frac{1}{2} c (\dot{x}_2^2 + \dot{y}_2^2 + \dot{z}_2^2) \quad (2.21)$$

In order to write the Lagrangian for the damper, it is necessary to express Equation (2.19) in terms of the x_2, y_2, z_2 coordinate system. The equation of transformation between these two systems is given by

$$\begin{pmatrix} x \\ y \\ z \end{pmatrix} = \begin{pmatrix} \cos \alpha \cos \theta & -\sin \alpha & \cos \alpha \sin \theta \\ \sin \alpha \cos \theta & \cos \alpha & \sin \alpha \sin \theta \\ -\sin \theta & 0 & \cos \theta \end{pmatrix} \begin{pmatrix} \cos \phi'' & -\sin \phi'' & 0 \\ \sin \phi'' & \cos \phi'' & 0 \\ 0 & 0 & 1 \end{pmatrix} \begin{pmatrix} x_2 \\ y_2 \\ z_2 + d_4 \end{pmatrix}$$

or

$$\begin{pmatrix} x \\ y \\ z \end{pmatrix} = \begin{pmatrix} \cos \alpha \cos \theta \cos \phi'' & -\cos \alpha \cos \theta \sin \phi'' & \cos \alpha \sin \theta \\ -\sin \alpha \sin \phi'' & -\sin \alpha \cos \phi'' & \\ \sin \alpha \cos \theta \cos \phi'' & -\sin \alpha \cos \theta \sin \phi'' & \sin \alpha \sin \theta \\ +\cos \alpha \sin \phi'' & +\cos \alpha \cos \phi'' & \\ -\sin \theta \cos \phi'' & \sin \theta \sin \phi'' & \cos \theta \end{pmatrix} \begin{pmatrix} x_2 \\ y_2 \\ z_2 + d_4 \end{pmatrix}$$

Therefore,

$$\begin{aligned} x &= x_2 (\cos \alpha \cos \theta \cos \phi'' - \sin \alpha \sin \phi'') - y_2 (\cos \alpha \cos \theta \sin \phi'' + \sin \alpha \cos \phi'') + (z_2 + d_4) \cos \alpha \sin \theta \\ y &= x_2 (\sin \alpha \cos \theta \cos \phi'' + \cos \alpha \sin \phi'') + y_2 (-\sin \alpha \cos \theta \sin \phi'' + \cos \alpha \cos \phi'') + (z_2 + d_4) \sin \alpha \sin \theta \\ z &= -x_2 \sin \theta \cos \phi'' + y_2 \sin \theta \sin \phi'' + (z_2 + d_4) \cos \theta \end{aligned}$$

or

$$\begin{aligned} x &= \cos \alpha \cos \theta (x_2 \cos \phi'' - y_2 \sin \phi'') - \sin \alpha (x_2 \sin \phi'' + y_2 \cos \phi'') + (z_2 + d_4) \cos \alpha \sin \theta \\ y &= \sin \alpha \cos \theta (x_2 \cos \phi'' - y_2 \sin \phi'') + \cos \alpha (x_2 \sin \phi'' + y_2 \cos \phi'') + (z_2 + d_4) \sin \alpha \sin \theta \\ z &= -\sin \theta (x_2 \cos \phi'' - y_2 \sin \phi'') + (z_2 + d_4) \cos \theta \end{aligned}$$

or

$$\left. \begin{aligned} x &= a \alpha_c \theta_c - b \alpha_s + (z_2 + d_4) \alpha_c \theta_s \\ y &= a \alpha_s \theta_c + b \alpha_c + (z_2 + d_4) \alpha_s \theta_s \\ z &= -a \theta_s + (z_2 + d_4) \theta_c \end{aligned} \right\} \quad (2.22)$$

where

$$\left. \begin{aligned} a &= x_2 \cos \phi'' - y_2 \sin \phi'' \\ b &= x_2 \sin \phi'' + y_2 \cos \phi'' \end{aligned} \right\} \quad (2.23)$$

using the argument with the subscript indicating the function, i.e., $\sin \theta$ becomes θ_s .

Differentiating Equation (2.22) results in the equations

$$\left. \begin{aligned} \dot{x} &= \dot{a}\alpha_c\theta_c - \dot{b}\alpha_s + \dot{z}_2\alpha_c\theta_s - \dot{\alpha}\alpha_s p + \dot{\theta}\alpha_c q - \dot{b}\dot{\alpha}\alpha_c \\ \dot{y} &= \dot{a}\alpha_s\theta_c + \dot{b}\alpha_c + \dot{z}_2\alpha_s\theta_s + \dot{\alpha}\alpha_c p + \dot{\theta}\alpha_s q - \dot{b}\dot{\alpha}\alpha_s \\ \dot{z} &= -\dot{a}\theta_s + \dot{z}_2\theta_c - \dot{\theta}p \end{aligned} \right\} \quad (2.24)$$

where

$$\left. \begin{aligned} p &= a\theta_c + (z_2 + d_4)\theta_s \\ q &= -a\theta_s + (z_2 + d_4)\theta_c \end{aligned} \right\} \quad (2.25)$$

Therefore,

$$\begin{aligned} \dot{x}^2 + \dot{y}^2 + \dot{z}^2 &= \dot{a}^2 + \dot{b}^2 + \dot{z}_2^2 + \dot{\alpha}^2 p^2 + \dot{\theta}^2 (p^2 + q^2) + \dot{b}^2 \dot{\alpha}^2 + 2\dot{a}\dot{\theta}(q\theta_c + p\theta_s) \\ &\quad + 2\dot{a}\dot{\alpha}b\theta_c + 2\dot{b}\dot{\alpha}p + 2\dot{z}_2\dot{\theta}(q\theta_s + p\theta_c) - 2\dot{z}_2\dot{\alpha}b\theta_s - 2\dot{\theta}\dot{\alpha}qb \end{aligned} \quad (2.26)$$

Substituting Equations (2.23) and (2.25) into Equation (2.26) results in

$$\begin{aligned} \dot{x}^2 + \dot{y}^2 + \dot{z}^2 &= \dot{x}_2^2 + \dot{y}_2^2 + \dot{z}_2^2 + \dot{\phi}''^2 (x_2^2 + y_2^2) + 2\dot{\phi}'' (\dot{y}_2 x_2 - \dot{x}_2 y_2) + \dot{\alpha}^2 [x_2^2 \phi_s''^2 + 2x_2 y_2 \phi_s' \phi_c'' + y_2^2 \phi_c''^2 \\ &\quad + \theta_c^2 (x_2^2 \phi_c''^2 - 2x_2 y_2 \phi_s' \phi_c'' + y_2^2 \phi_s''^2) + 2(x_2 \phi_c'' - y_2 \phi_s'') (z_2 + d_4) \theta_s \theta_c + (z_2 + d_4)^2 \theta_s^2] \\ &\quad + 2\dot{\theta} [\dot{x}_2 \phi_c'' - \dot{y}_2 \phi_s'' - \phi'' (x_2 \phi_s'' + y_2 \phi_c'')] (z_2 + d_4) + 2\dot{\alpha} \theta_c [\dot{x}_2 \phi_c'' - y_2 \phi_s'' - \phi'' (x_2 \phi_s'' \\ &\quad + y_2 \phi_c'')] (x_2 \phi_s'' + y_2 \phi_c'') + 2\dot{\alpha} [\dot{x}_2 \phi_s'' + \dot{y}_2 \phi_c'' + \phi'' (x_2 \phi_c'' - y_2 \phi_s'')] [\theta_c (x_2 \phi_c'' - y_2 \phi_s'') \\ &\quad + (z_2 + d_4) \theta_s] + 2\dot{\theta} \dot{z}_2 [(x_2 \phi_c'' - y_2 \phi_s'') \cos 2\theta + 2(z_2 + d_4) \theta_s \theta_c] - 2\dot{\alpha} \dot{z}_2 \theta_s (x_2 \phi_s'' + y_2 \phi_c'') \\ &\quad - 2\dot{\alpha} \dot{\theta} [-\theta_s (x_2^2 - y_2^2) \phi_s' \phi_c'' + \theta_c (x_2 \phi_s'' + y_2 \phi_c'') (z_2 + d_4) - \theta_s x_2 y_2 (\phi_c''^2 - \phi_s''^2)] \end{aligned} \quad (2.27)$$

By assuming that $\theta = \frac{\pi}{2} - \theta'$, where θ' is small, and that α and ϕ'' are small; and by dropping third order terms in α , α , θ' , $\dot{\theta}$, ϕ'' , $\dot{\phi}''$, $\dot{x}_2$, $\dot{y}_2$, $\dot{z}_2$ Equation (2.27) becomes

$$\begin{aligned} T_4 &= \frac{m_4}{2} (\dot{x}^2 + \dot{y}^2 + \dot{z}^2) \cong \frac{m_4}{2} \{ \dot{x}_2^2 + \dot{y}_2^2 + \dot{z}_2^2 + \phi''^2 (x_2^2 + y_2^2) + 2\dot{\phi}'' (\dot{y}_2 x_2 - \dot{x}_2 y_2) + \dot{\alpha}^2 [y_2^2 + (z_2 + d_4)^2] \\ &\quad + \theta^2 [x_2^2 + (z_2 + d_4)^2] + 2\dot{\theta} (\dot{x}_2 - \phi'' y_2) (z_2 + d_4) + 2\dot{\alpha} (\dot{y}_2 + \phi'' x_2) (z_2 + d_4) - 2\dot{\alpha} \dot{z}_2 y_2 - 2\dot{\theta} \dot{z}_2 x_2 + 2\dot{\alpha} \dot{\theta} x_2 y_2 \} \end{aligned} \quad (2.28)$$

In order to write the Lagrangian equations, the following quantities are derived:

$$\begin{aligned} \frac{\partial T_4}{\partial \dot{\alpha}} &= m_4 \{ \dot{\alpha} [y_2^2 + (z_2 + d_4)^2] + (\dot{y}_2 + \phi'' x_2) (z_2 + d_4) - \dot{z}_2 y_2 + \dot{\theta} x_2 y_2 \} \\ \frac{d}{dt} \left(\frac{\partial T_4}{\partial \dot{\alpha}} \right) &= m_4 \{ \ddot{\alpha} [y_2^2 + (z_2 + d_4)^2] + \dot{\alpha} [2\dot{y}_2 y_2 + 2\dot{z}_2 (z_2 + d_4)] + (\ddot{y}_2 + \phi'' \ddot{x}_2 + \dot{\phi}'' \dot{x}_2) (z_2 + d_4) \\ &\quad + (\dot{y}_2 + \phi'' x_2) \dot{z}_2 - (\dot{z}_2 y_2 + \dot{z}_2 \dot{y}_2) + \ddot{\theta} x_2 y_2 + \dot{\theta} \dot{x}_2 y_2 + \dot{\theta} x_2 \dot{y}_2 \} \\ &\cong m_4 \{ \ddot{\alpha} [y_2^2 + (z_2 + d_4)^2] + \ddot{y}_2 (z_2 + d_4) - \ddot{z}_2 y_2 + \ddot{\theta} x_2 y_2 \} \\ \frac{\partial T_4}{\partial \dot{\theta}} &= m_4 \{ \dot{\theta} [x_2^2 + (z_2 + d_4)^2] + (\dot{x}_2 - \phi'' y_2) (z_2 + d_4) - \dot{z}_2 x_2 + \dot{\alpha} x_2 y_2 \} \\ \frac{d}{dt} \left(\frac{\partial T_4}{\partial \dot{\theta}} \right) &= m_4 \{ \ddot{\theta} [x_2^2 + (z_2 + d_4)^2] + 2\dot{\theta} [x_2 \dot{x}_2 + (z_2 + d_4) \dot{z}_2] + (\ddot{x}_2 - \phi'' \ddot{y}_2 - \dot{\phi}'' \dot{y}_2) (z_2 + d_4) \\ &\quad + (\dot{x}_2 - \phi'' y_2) \dot{z}_2 - (\dot{z}_2 x_2 + \dot{z}_2 \dot{x}_2) + \ddot{\alpha} x_2 y_2 + \dot{\alpha} (\dot{x}_2 y_2 + x_2 \dot{y}_2) \} \end{aligned} \quad (2.29)$$

or •

$$\frac{d}{dt} \left(\frac{\partial T_4}{\partial \dot{\theta}} \right) \cong m_4 \{ [x_2^2 + (z_2 + d_4)^2] \ddot{\theta} + \ddot{x}_2(z_2 + d_4) - \ddot{z}_2 x_2 + \ddot{\alpha} x_2 y_2 \} \quad (2.30)$$

$$\frac{\partial T_4}{\partial \dot{\phi}''} = m_4 [\dot{\phi}'' (x_2^2 + y_2^2) + (\dot{y}_2 x_2 - \dot{x}_2 y_2) - \dot{\theta} y_2 (z_2 + d_4) + \dot{\alpha} x_2 (z_2 + d_4)]$$

$$\begin{aligned} \frac{d}{dt} \left(\frac{\partial T_4}{\partial \dot{\phi}''} \right) = m_4 [(x_2^2 + y_2^2) + 2\dot{\phi}'' (x_2 \dot{x}_2 + y_2 \dot{y}_2) + (\dot{y}_2 x_2 - \dot{x}_2 y_2) \\ + (\ddot{\alpha} x_2 + \dot{\alpha} \dot{x}_2 - \dot{\theta} \dot{y}_2) (z_2 + d_4) + (\dot{\alpha} x_2 - \dot{\theta} y_2) \dot{z}_2] \cong 0 \end{aligned} \quad (2.31)$$

$$\frac{\partial T_4}{\partial \dot{x}_2} = m_4 [\dot{x}_2 - \dot{\phi}'' y_2 + \dot{\theta} (z_2 + d_4)]$$

$$\frac{d}{dt} \left(\frac{\partial T_4}{\partial \dot{x}_2} \right) = m_4 [\ddot{x}_2 - \ddot{\phi}'' y_2 - \dot{\phi}'' \dot{y}_2 + \ddot{\theta} (z_2 + d_4) + \dot{\theta} \dot{z}_2]$$

or

$$\frac{d}{dt} \left(\frac{\partial T_4}{\partial \dot{x}_2} \right) \cong m_4 [\ddot{x}_2 + (z_2 + d_4) \ddot{\theta}] \quad (2.32)$$

$$\frac{\partial T_4}{\partial \dot{y}_2} = m_4 [\dot{y}_2 + \dot{\phi}'' x_2 + \dot{\alpha} (y_2 + d_4)]$$

$$\frac{d}{dt} \left(\frac{\partial T_4}{\partial \dot{y}_2} \right) = m_4 [\ddot{y}_2 + \ddot{\phi}'' x_2 + \dot{\phi}'' \dot{x}_2 + \ddot{\alpha} (y_2 + d_4) + \dot{\alpha} \dot{z}_2]$$

or

$$\frac{d}{dt} \left(\frac{\partial T_4}{\partial \dot{y}_2} \right) \cong m_4 [\ddot{y}_2 + (z_2 + d_4) \ddot{\alpha}] \quad (2.33)$$

$$\frac{\partial T_4}{\partial \dot{z}_2} = m_4 [\dot{z}_2 - \dot{\alpha} y_2 - \dot{\theta} x_2]$$

$$\frac{d}{dt} \left(\frac{\partial T_4}{\partial \dot{z}_2} \right) \cong m_4 [\ddot{z}_2 - \ddot{\alpha} y_2 - \ddot{\theta} x_2] \quad (2.34)$$

$$\frac{\partial T_4}{\partial \alpha} = \frac{\partial T_4}{\partial \theta} = \frac{\partial T_4}{\partial \phi''} = 0 \quad (2.35)$$

$$\frac{\partial T_4}{\partial x_2} = m_4 [\dot{\phi}''^2 x_2 + \dot{\phi}'' \dot{y}_2 + \dot{\theta}^2 x_2 - \dot{\theta} \dot{z}_2 + \dot{\alpha} \dot{\theta} y_2] \quad (2.36)$$

$$\frac{\partial T_4}{\partial y_2} = m_4 [\dot{\phi}''^2 y_2 - \dot{\phi}'' \dot{x}_2 + \dot{\alpha}^2 y_2 - \dot{\theta} \dot{\phi}'' (z_2 + d_4) - \dot{\alpha} \dot{z}_2 + \dot{\alpha} \dot{\theta} x_2] \quad (2.37)$$

$$\frac{\partial T_4}{\partial z_2} = m_4 [(\dot{\alpha}^2 + \dot{\theta}^2) (z_2 + d_4) + \dot{\theta} (\dot{x}_2 - \dot{\phi}'' y_2) + \dot{\alpha} (\dot{y}_2 + \dot{\phi}'' x_2)] \quad (2.38)$$

$$\frac{\partial V}{\partial \alpha} = \frac{\partial V}{\partial \theta} = \frac{\partial V}{\partial \phi''} = 0 \quad (2.39)$$

$$\frac{\partial V}{\partial x_2} = k x_2 \quad (2.40)$$

$$\frac{\partial V}{\partial y_2} = k y_2 \quad (2.41)$$

$$\frac{\partial V}{\partial z_2} = k z_2 \quad (2.42)$$

$$\frac{\partial F}{\partial \dot{\alpha}} = \frac{\partial F}{\partial \dot{\theta}} = \frac{\partial F}{\partial \dot{\phi}'} = 0 \quad (2.43)$$

$$\frac{\partial F}{\partial \dot{x}_2} = c\dot{x}_2 \quad (2.44)$$

$$\frac{\partial F}{\partial \dot{y}_2} = c\dot{y}_2 \quad (2.45)$$

$$\frac{\partial F}{\partial \dot{z}_2} = c\dot{z}_2 \quad (2.46)$$

Writing the Lagrangians in the form

$$\sum_{s=1}^4 \frac{d}{dt} \left(\frac{\partial T_s}{\partial \dot{q}_i} \right) - \sum_{s=1}^4 \frac{\partial T_s}{\partial q_i} + \frac{\partial V}{\partial q_i} + \frac{\partial F}{\partial \dot{q}_i} = Q_i \quad (2.47)$$

the equations become

$$\{ m_1 \dot{d}_1^2 + m_2 \dot{d}_2^2 + m_3 \dot{d}_3^2 + m_4 [y_2^2 + (z_2 + d_4)^2] + I_T + I_{x_2} \ddot{\alpha} \} + \dot{\theta} \dot{\phi}'' (-I_{x_2} + I_{v_2} - I_{z_2}) - \dot{\theta} \dot{\phi}' + m_4 x_2 y_2 \ddot{\theta} = -E(s) + m_4 [\ddot{z}_2 y_2 - \ddot{y}_2 (z_2 + d_4)] - T_N \dot{\phi}'' \quad (2.48)$$

$$\{ m_1 \dot{d}_1^2 + m_2 \dot{d}_2^2 + m_3 \dot{d}_3^2 + m_4 [x_2^2 + (z_2 + d_4)^2] + I_T + I_{v_2} + I_{v_3} \} \ddot{\theta} + \dot{\phi}'' \dot{\epsilon} (I_{x_3} + I_{v_3} - I_{z_3}) + \dot{\phi}'' \dot{\alpha} (-I_{x_2} + I_{v_2} + I_{z_2} - I_{x_3} + I_{v_3} + I_{z_3}) + I_S \dot{\alpha} \dot{\phi}' + m_4 x_2 y_2 \ddot{\alpha} = T_N - E(s) \dot{\phi}'' + m_4 [\ddot{z}_2 x_2 - \ddot{x}_2 (z_2 + d_4)] \quad (2.49)$$

$$(I_{z_2} + I_{z_3}) \ddot{\phi}'' + \dot{\theta} \dot{\epsilon} (-I_{x_3} + I_{v_3} - I_{z_3}) + \dot{\theta} \dot{\alpha} (I_{x_2} - I_{v_2} - I_{z_2} + I_{x_3} - I_{v_3} - I_{z_3}) = A(s) \quad (2.50)$$

$$I_S \ddot{\phi}' - I_S (\dot{\alpha} \dot{\theta}) = -A(s) \quad (2.51)$$

$$-[I_{x_3} (\ddot{\alpha} - \ddot{\epsilon} - \dot{\theta} \dot{\phi}'') + (I_{v_3} - I_{z_3}) \dot{\theta} \dot{\phi}''] = E(s) \quad (2.52)$$

$$m_4 \ddot{x}_2 + c\dot{x}_2 + kx_2 = -m_4 (z_2 + d_4) \ddot{\theta} \quad (2.53)$$

$$m_4 \ddot{y}_2 + c\dot{y}_2 + ky_2 = -m_4 (z_2 + d_4) \ddot{\alpha} \quad (2.54)$$

$$m_4 \ddot{z}_2 + c\dot{z}_2 + kz_2 = m_4 (x_2 \ddot{\theta} + y_2 \ddot{\alpha}) \quad (2.55)$$

where

$$E(s) = -K_E \frac{(\tau_{E1} p + 1)}{(\tau_{E2} p + 1)} (\epsilon - \alpha) - K_{dE} (\dot{\epsilon} - \dot{\alpha}) - T_{FE} \quad (2.56)$$

$$A(s) = -K_A \frac{(\tau_{A1} p + 1)}{(\tau_{A2} p + 1)} (\phi'') - K_{dA} (\phi'' - \phi') - T_{FA} \quad (2.57)$$

$$T_N = F_G d_G \quad (2.58)$$

and

K_E, K_A = elevation and azimuth servo gain, ft-lb/rad

τ_{E1}, τ_{A1} = elevation and azimuth servo lead time constant, sec

τ_{E2}, τ_{A2} = elevation and azimuth servo lag time constant, sec

K_{dE}, K_{dA} = torque motor friction, lb-ft/rad/sec

T_{FE}, T_{FA} = friction torques, due to bearings, slip-rings and motor brushes, lb-ft

T_N = precession torque, lb-ft

F_G = precession gas thrust, lb

d_G = distance from spacecraft c.g. to line of action of precession gas thrust, ft

* If the damper is considered to have only two degrees of freedom, that x_2 , y_2 and z_2 are small compared to d_4 , and that the upper structure azimuth angular velocity and pointed instrument angular velocity are small, Equations (2.48) through (2.58) simplify to:

$$\left. \begin{aligned} (A+I_T+I_{x_2})\ddot{\alpha}-\dot{\theta}\dot{\phi}'I_s+m_4x_2y_2\ddot{\theta} &= -E(s)-m_4\dot{y}_2\dot{d}_4-T_N\phi'' \\ (A+I_T+I_{y_2}+I_{y_3})\ddot{\theta}+\dot{\alpha}\dot{\phi}'I_s+m_4x_2y_2\ddot{\alpha} &= T_N-E(s)\phi''-m_4\dot{x}_2\dot{d}_4 \\ (I_{x_2}+I_{x_3})\ddot{\phi}'' &= A(s) \\ I_s\ddot{\phi}' &= -A(s) \\ (\ddot{\epsilon}-\ddot{\alpha})I_{x_3} &= E(s) \\ \ddot{x}_2+2n\dot{x}_2+p'^2x_2 &= -d_4\ddot{\theta} \\ \ddot{y}_2+2n\dot{y}_2+p'^2y_2 &= -d_4\ddot{\alpha} \end{aligned} \right\} \quad (2.59)$$

where

$$\left. \begin{aligned} 2n &= \frac{c}{m_4} \\ p'^2 &= \frac{k}{m_4} \\ A &= m_1\dot{d}_1^2 + m_2\dot{d}_2^2 + m_3\dot{d}_3^2 + m_4\dot{d}_4^2 \end{aligned} \right\} \quad (2.60)$$

Equations (2.59) were solved on the Beech Aircraft CRC analog computer using the following initial conditions:

$I_s = 20.74$ slug-ft ²	$d_3 = 0.64$ ft
$I_T = 15.75$ slug-ft ²	$d_4 = 1.83$ ft
$I_{x_2} = 1.0$ slug-ft ²	$F_G = 0.05$ lb
$I_{y_2} = 1.0$ slug-ft ²	$d_G = 1.92$ ft
$I_{z_2} = 0.65$ slug-ft ²	$K_{d_e} = K_{d_A} = 0.015$ lb-ft/
$I_{x_3} = 1.3$ slug-ft ²	rad/sec
$I_{y_3} = 0.9$ slug-ft ²	$T_{F_B} = T_{F_A} = 0.063$ lb-ft
$I_{z_3} = 1.0$ slug-ft ²	$\dot{\phi}' = \text{rad/sec}$
$m_1 = 11.8$ slugs	$p' = 4.136$ rad/sec
$m_2 = 1.65$ slugs	$c = 2 \times 10^{-3}$ lb-sec/
$m_3 = 1.86$ slugs	in.
$m_4 = 0.031$ slug	$2n = 0.766$ rad/sec
$K_A = 400$ ft-lb/rad	$\tau_{A1} = E_1 = 0.30$ sec
$K_B = 250$ ft-lb/rad	$\tau_{A1} = E_2 = 0.017$ sec
$d_1 = -0.28$ ft	
$d_2 = 0.36$ ft	

Results of Computer Study

Time history traces of the computer study were obtained on a six-channel Offner recorder.

The outputs recorded were the two servo pointing errors and the angular velocities of the spacecraft and nutation damper. (Velocities rather than angular displacements were recorded because of the scaling problems that would have arisen in plotting displacements during long periods of torquing.)

Results of the study showed that under various input conditions the azimuth and elevation servos kept the experiment package pointed at the center of the sun to within 1.0 arc-minute of error. The major error sources were the functional and motor damping errors in the case of the azimuth servo and frictional error in the case of the elevation servo.

The azimuth servo error was very slightly modulated by the nutation, but this modulation was quickly reduced to zero by the nutation damper. This phenomenon occurred immediately after the precession torque was applied and after the precession torque was removed. The explanation for the recurrence of the nutation at precession torque removal is given on page 36.

It was observed that the damping constant of the nutation damper could be lowered to a value which gave the best overall damping of the system. Further investigation indicated that the damper caused no appreciable destabilizing effects on the complete system through second-order cross-coupling terms. The damping time constant for the system was found to be minimum when the natural frequency of the nutation damper was approximately equal to the open-loop nutation frequency.

NUTATION DAMPER DEVELOPMENT

General Description of the Problem

The basic problem was the need for a damping mechanism which would effectively damp out nutational motion of the Orbiting Solar Observatory. The need for the damper was derived from the basic design concept of the OSO I, i.e., to monitor solar spectral emission phenomena.

A schematic drawing of the spacecraft (Figure 2-2) shows the angular relationships used to describe the position of the spacecraft with respect to the fixed XYZ coordinate system. In the figure, the spacecraft attitude is in error by the angle ϵ . S is the spin axis of the spacecraft and is normal to N , the line of nodes. Jet thrusts produce torques of either sense about the line of nodes, precessing the trihedron, O-M-N-S, so as to vary the elevation angle, α . The zero error condition occurs when the line of sight, or solar vector, is coincident with the line of nodes. With the choice of coordinate directions shown in the figure, the jet control system precesses the spacecraft in the XY-plane and the roll angle, θ , is always very nearly $\pi/2$.

In the absence of damping mechanisms, the equations of motion of the spacecraft during torquing are as follows:

$$\left. \begin{aligned} \alpha &= \frac{M_n}{I\omega} t - \frac{M_n}{I\omega p} \sin pt \\ \theta &= \frac{\pi}{2} + \frac{M_n}{I\omega p} (1 - \cos pt) \end{aligned} \right\} \quad (2.61)$$

where

M_n = moment about line of nodes, lb-in.

$I\omega$ = axial moment of inertia of rotating body, lb.-in.-sec²

ω = spin rate, rad/sec

t = time, sec

$p = I\omega/I_1$, rad/sec

I = total transverse moment of inertia of spacecraft, lb-in.-sec²

These equations are valid provided α and θ are small in comparison to the spin rate ω and provided the initial values of α and θ are taken to be zero.

From Equation (2.61), it can be seen that a torque M_n about the line of nodes causes pseudo-regular precession of the spacecraft. The spin axis S moves essentially in the XY-plane with α increasing at the mean rate $\dot{\alpha} = M_n/I\omega$. Superimposed on this average motion is the nutational motion indicated by the sinusoidal terms in Equation (2.61). If the torque, M_n , is instantly stopped, the average motion stops but the nutation continues and becomes, in fact, free body precession. To a body mounted on the non-rotating instrument platform, the residual nutational motion is a rotary translation about the spin axis of angular amplitude, $M_n/I\omega p$, and frequency, p .

In deriving Equation (2.61) it was assumed that an external moment, M_n , about the line of nodes was torquing the spacecraft. Such a moment could be produced by the gas jets, aerodynamic drag, or the divergence of the earth's gravitational field. Furthermore, reaction of the elevation servo motor on the instrument platform during initial instrument acquisition produced a moment, M_m , which precessed the spacecraft in the roll plane. Other external torques could also have had components capable of causing precession in this plane.

Unless a nutation damper was provided to put the spacecraft to sleep after each external torque application, the result of successive applications would have been to increase or decrease the amplitude of nutation depending upon the direction, magnitude, and time of application of a given torque. It was necessary

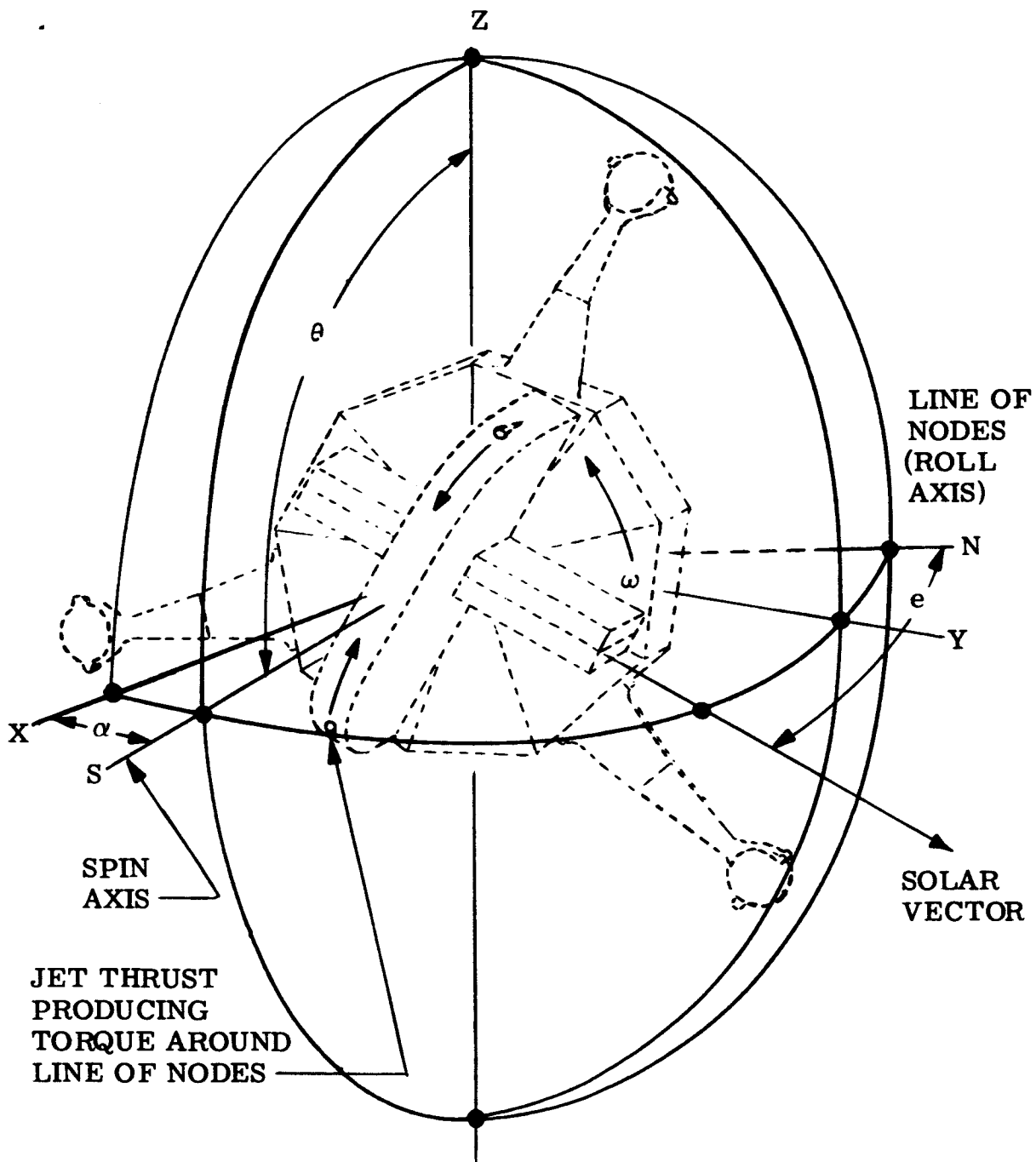


FIGURE 2-2.—Reference coordinate system for nutation damper study.

to eliminate such random motion in order to assure spacecraft stability.

Design Requirements

The general design requirements that were established for the nutation damper are as follows:

1. *Damping rate.*—Compatible with a time constant of ten (10) seconds for the exponential decay of the nutation amplitude.
2. *Weight of damper.*—Minimum weight for established damping rate.
3. *Size of damper.*—Compact as possible and compatible with overall spacecraft geometry.
4. *Nutation frequency.*—Dependent upon principal moments of inertia and spin rate of spacecraft. Calculated frequency of 4.71 rad/sec was based on the following values:

$$\begin{aligned}\omega &= 0.5 \text{ rps} \\ I &= 30 \text{ slug-ft}^2 \\ I_1 &= 20 \text{ slug-ft}^2\end{aligned}$$

5. *Maximum anticipated external torque.*—1 lb-ft (reaction of elevation servo motor on instrument platform during initial acquisition of instrument).

6. *Angular amplitude of nutation due to maximum torque.*—Calculated value of 7.4 minutes of arc, based on spacecraft parameter values above.

7. *Minimum allowable residual nutation amplitude.*—Less than 1 arc-minute (compatible with accuracy of fine elevation pointing control).

8. *Temperature range.*—0° to 50° C.

9. *Test requirements.*—The ground operational test requirement for the nutation damper was that its in-flight performance be conclusively demonstrated in the laboratory. Appropriate test apparatus had to be devised to meet this requirement.

The presence of gravitational acceleration during ground testing in contrast to the absence of acceleration in orbital flight contributed significantly to the complexity of the problem. The behavior of many types of dampers is altered by the effect of gravity, often to the extent that a device that performs well in one environment will not function in the other. Furthermore, a complete mathematical analysis

of the behavior of the damper selected for use was imperative in order to evaluate accurately the effect of gravity on its performance. Simplicity of and confidence in the ground testing operation thus became strong design considerations.

General Approach to the Solution

Various Types of Dampers Considered

In the process of finding a damper that met all of the specified requirements, many different types of damping devices were considered. Most of those given serious consideration were passive dampers; that is, dampers driven by the nutational motion they eventually damp out. Since the Orbiting Solar Observatory spun about its axis of greatest moment of inertia, a passive damper could be used to absorb the nutation energy without adversely affecting the spin. A passive damper was preferred over an active one for this application because of its reliability and simplicity of operation, and because it did not require an outside power source.

The three general types of dampers investigated were pendulum dampers, inertial dashpot dampers, and a torus partially filled with mercury. Simple, spherical, and torsional pendulums, and a ball rolling in a curved tube were considered. Practicality of mounting the various type dampers on either the rotating or nonrotating sections of the spacecraft was also taken into consideration. Some of the damping schemes included among the various combinations of damper type and position were easily seen to be impractical. The investigation of each of the other combinations continued until the damper could be ruled out or until a theoretical evaluation of its performance could be made.

In the following discussion, the results of the analysis of each type of damper concluded to be feasible for spacecraft use are summarized. For those types not chosen for this particular application the reason is given for not selecting the damper.

Mercury Ring Damper—A hollow toroidal ring partially filled with mercury or some other more viscous liquid, has been proven to be an effective nutation damper. A theoretical analy-

sis of the mercury ring damper has been published by Carrier and Miles.*

This type of damper could not be used on the Orbiting Solar Observatory because of the particular spacecraft launch program. The third stage of the Thor-Delta vehicle which boosted the OSO spacecraft into orbit did not spin up until shortly before the third-stage rocket was fired. At the time of spin-up, the mercury in the partially filled torus would not, in general, have been equally distributed radially. Since no reactive force was present to redistribute the liquid after spin-up, the static unbalance would have persisted and the spacecraft would not have spun about its axis of symmetry. The resulting motion of the figure axis would have had the same undesirable effect on the pointing control system as would nutation of the same amplitude.

Inertial Dashpot Dampers—This name is used to describe the types of dampers shown schematically in Figures 2-3 and 2-4. The damper in Figure 2-3 has one degree of freedom and the one in Figure 2-4 has two degrees of freedom. The principle of operation is essentially the same for both. When either type damper is mounted on the spacecraft so that nutational motion displaces the damper case in the direction shown in the figures, nutation energy is transferred to the inertial mass through the springs from which it is suspended. If the damper is filled with a viscous fluid, this energy is dissipated at a rate proportional to the relative velocity of the inertial mass with respect to the case.

A simplified analysis of this type damper was made assuming that the damper was being driven sinusoidally in the one or two planes of freedom at a constant frequency and amplitude. The results of this analysis are summarized as follows:

1. The energy absorption rate of the damper with two degrees of freedom was twice that of the other type for a given inertial mass.
2. Maximum energy absorption occurred when the undamped natural frequency of vibration of the damper was equal to the frequency of

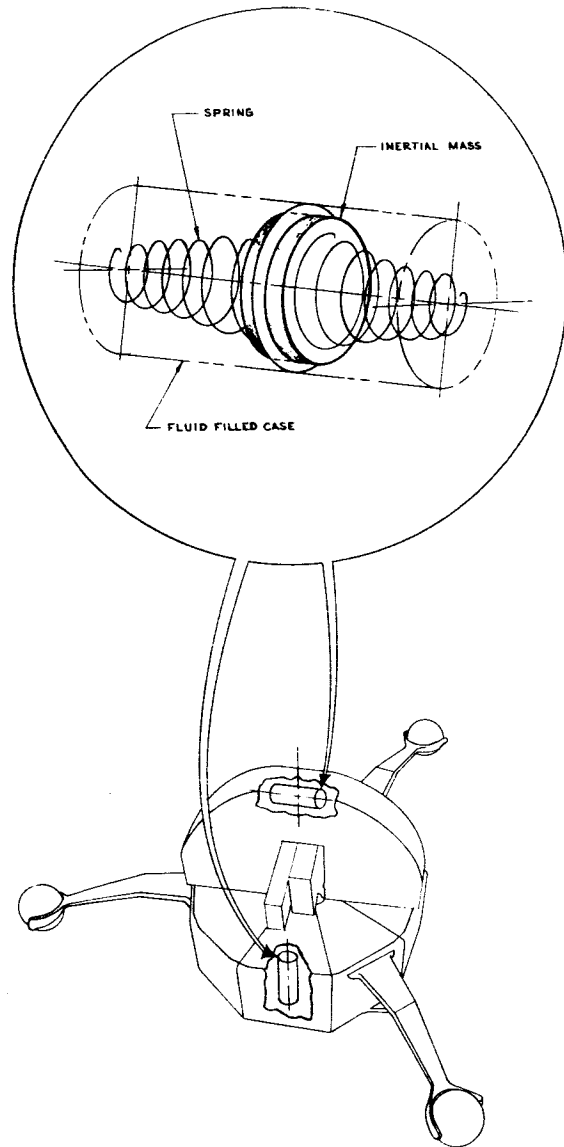


FIGURE 2-3.—Inertial dashpot damper—one degree of freedom.

the forcing function. This condition is commonly known as "resonance".

3. At the assumed nutation frequency and at resonance, a weight of approximately 1 lb for the inertial mass in the damper with two degrees of freedom resulted in an energy absorption rate equal to one-tenth the kinetic energy of spacecraft nutation. Since both the energy absorption rate of the damper and the kinetic energy of nutation are proportional to the

* G. F. Carrier and J. W. Miles, *On The Annual Damper for a Freely Precessing Gyroscope*, S.T.L. Report No. EM9-3, 29 January 1959.

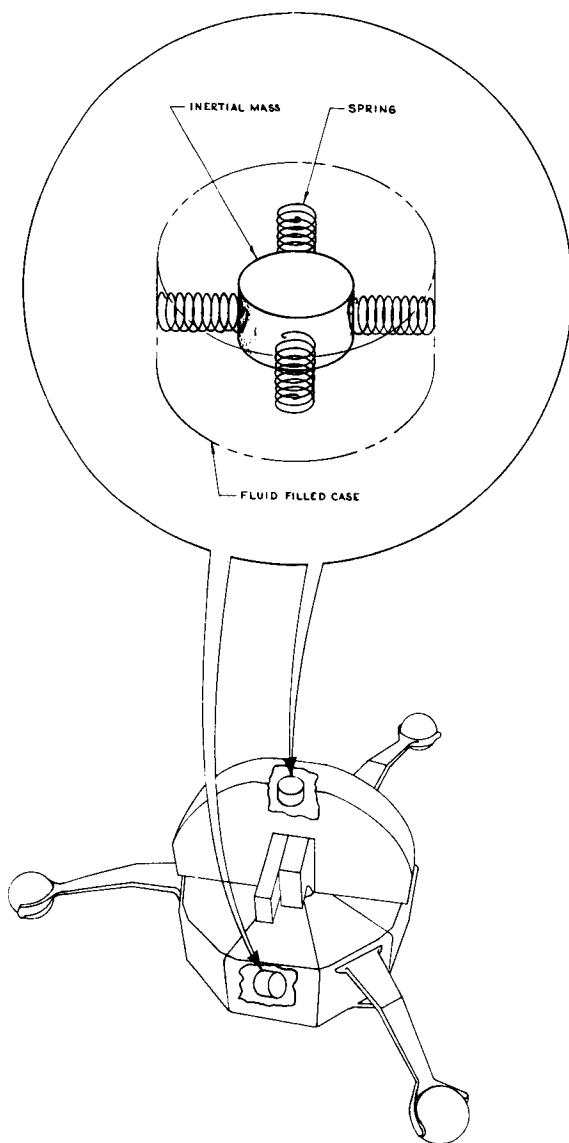


FIGURE 2-4.—Inertial dashpot damper—two degrees of freedom.

square of nutation amplitude, this value corresponded to a time constant of 10 seconds for the exponential decay of nutation amplitude of the spacecraft-damper system.

These results were identical to those obtained the simple and spherical pendulum dampers using the steady forced vibration analysis. Such a conclusion was anticipated since within the accuracy of small angle approximations, the analyses of the two types of dampers with

the same degree of freedom were basically the same.

The dashpot damper was given serious consideration because, unlike the pendulum damper its natural frequency of vibration is not directly affected by gravity. It was not used on OSO I primarily because no practical means of testing its operation in the laboratory could be devised. In the gravity field, the relatively weak springs required to yield the desired linear spring rate of approximately 0.2 lb/in. could not support the weight of the inertial mass, either vertically or horizontally, without preloading the springs beyond their linear range.

Methods for artificially supporting the weight during ground test either altered the design of the damper substantially or introduced friction. Both of these situations were undesirable because they complicated or rendered virtually impossible the task of correlating ground test results with anticipated in-flight performance.

Torsional Pendulum Damper—The torsional pendulum damper, illustrated in Figure 2-5, was investigated because its operation is unaffected by gravitational acceleration. The wheel-like inertial mass is supported at its center of gravity by a torsional spring. During ground tests the weight of the wheel had to be supported by the torsional spring; however, it was found that the wire required to obtain the desired spring constant in this instance could have been preloaded sufficiently in tension to support the weight. For lower spring rates the outer rim of the inertial wheel can be made hollow to partially float the wheel in the viscous fluid that fills the damper.

The simplified analysis of torsional damper performance indicated that damping efficiency was much less than for the simple or spherical pendulum types. The expression for the energy absorption rate of the torsional pendulum device at resonance is of the same general form as that of the simple pendulum. However, the linear amplitude is proportional to the radius of the wheel for the torsional pendulum, but is proportional to the distance from the center of gravity of the spacecraft to the damper for the simple pendulum. This latter distance was about two feet, whereas for a torsional pendulum, damper with an inertial mass of one

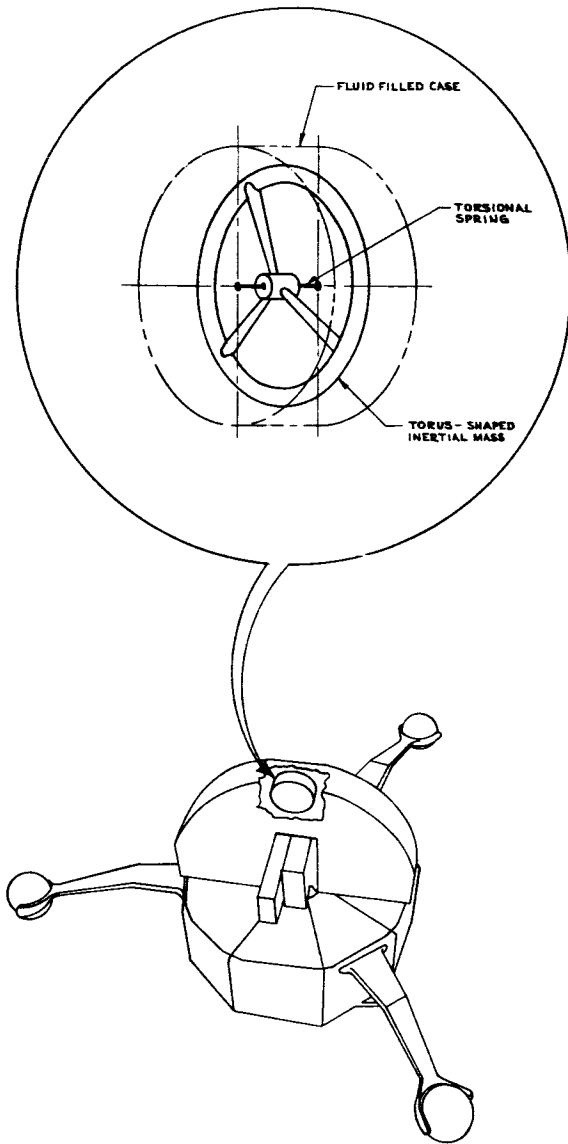


FIGURE 2-5.—Torsional pendulum damper.

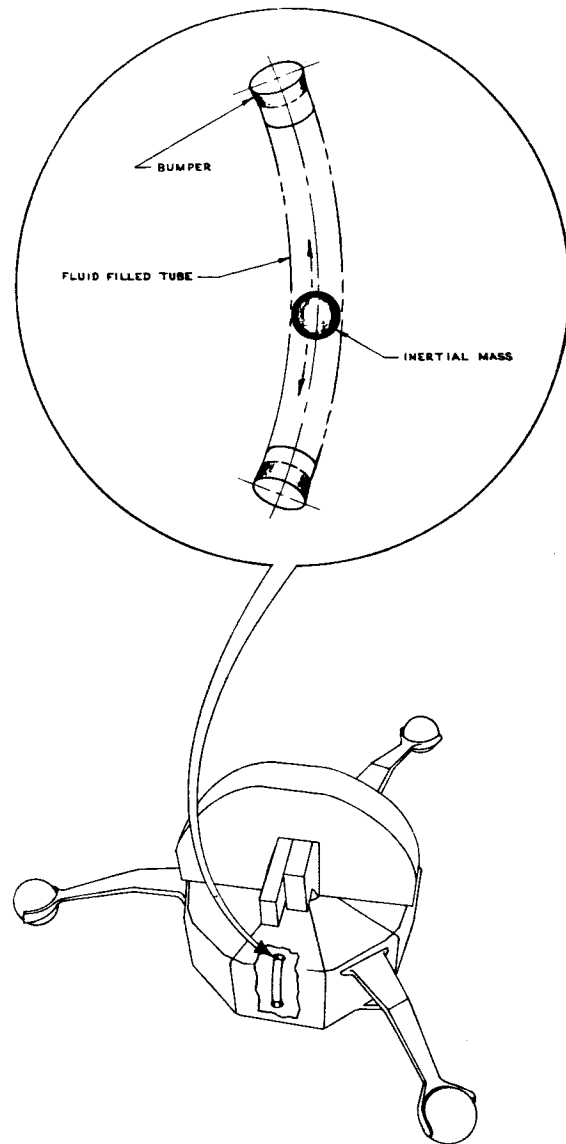


FIGURE 2-6.—Rolling ball damper.

pound, a typical value for the wheel radius is three to four inches. Thus, the simple pendulum is six to eight times more efficient, in terms of damper weight for a given damping rate, than the torsional pendulum.

Rolling Ball Damper—A ball free to roll in a closed curved tube can be used to damp out nutation if mounted on the rotating body as shown in Figure 2-6. The tube may be filled with a viscous fluid to combine viscous damping with the damping caused by the rolling

friction of the ball, or shock-absorbing "bumpers" may be placed at either end of the tube to absorb energy.

This type of damper exhibits one distinct advantage over any of the types mounted on the nonrotating body. The frequency of vibration of the damper, like the nutation frequency, is directly proportional to the spin rate of the spacecraft. Therefore, the damper remains *tuned* or in resonance with the disturbing frequency for any given spin rate.

The principal disadvantage of this or any other type of damper mounted on the rotating body was that the frequency of the disturbance it experienced was equal to the difference between spin frequency and nutation frequency. This disturbing frequency was always less than nutation frequency for a disk-shaped rotating body spinning about its axis of greatest moment of inertia. For this spacecraft, it was about one-third of the frequency of nutation.

The undesirable effect of a lower frequency disturbance can be seen by examining the energy absorption equation derived for the various dampers. In each case, the absorption rate at resonance is proportional to the fourth power of the disturbing frequency. To maintain a given damping rate as the disturbing frequency is lowered, either the inertial mass of the damper must be increased or the damping constant decreased. Decreasing the damping constant results in an increase in the required amplitude of vibration of the inertial mass. Within the space limitations of the rotating section of the OSO I, this type of damper would have been difficult to design to satisfy both the damping rate requirement and minimum weight requirement.

Other problems associated with this type damper were:

1. Because of static friction, the damper would not have completely damped out nutation and could have caused dynamic unbalance of the rotating body.

2. The effect of gravity on the behavior of the damper during ground testing was significant for the relatively low spin rate specified for the OSO I. The ground checkout of the nutation damper in spacecraft simulation tests would have been difficult.

Simple Pendulum Damper—The simple pendulum and spherical pendulum dampers are analogous to the two types of inertial dashpot dampers discussed earlier: the simple pendulum is restricted to one degree of freedom, whereas the spherical pendulum has two degrees of freedom. Otherwise, the principle of operation and the analysis of the behavior of the two types is essentially the same. The simple pendulum damper was not chosen because it was only one-half as efficient in damping out nutational

motion as the spherical pendulum damper.

Spherical Pendulum Damper—Selection of the spherical pendulum damper was based on the following considerations:

1. Damping efficiency of this damper was concluded to be equal to or better than any of the other feasible types.

2. It was possible to make a theoretical analysis of its behavior that could be conclusively substantiated in simple experimental tests.

3. The effect of gravity on its behavior could be evaluated accurately.

4. A modified version of the damper could be easily incorporated into the spacecraft ground simulation tests on the air-bearing fixture.

The possibility of mounting this type damper on either the rotating or nonrotating sections of the spacecraft was investigated. Valid arguments for choosing either location could have been made. The decision to mount it on the nonrotating body was based on the following conclusions:

1. On the rotating body, a pendulum damper would have to be mounted in an inverted position in the centrifugal force field in order to tune the damper to the relatively low frequency disturbance it would experience. The load on the wire supporting the bob of an inverted pendulum becomes critical as the natural frequency of the pendulum approaches zero. The tuned frequency required in this application was sufficiently low that a slight increase in spin rate above nominal value resulted in buckling of the support wire. The resulting restriction on spin rate deviation was considered too severe.

2. Since the frequency of the disturbance experienced by the damper mounted on the rotating body was much lower than nutation frequency, the damping efficiency was also lower than at the alternate location, assuming fixed values for mass and damping constant. Reduction of the damping constant to compensate for the lower frequency value was not feasible in this case. The resulting amplitude of vibration of the pendulum bob was large enough that the support wire would have failed in bending at maximum displacement.

3. Ground testing was greatly simplified by selecting the location on the nonrotating body.

Analysis of the Spherical Pendulum Damper

There were certain disadvantages to mounting the nutation damper on the nonrotating body, e.g., the damper did not function when the spacecraft was in the shadow of the earth. During that time, the servo motors were inoperative and the nonrotating body gradually acquired angular momentum from the rotating section until both bodies were spinning at the same rate. When the pendulum damper was rotated, the normal static equilibrium position of the bob was no longer a stable point, and the bob was moved by centrifugal force to a position of lower potential against the damper case. It maintained this fixed position relative to the case for as long as rotation continued. In assuming the new position, the bob created a small dynamic unbalance in the spacecraft. Both the loss of the damper and the small amount of wobble caused by the dynamic unbalance could be tolerated since both conditions were remedied shortly after the spacecraft emerged into sunlight again. Another undesirable feature was that the natural undamped frequency of vibration was fixed. Since the nutation frequency varied directly as spin rate, the damper would not have remained sharply tuned to nutation frequency as the spin rate varied between allowable limits. However, the damper was designed to be rather efficient in damping out nutations with frequencies 10 percent above or below the nominal value. Disturbances of these maximum or minimum allowable frequencies were damped out with a time constant of about 20 seconds, or twice the optimum value.

Because of the complexity in operation of the spacecraft-damper rigid-body system, a complete mathematical solution of the rigid-body problem was not attempted initially. Even when small angle approximations are used, the differential equations which describe the motion of the system are nonlinear and form a set of seven simultaneous second order equations. Analytical solution of these equations is not possible. Numerical solutions are possible using an analog computer, but such solutions require numerical values for all of the parameters of the system. A para-

metric study of the system was considered to be impractical until some knowledge could be gained concerning the general behavior of the nutation damper and the servo mechanisms connecting the three rigid bodies of the spacecraft.

The more practical approach adopted was first to analyze the behavior of the damper independent of system operation. This step was particularly important in simplifying the selection of the most efficient damper of many types considered. The analysis of each of the dampers under steady forced vibration was used for this purpose. The approximate values for damper parameters obtained from this analysis were then used to design models that could be checked out experimentally on ground test fixtures which simulate the operation of the damper-spacecraft system during flight. Using the Optimum values for damper parameters gained from the simplified analyses and accompanying experimental work, numerical solutions of the equations of motion of the complete rigid-body system were obtained.

Steady Forced Vibration Analysis—From this simplified analysis, the equations of motion of the damper under sinusoidal excitation of constant amplitude and frequency were obtained. An expression was then derived which gave, in terms of damper parameters, the rate at which the damper was capable of dissipating energy. When the required energy dissipation rate was substituted into this expression, approximate values for the damper parameters could be determined. Of particular interest was the weight of the vibrating mass of the damper since damping efficiency is directly proportional to this weight.

The steady forced vibration analysis could not be used to determine whether the torque applied to the spacecraft by the damper was in the proper direction to damp out nutation. This determination was made in each case by direct analysis of the vector diagram of the rigid-body system.

Another deficiency of this analysis was that it did not take into account the effect of the damper in decreasing nutation amplitude as kinetic energy was taken out of the system. Since (in this approach) constant amplitude

of the forcing function was assumed, a constant energy dissipation rate was obtained. Actually, both of these quantities constantly decrease as nutation is damped out. It was reasoned that a more accurate description of the motion of the damper-spacecraft system during nutation could be obtained from the analysis of the behavior of a two-body system during free vibration.

Two-Body Free Vibration Analysis—The two-body system used in this analysis was the spherical pendulum damper mounted on the pendulum platform test fixture. The problem was simplified by considering only single-plane motion of the system. Use of the damper-test fixture system in single-plane vibration as a mathematical model of the damper-spacecraft system during nutational motion greatly simplified the analysis of the problem.

Despite the simplifying assumptions, this approach was still considerably more complex than the steady forced vibration analysis. The differential equations of motion of the system were easily written but an analytical solution in terms of system parameters was not possible. Because of the lengthy and tedious process involved in obtaining numerical solutions by hand, an electronic analog computer was used. The parametric solution of the equations, obtained with the aid of the computer, yielded optimum values for damper parameters. Results of this study were substantiated by experimental tests of the damper on the pendulum platform test fixture.

Analysis of Complete Rigid-Body System—This analysis was performed to verify the results of the simplified analyses and preliminary experimental tests. The equations of motion of the complete nutation damper-spacecraft rigid-body system were derived and solved numerically on an analog computer. These results were compared with those obtained from the simplified two-body free vibration analysis. A sufficient number of computer solutions for the complete system was obtained not only to confirm the fact that the damper would damp out nutational motion of the spacecraft but also that the results of the simplified analyses and the tests performed on the pendulum platform were correct. The

complete analysis was also of importance in evaluating dynamic stability of the servo-connected spacecraft rigid-body system and in gaining a more complete understanding of the gyroscopic motion of such a complex system.

As previously mentioned, the equations of motion of the system formed a set of seven simultaneous, second order, nonlinear differential equations. These resulted from the seven degrees of freedom of the complete system. The nutation damper had two degrees of freedom and the spacecraft five (three for the rotating body, an additional freedom in yaw for the nonrotating section, and a further one in pitch for the pointed solar experiments).

Since the three rigid bodies making up the spacecraft were connected by the azimuth and elevation servo motors, the dynamic response of these servos had to be taken into account when analyzing the dynamic behavior of the spacecraft. For this reason, the equations of motion of the spacecraft without the damper were derived first and solved on an analog computer. This analysis was essentially a parametric study of servo characteristics and was used to augment breadboard tests of the attitude control system. The nutation damper was incorporated into the system only after satisfactory response of the servo system had been established.

Design Procedure

In the following discussion the pertinent results of the mathematical analysis of the problem are presented.

Steady Forced Vibration Analysis

For the spherical pendulum damper, the equation of relative motion of the pendulum bob with respect to the case has the general form:

$$x = A'' \sin (pt - \alpha) \quad (2.62)$$

in the plane in which the forcing function is $A' \sin pt$, and

$$y = A'' \cos (pt - \alpha) \quad (2.63)$$

in the plane in which the forcing function is

$A' \cos pt$. In Equations (2.62) and (2.63), the following notation is used:

x and y = linear displacement of the bob
 A'' = relative linear amplitude of vibration of the bob
 α = phase lag angle of the relative motion with respect to the forcing function

and

$p' = k/m$ — natural undamped frequency of vibration of pendulum

$2n = c/m$

k = spring constant of pendulum arm

m = mass of bob

c = damping constant of fluid

$$\alpha = \tan^{-1} \frac{2np}{p'^2 - p^2} \quad (2.64)$$

The relative amplitude of the bob is given by the expression:

$$A'' = A' \frac{p^2/p'^2}{\sqrt{(1 - p^2/p'^2)^2 + 4n^2 p^2/p'^4}} \quad (2.65)$$

The energy dissipation rate of the damper during steady-state vibration is determined to be:

$$-\dot{E} = A'^2 \frac{cp^6/p'^4}{(1 - p^2/p'^2)^2 + 4n^2 p^2/p'^4} \quad (2.66)$$

For the small values of the damping constant, c , the maximum dissipation rate occurs at resonance, when $p/p' = 1$, and Equation (2.66) reduces to:

$$-\dot{E} = A'^2 \frac{cp^4}{4n^2} = A'^2 \frac{m^2 p^4}{c} \quad (2.67)$$

In order to determine the damping rate corresponding to a given energy dissipation rate, it is necessary to utilize the following expressions for the spacecraft:

$$\text{Nutation frequency, } p = I\omega/I_1 \quad (2.68)$$

$$\text{Nutation amplitude, } A' = Ar - Mr/I\omega p \quad (2.69)$$

Kinetic energy of spacecraft due to nutation,

$$\Delta KE = M^2 I_1 / 2I^2 \omega^2 = \frac{1}{2} I_1 p^2 A'^2 = A'^2 \frac{I_1 p^2}{2r^2} \quad (2.70)$$

In Equations (2.68) through (2.70) the notation is as follows:

A' = linear amplitude of nutation at the damper

A = angular amplitude of nutation

r = distance from center of gravity of spacecraft to damper

I = axial moment of inertia of rotating section of spacecraft

I_1 = transverse moment of inertia of entire spacecraft

ω = spin rate

M = torquing moment causing nutation

From Equations (2.67) and (2.70) it is obvious that both the kinetic energy of nutation and the ability of the damper to dissipate that energy are directly proportional to the square of the amplitude of nutation. Therefore, the time constant for the exponential decay of nutation amplitude is determined from the expression:

$$T = \Delta KE / (-\dot{E}) \quad (2.71)$$

For an assumed time constant of ten seconds, we have:

$$-\dot{E} = 0.1 \Delta KE \quad (2.72)$$

Substituting Equations (2.67) and (2.70) into Equation (2.72) yields the expression:

$$\frac{m^2 p^4}{c} = \frac{(0.1) I_1 p^2}{2r^2}$$

or

$$\frac{m^2}{c} = \frac{(0.05) I_1}{r^2 p^2} \quad (2.73)$$

For a given spacecraft configuration and damper position, the right side of Equation (2.73) is constant. Therefore, the required mass of the pendulum bob is directly proportional to the square root of the damping constant. A practical lower limit for the value of c is imposed by the physical dimensions of the damper.

From Equation (2.74) it is determined that at resonance the ratio of forced relative amplitude to forcing amplitude or magnification factor is:

$$\beta = \frac{A''}{A'} = \frac{mp}{c} \quad (2.74)$$

Since case and fluid weight increase but bob weight decreases with decreasing damping constant, a parametric study is required to determine minimum damper weight. Instead of conducting such a study, reasonable values for mass, damping constant, and damper geometry were arbitrarily selected in designing a test model. Later tests and analysis showed that an optimum value for the damping constant, independent of mass, existed in the actual application.

The following values for spacecraft parameters were assumed, based on the best available design data at that time:

$$\begin{aligned} I &= 30 \text{ slug-ft}^2 \\ I_1 &= 20 \text{ slug-ft}^2 \\ \omega &= 0.5 \text{ rp} = 3.14 \text{ rad/sec} \\ M_{\max} &= 1 \text{ lb-ft} \\ r &= 24 \text{ in.} \end{aligned}$$

Calculated values were:

$$\begin{aligned} p &= I\omega/I_1 = 4.71 \text{ rad/sec} \\ A &= M/I\omega p = 2.25 \times 10^{-3} \text{ rad} \\ A' &= Ar = 0.054 \text{ in.} \\ \Delta KE &= M^2 I_1 / 2 I^2 \omega^2 = 0.0135 \text{ lb-in.} \\ -\dot{E} &= 0.1 \Delta KE = 0.00135 \text{ lb in./sec} \end{aligned}$$

From Equation (2.67) we determine that:

$$c = \frac{A'^2 m^2 p^4}{-\dot{E}} = 1066.63 m^2 \quad (2.75)$$

For an assumed weight of 1 lb for the pendulum bob, $m = 2.59 \times 10^{-3} \text{ lb-sec}^2/\text{in.}$ and from Equation (2.75): $c = 7.16 \times 10^{-3} \text{ lb/in./sec.}$ Then from Equation (2.74), $\beta = mp/c = 1.7$.

The actual value of the damping constant for a particular damper configuration depends upon damper geometry and the physical properties of the fluid. Various fluid dynamics equations, based for the most part on idealized conditions, can be used to approximate this value in a given case; however, it was felt that a realistic value could be obtained only by experiment. From the approximations obtained from the theoretical equations, the value given above seemed easily attainable with commercially available silicone fluids.

As a result of this approximate analysis of the problem, a value of one pound was selected for the weight of the pendulum bob in the damper model that was built for later experimental tests. The next logical step was to determine the physical requirements of the pendulum arm to yield the desired natural frequency of 4.71 rad/sec for the damper.

Frequency Analysis for Damper

By using the energy method for determining the natural frequency of vibration of a cantilever beam with a concentrated end load, the following frequency equation is derived for the pendulum damper in the gravity field:

$$p = 34.4 \sqrt{\frac{EI + 0.405 W_e l^2}{W l^3 + 2.467 W l \rho_1^2}} \quad (2.76)$$

where

E = modulus of elasticity of spring wire
pendulum arm

$I = \frac{\pi d^4}{64}$ = moment of inertia of wire (d = wire diameter) about centerline

W = weight of pendulum bob

W_e = effective weight of bob in damping fluid

l = length of pendulum arm

ρ_1 = radius of gyration of pendulum bob about its transverse axes

and the double-signed term takes into account the effect of mounting the pendulum in either an upright or inverted position in the gravity field. The positive sign is used for the upright pendulum and the negative sign for the inverted one. In the gravity-free environment of orbital flight, the double-signed term drops out and Equation (2.76) takes the modified form:

$$p = 1.745 \sqrt{\frac{EI}{m l^3 + 2.467 m l \rho_1^2}} \quad (2.77)$$

where

m = mass of pendulum bob

By experimentally verifying Equation (2.76) for the two orientations in the gravity field, Equation (2.77) is indirectly substantiated. It was readily apparent from Equation (2.76) that

the desired natural frequency for the pendulum could not be attained in the upright position without unreasonable pendulum lengths. Therefore, the inverted position was assumed for all ground tests conducted to evaluate damper performance. Equation (2.76) was used to determine the diameter of a steel music wire required to yield the desired damper frequency. In this calculation the pendulum bob weight of one pound previously selected was used, and the value, $l=3$ in., was arbitrarily chosen.

For a spherical bob, $\rho_1^2 = \frac{2}{5}r^2$, and for a one-pound brass sphere, $r=0.92$ in.; then $\rho_1^2=0.339$. The modulus of elasticity for steel, E , is 3×10^7 lb/in². Therefore, to obtain the desired frequency, $p=4.70$ rad/sec; $I=1.40 \times 10^{-7} = \pi d^4/64$ and $d=0.041$ in. To obtain the desired frequency for the damper in orbital flight, Equation (2.77) is used to determine the correct wire diameter; $d=0.025$ in.

Damper Model Design

A preliminary model of the spherical pendulum damper was designed and built, using the values for damper parameters obtained from the steady forced vibration analysis. The dimensioned drawing of this model is presented in Figure 2-7.

The damper was designed for easy modification. The case could be easily disassembled, and the spring wire pendulum arm was screw-attached at either end. Thus the damping constant could be varied by changing fluid, and the frequency could be varied by adjusting pendulum arm length or by replacement with a wire of different diameter. The cylindrical wall of the case was made of clear plastic so that visual observation of pendulum bob tests could be made.

As discussed previously, the two fixed parameters for the model were: weight for bob of one

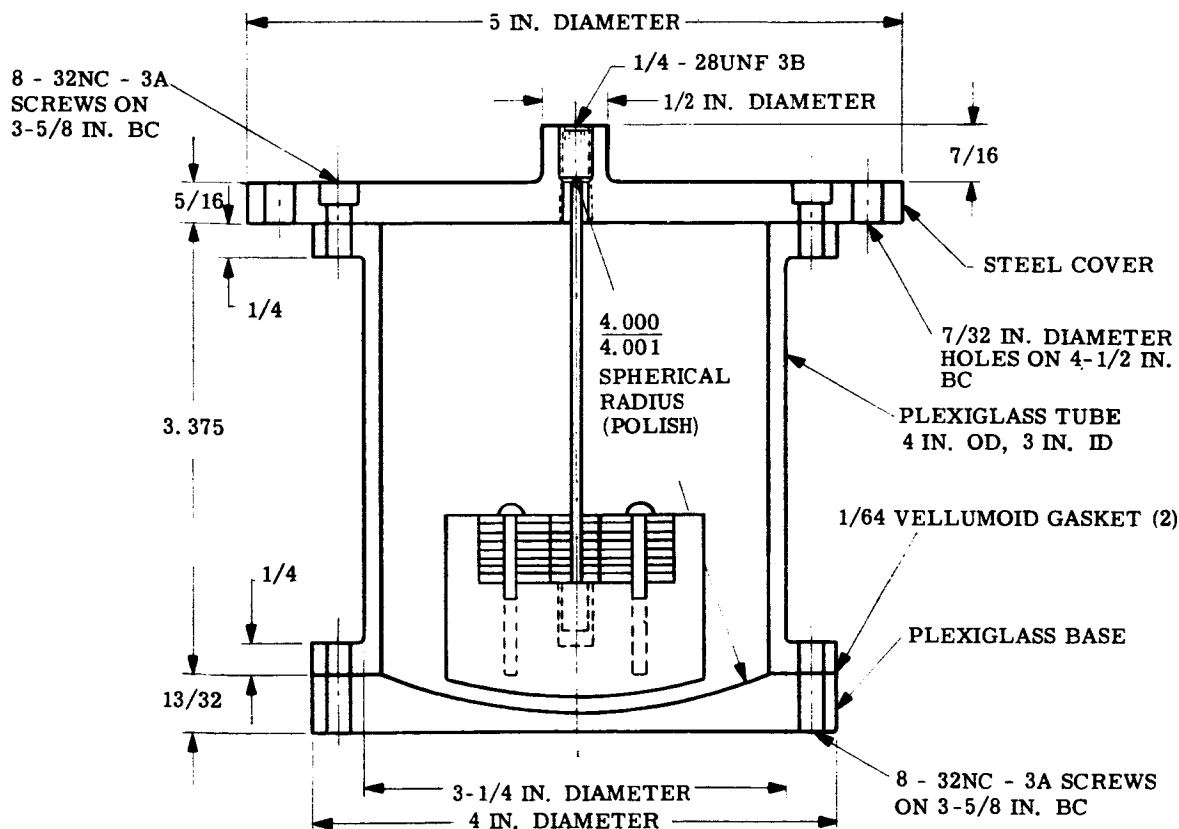


FIGURE 2-7.—Dimensional drawing of first damper test model.

pound and pendulum arm length of three inches. The initial bob was cylindrical in shape with removable washers for weight variation above or below the nominal value and with a convex spherical surface opposite the pendulum arm. The inside surface of the case nearest the bob was concave spherical with the same radius of curvature as the adjacent bob surface. A comparatively small clearance between these surfaces could thus be maintained during oscillatory motion of the bob. This feature was incorporated into the design at a time when it was felt that it would be necessary to utilize the viscous shear of the liquid between these surfaces in obtaining desired damping rates. It was later realized that sufficient damping could be obtained simply by the motion of the immersed body through the fluid. A spherical shape was then adopted to simplify the geometry.

The length of the damper case was chosen so that the clearance between the spherical surfaces in the original configuration could be adjusted from zero to $\frac{1}{4}$ inch or more. The inside diameter of the cylindrical case wall was sufficiently large to allow a relative linear amplitude of about $\frac{1}{2}$ inch for the bob. This corresponds to a magnification factor of five for a maximum forcing amplitude of 0.1 inch.

Two-Body Free Vibration Analysis

This analysis was conducted to understand more clearly the results of damper tests on the pendulum platform test fixture. The nature of the motion of the damper-platform rigid-body system in free vibration was sufficiently different from that assumed in the steady forced vibration analysis that the simplified analysis did not fully explain the results of these tests. With the aid of an analog computer, a parametric study of this system was made which yielded results that agreed very well with those obtained from the experimental work.

The differential equations of motion programmed on the analog computer were of the general form:

$$\left. \begin{aligned} a_{11}\ddot{x} + b_{11}\dot{x} + c_{11}x + b_{12}\dot{y} &= 0 \\ a_{22}\ddot{y} + b_{22}\dot{y} + c_{22}y + b_{12}\dot{x} &= 0 \end{aligned} \right\} \quad (2.78)$$

where x and y are principal coordinates, and

the coefficients are constants determined from system parameters. These principal coordinates are related to the generalized coordinates of the physical problem by the expressions:

$$\left. \begin{aligned} x_1 &= x + y \\ x_2 &= \beta_1 x + \beta_2 y \end{aligned} \right\} \quad (2.79)$$

where

x_1 = linear displacement of pendulum bob

x_2 = linear displacement of the platform

and

β_1, β_2 are again constants determined from system parameters.

The relations which define the constants in Equations (2.78) and (2.79) are given on page 46 in the derivation of these equations.

Solutions of Equations (2.78) and (2.79) obtained from the computer were in the form of time plots of the linear displacements of the system. The rate at which the motion of the platform was being damped out by the damper could be measured directly from these plots. These results could also be compared directly with similar plots obtained from the motion of the physical system.

The objective of the study was to determine optimum values for frequency and damping constant of the damper for both the air-bearing fixture and spacecraft applications. Optimum values are those which minimize pendulum bob weight for a given application while maintaining the required time constant for the decay rate of nutational motion. That such optimum values existed was evident from the series of tests conducted with the damper on the test fixture.

The following values were assumed in the analog computer parametric study of the two-body problem:

For the spacecraft:

$$m_1 = 0.164 \text{ lb-sec}^2/\text{in.}$$

$$m_2 = 2.61 \times 10^{-3} \text{ lb-sec}^2/\text{in.}$$

$$p_1 = 4.13 \text{ rad/sec}$$

$$3.72/\text{sec} \leq p_2 \leq 4.54/\text{sec}$$

$$c_1 = 0 \text{ or } 5 \times 10^{-3} \text{ lb-sec/in.}$$

$$5 \times 10^{-4} \leq c_2 \leq 10^{-2} \text{ lb-sec/in.}$$

For the air-bearing fixture:

$$m_1 = 0.234 \text{ lb-sec}^2/\text{in.}$$

$$m_2 = 2.61 \times 10^{-3} \text{ lb-sec}^2/\text{in.}$$

$$p_1 = 3.92 \text{ rad/sec}$$

$$3.53/\text{sec} \leq p_2 \leq 4.32/\text{sec}$$

$$c_1 = 0 \text{ or } 5 \times 10^{-3} \text{ lb-sec/in.}$$

$$5 \times 10^{-4} \leq c_2 \leq 10^{-2} \text{ lb-sec/in.}$$

where m_2 , p_2 , c_2 refer to mass of pendulum bob, frequency, and damping constant of damper, respectively; and m_1 , p_1 , and c_1 refer to similar parameters of the platform. The nonzero value of c_1 was assumed to correlate computer results with test results from the platform fixture. This damping constant for the decay of free vibrations of that fixture was determined experimentally.

The procedure that was followed in optimizing damper parameters was:

1. To optimize c_2 for fixed value of p_2 equal to p_1 (it was determined from the test work that the optimum value of p_2 was very nearly this value).
2. Using the value of c_2 obtained in Step 1, to find the optimum value of p_2 .
3. To determine the value of m_2 that yielded the desired time constant of 10 seconds for optimum values of c_2 and p_2 .
4. Using the obtained values of m_2 , p_2 , and c_2 , to vary p_1 over the range from nominal value

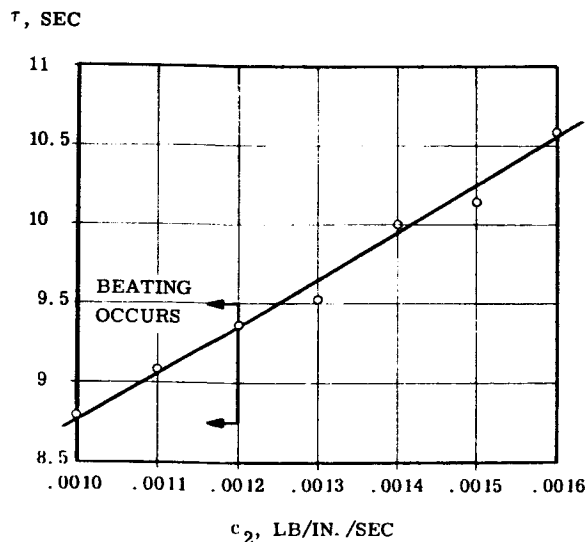


FIGURE 2-8.—Time constant versus fluid damping constant.

minus 10 percent to nominal value plus 10 percent to determine the response of the damper to this variation in forcing frequency.

Results of the study are presented in Figures 2-8 through 2-11.

Figure 2-8 is the plot of time constant versus the damping constant, c_2 , obtained from Step 1 above. The optimum value of $c_2 = 0.0012$ determined from this study was later found to be slightly in error due to the assumption that $p_1 = p_2$. After completion of Step 2, Step 1 was repeated, and the value $c_2 = 0.00175$ was found to be the minimum value for which beating did not occur when the damper was sharply tuned.

Figure 2-9 shows the results obtained from Step 2. This plot of time constant versus damper frequency shows that the damper had to be tuned to a frequency slightly less than nutation frequency for maximum damping rate.

Figure 2-10 is the plot of time constant versus pendulum bob weight obtained from Step 3. A bob weight of about 0.75 lb yielded a 10-second time constant for the assumed nutation frequency of 4.13 rad/sec.

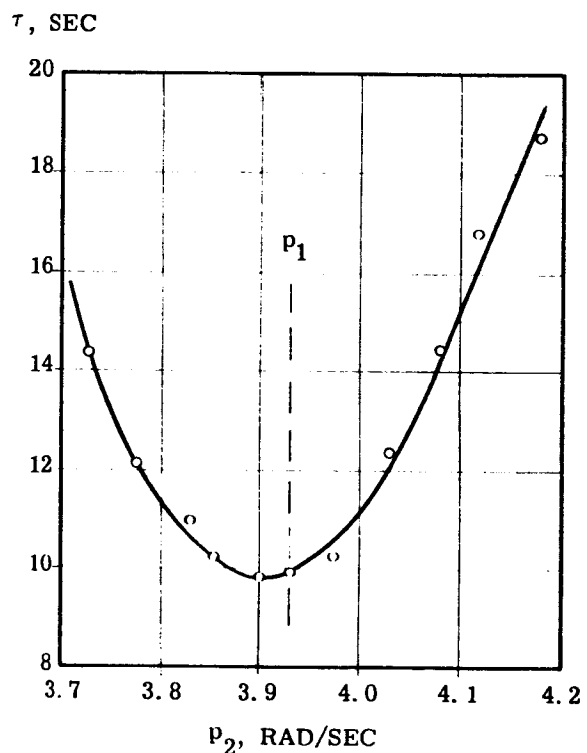


FIGURE 2-9.—Time constant versus damper frequency.

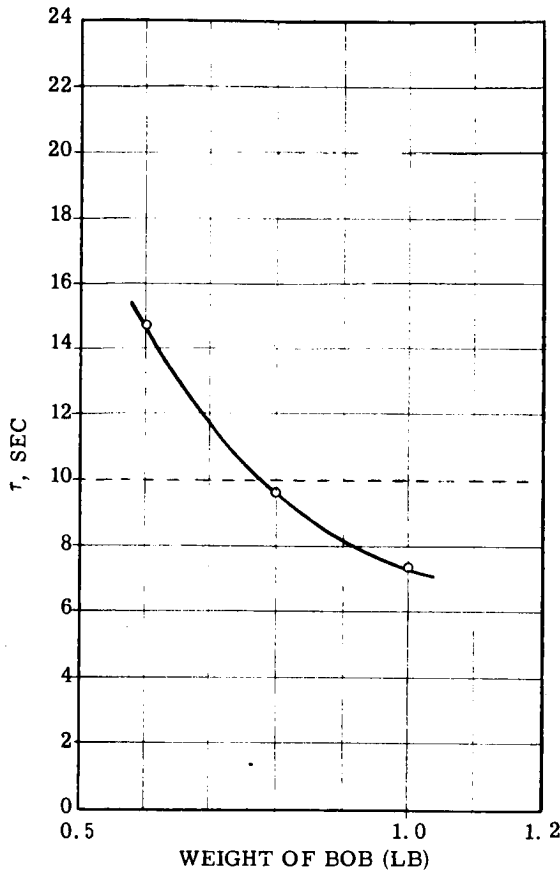


FIGURE 2-10.—Time constant versus pendulum bob weight.

Figure 2-11 shows the variation of damping rate with nutation frequency. Within the frequency range of nominal value ± 10 percent, the damper was very efficient.

Detailed Free-Body Dynamics Analysis

This analysis presents first the gyroscopic motion of a rigid symmetrical body being constantly torqued, then the motions of the same body with no external torquing, and finally a nonsymmetrical body being torqued. The analysis also presents the working equations of motion of the spacecraft as a basis for the nutation damper work.

Gyroscopic Motions Due to Constant Torque

Equations of Motion—Euler's equations relating to the trihedron axes *OMNS* (Figure 2-12) are written as follows:

$$\left. \begin{aligned} \frac{dH_m}{dt} + \omega'_n H_s - \omega'_s H_n &= M_m \\ \frac{dH_n}{dt} + \omega'_m H_m - \omega'_m H_s &= M_n \\ \frac{dH_s}{dt} + \omega'_m H_n - \omega'_n H_m &= M_s \end{aligned} \right\} \quad (2.80)$$

where H_M , H_N , and H_S are the angular momenta of the body with respect to *M*, *N*, and *S* axes. The angular velocities of the trihedron about the *N*, *M*, and *S* axes, denoted by ω'_n , ω'_m and ω'_s , respectively, are:

$$\omega'_n = \dot{\theta}; \quad \omega'_m = -\dot{\alpha} \sin \theta; \quad \omega'_s = \dot{\alpha} \cos \theta \quad (2.81)$$

The body angular velocities are then those of the trihedron plus body motion with respect to the trihedron, namely,

$$\omega_n = \dot{\theta}; \quad \omega_m = -\dot{\alpha} \sin \theta; \quad \omega_s = \dot{\phi} + \dot{\alpha} \cos \theta \quad (2.82)$$

For a body of revolution, *O-M* and *O-N* are always principal axes and the angular momenta are given by the relations:

$$H_n = I_1 \dot{\theta}; \quad H_m = I_1 \dot{\alpha} \sin \theta; \quad H_s = I(\dot{\phi} + \dot{\alpha} \cos \theta) \quad (2.83)$$

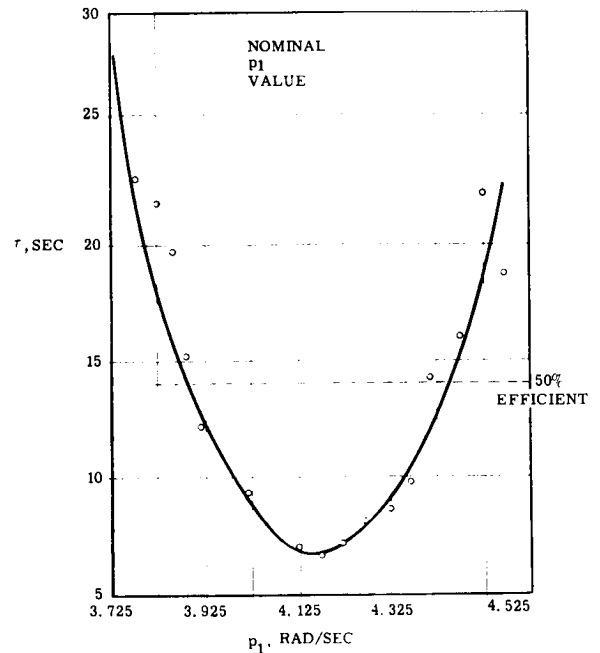


FIGURE 2-11.—Time constant versus nutation frequency ($p_2 = 4.1$ rad/sec; $c_2 = 0.0012$ lb/in./sec).

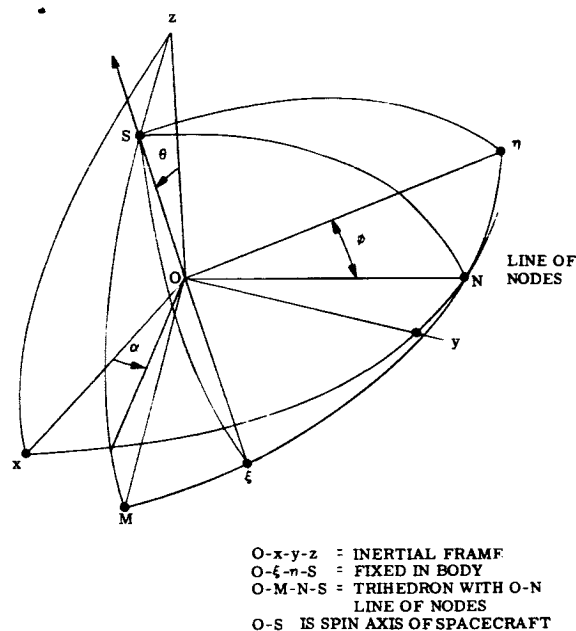


FIGURE 2-12.—Reference coordinate system.

where

I_1 = transverse moment of inertia

I = axial moment of inertia

If Equations (2.81) and (2.83) are substituted into Equations (2.80) the following equations result:

$$\left. \begin{aligned} -I_1 \frac{d}{dt} (\dot{\alpha}^2 \sin \theta) + I\dot{\theta}(\dot{\phi} + \dot{\alpha} \cos \theta) \\ - I_1 \dot{\theta} \dot{\alpha} \cos \theta = M_m \\ I_1 \ddot{\theta} - I_1 \dot{\alpha}^2 \sin \theta \cos \theta + I \dot{\alpha} \sin \theta \\ (\dot{\phi} + \dot{\alpha} \cos \theta) = M_n \\ I \frac{d}{dt} (\dot{\phi} + \dot{\alpha} \cos \theta) = M_s \end{aligned} \right\} \quad (2.84)$$

Equations (2.84) are simplified by assuming that the velocities of the trihedron, θ and α , are small compared to $\dot{\phi}$. Eliminating all terms with products or higher powers of the small velocities in Equations (2.84), we have:

$$\left. \begin{aligned} -I_1 \ddot{\alpha} \sin \theta + I \ddot{\theta} \dot{\phi} &= M_m \\ I_1 \ddot{\theta} + I \dot{\phi} \dot{\alpha} \sin \theta &= M_n \\ I \frac{d}{dt} (\dot{\phi} + \dot{\alpha} \cos \theta) &= M_s \end{aligned} \right\} \quad (2.85)$$

If $\theta = \pi/2$ initially and $M_m = M_s = 0$, this angle will not differ greatly from its initial value during torquing, and Equations (2.85) are further simplified. From the last of these equations, it is apparent that

$$\frac{d\dot{\phi}}{dt} = 0$$

or that $\dot{\phi} = \omega = \text{constant}$.

The first two equations then are written:

$$\left. \begin{aligned} -I_1 \ddot{\alpha} + I \omega \dot{\theta} &= 0 \\ I_1 \ddot{\theta} + I \omega \dot{\alpha} &= M_n \end{aligned} \right\} \quad (2.86)$$

For the assumed initial conditions: $\alpha(0)=\dot{\alpha}(0)=\dot{\theta}=0$; $\theta(0)=\pi/2$, the Laplace transforms of Equations (2.86) are:

$$\left. \begin{aligned} -I_1 s^2 f(s) + I_0 s g(s) - I_0 \left(\frac{\pi}{2} \right) &= 0 \\ I_1 s^2 g(s) - I_1 \left(\frac{\pi}{2} \right) s + I_0 s f(s) &= M_n/s \end{aligned} \right\} \quad (2.87)$$

The solution of these algebraic equations for the functions $f(s)$ and $g(s)$ yields:

$$\left. \begin{aligned} f(s) &= \frac{M_n}{I\omega} \frac{1}{s^2} - \frac{M_n}{I\omega p} \frac{p}{s^2 + p^2} \\ g(s) &= \left(\frac{\pi}{2} = \frac{M_n}{I\omega p} \right) \frac{1}{s} - \frac{M_n}{I\omega p} \frac{s}{s^2 + p^2} \end{aligned} \right\} \quad (2.88)$$

where $p = I\omega/I_1$.

The inverse transforms of Equations (2.88) are:

$$\left. \begin{aligned} \alpha &= \frac{M_n}{I\omega} t - \frac{M_n}{I\omega p} \sin pt \\ \theta &= \frac{\pi}{2} + \frac{M_n}{I\omega p} (1 - \cos pt) \end{aligned} \right\} \quad (2.89)$$

From Equations (2.92) it is determined that the curve of the intersection of the figure axis and a spherical surface described about the fixed point of the gyroscopic body is a cycloid. This curve is shown in Figure 2-13. The cycloidal motion of the figure axis has the period, $\tau = 2\pi/p$, and the amplitude, $a = 2M_{*}/I\omega p$. Since the body is spinning about this axis of

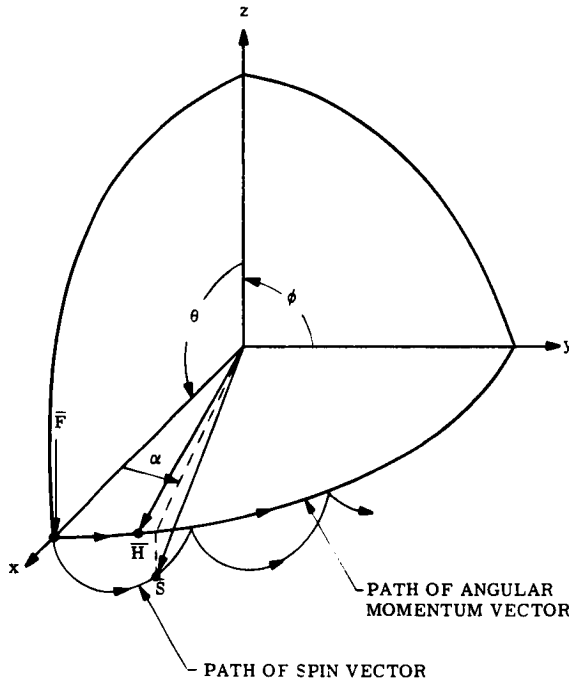


FIGURE 2-13.—Path of spin axis of spacecraft during constant torquing.

symmetry, the cycloidal curve represents the path of the spin vector, S , on the spherical surface during constant torquing, and Equations (2.89) can be used to determine its position and velocity at any given time.

Path of Angular Momentum Vector on Spherical Surface—The total angular momentum of the gyroscopic body is given by the vector equation:

$$\mathbf{H} = I\omega + I_1\dot{\alpha} + I_1\dot{\theta} \quad (2.90)$$

In terms of the unit vectors, $\mathbf{i}$, $\mathbf{j}$, and $\mathbf{k}$, along the S , N , and Z axes, respectively, we have

$$\mathbf{H} = I\omega \cos\left(\theta_s - \frac{\pi}{2}\right)\mathbf{i} - I\omega \sin\left(\theta_s - \frac{\pi}{2}\right)\mathbf{k} + I_1\dot{\theta}\mathbf{j} + I_1\dot{\alpha}\mathbf{k} \quad (2.91)$$

If $\mathbf{H}$ lies in the x - y plane at all times, the scalar product, $\mathbf{H} \cdot \mathbf{k}$, should vanish. From Equation (2.91) it is determined that:

$$\mathbf{H} \cdot \mathbf{k} = -I\omega \sin\left(\theta_s - \frac{\pi}{2}\right) + I_1\dot{\alpha} \quad (2.92)$$

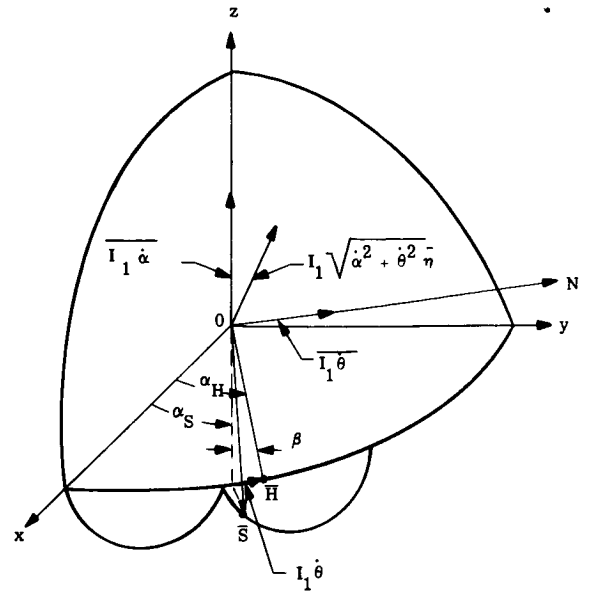


FIGURE 2-14.—Angular velocities during constant torquing.

Since the angle, θ_s , varies from the value $\pi/2$ only by the small amplitude sinusoidal term given in the second of Equations (2.89), Equation (2.92) may be rewritten:

$$\mathbf{H} \cdot \mathbf{k} = -I\omega \left(\theta_s - \frac{\pi}{2}\right) + I_1\dot{\alpha} \quad (2.93)$$

Substituting Equations (2.89) into Equation (2.93), we determine that

$$\begin{aligned} \mathbf{H} \cdot \mathbf{k} &= -I\omega \frac{M_n}{I\omega p} (1 - \cos pt) + I_1 \frac{M_n}{I\omega} (1 - \cos pt) \\ &= \left(-\frac{I\omega}{p} + I_1\right) \frac{M_n}{I\omega} (1 - \cos pt) = 0 \end{aligned}$$

Therefore, $\mathbf{H}$ remains in the x - y plane during constant torquing.

Next, the rate at which $\mathbf{H}$ moves in this plane is determined. In Figure 2-14 let

- α_H = central angle between the x -axis and the angular momentum vector, $\mathbf{H}$
- α_s = central angle between the x -axis and the projection of the spin vector in the x - y plane (S' axis).
- β = central angle between $\mathbf{H}$ and S' .

Then

$$\alpha_H = \alpha_s + \beta$$

or

$$\alpha_H = \alpha_s + \sin^{-1} \frac{I_1 \dot{\theta}}{\sqrt{I^2 \omega^2 + I_1^2 (\dot{\alpha}^2 + \dot{\theta}^2)}} \quad (2.94)$$

If $\dot{\alpha}$ and $\dot{\theta}$ are sufficiently small in comparison to ω , Equation (2.94) can be simplified to:

$$\alpha_H = \alpha_s + \frac{I_1}{I\omega} \dot{\theta} \quad (2.95)$$

Equation (2.89) may be substituted into the differentiated form of Equation (2.95) to give:

$$\dot{\alpha}_H = \dot{\alpha}_s + \frac{1}{p} \ddot{\theta} = \frac{M_n}{I\omega} (1 - \cos pt) + \frac{1}{p} \left(\frac{M_n p}{I\omega} \right) = \frac{M_n}{I\omega} \quad (2.96)$$

Thus, except for small pulsations brought about by the $(\dot{\alpha}^2 + \dot{\theta}^2)$ term in Equation (2.94) H moves at the constant rate, $M_n/I\omega$, in the xy -plane.

The magnitude of these variations from the average rate can be determined by differentiating Equation (2.94) with respect to time and then making the appropriate substitutions for α and θ and their time derivatives from Equations (2.89). The resulting expression reveals that at

$$t = 2n\pi/p \quad (n=0, 1, \dots)$$

$\dot{\alpha}_H$ is exactly equal to the average rate, is slightly greater than this value at $t = n\pi/p$, and is slightly less at $t = (2n+1)\pi/2p$.

Free Motion of a Gyroscopic Body

The residual motion of the gyroscopic body after the constant torque, M_n , has been stopped is now analyzed. In Equations (2.86) assume that $M_n = 0$, but that initial velocities $\dot{\alpha}_0$ and $\dot{\theta}_0$ exist. The initial value problem is then stated:

$$\left. \begin{aligned} I_1 \ddot{\alpha} + I\omega \dot{\theta} &= 0 \\ I_1 \ddot{\theta} + I\omega \dot{\alpha} &= 0 \\ \alpha(0) &= \alpha_0; \dot{\alpha}(0) = \dot{\alpha}_0; \theta(0) = \theta_0; \dot{\theta}(0) = \dot{\theta}_0 \end{aligned} \right\} \quad (2.97)$$

Laplace transforms of Equations (2.97) are:

$$\left. \begin{aligned} -I_1 s^2 f(s) + I_1 s \alpha_0 + I_1 \dot{\alpha}_0 + I\omega s g(s) \\ -I\omega \theta_0 &= 0 \\ I_1 s^2 g(s) - I_1 s \theta_0 - I_1 \dot{\theta}_0 + I\omega s f(s) - I\omega \alpha_0 &= 0 \end{aligned} \right\} \quad (2.98)$$

from which

$$\left. \begin{aligned} f(s) &= \left(\alpha_0 + \frac{\dot{\theta}_0}{p} \right) \frac{1}{s} - \frac{\theta_0}{p} \frac{s}{s^2 + p^2} + \frac{\dot{\alpha}_0}{p} \frac{p}{s^2 + p^2} \\ g(s) &= \left(\theta_0 - \frac{\dot{\alpha}_0}{p} \right) \frac{1}{s} - \frac{\dot{\alpha}_0}{p} \frac{s}{s^2 + p^2} + \frac{\dot{\theta}_0}{p} \frac{p}{s^2 + p^2} \end{aligned} \right\} \quad (2.99)$$

The inverse transforms of Equations (2.99) are:

$$\left. \begin{aligned} \alpha &= \alpha_0 + \frac{\dot{\theta}_0}{p} - \frac{\dot{\theta}_0}{p} \cos pt + \frac{\dot{\alpha}_0}{p} \sin pt \\ \theta &= \theta_0 - \frac{\dot{\alpha}_0}{p} + \frac{\dot{\alpha}_0}{p} \cos pt + \frac{\dot{\theta}_0}{p} \sin pt \end{aligned} \right\} \quad (2.100)$$

Thus, the figure axis performs harmonic motion about the position,

$$\left(\alpha_0 + \frac{\dot{\theta}_0}{p}, \theta_0 - \frac{\dot{\alpha}_0}{p} \right),$$

described by the equations:

$$\left. \begin{aligned} \alpha - \frac{\dot{\alpha}_0}{p} \sin pt - \frac{\dot{\theta}_0}{p} \cos pt &= A \sin (pt - \beta_1) \\ \theta - \frac{\dot{\theta}_0}{p} \sin pt + \frac{\dot{\alpha}_0}{p} \cos pt &= A \sin (pt - \beta_2) \end{aligned} \right\} \quad (2.101)$$

where

$$A = \frac{1}{p} \sqrt{\dot{\alpha}_0^2 + \dot{\theta}_0^2}$$

$$\beta_1 = \tan^{-1} \frac{\dot{\theta}_0}{\dot{\alpha}_0}$$

$$\beta_2 = \tan^{-1} \left(-\frac{\dot{\alpha}_0}{\dot{\theta}_0} \right)$$

Since

$$\tan (\beta_1 - \beta_2) = \frac{\tan \beta_1 - \tan \beta_2}{1 + \tan \beta_1 \tan \beta_2} = \infty$$

$\beta_1 - \beta_2 = \pi/2$, and the path of the figure axis on

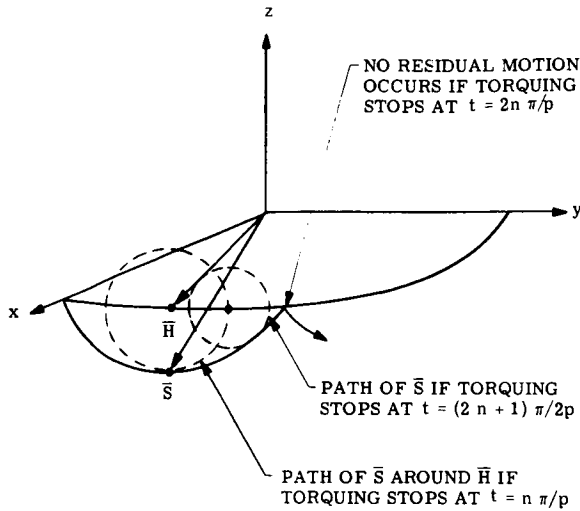


FIGURE 2-15.—Free body precession of spacecraft after constant torquing stops.

the spherical surface described about the fixed point of the gyroscopic body is a circle of radius,

$$r = \frac{1}{p} \sqrt{\dot{\alpha}_0^2 + \dot{\theta}_0^2}$$

From Equations (2.89), the center of this circle is determined to be:

$$\left. \begin{aligned} \alpha &= \alpha_0 + \frac{\dot{\theta}_0}{p} = \alpha_0 + \frac{M_n}{I\omega p} \sin pt_0 \\ \theta &= \theta_0 - \frac{\dot{\alpha}_0}{p} = \frac{\pi}{2} + \frac{M_n}{I\omega p} (1 - \cos pt_0) \\ &\quad - \frac{M_n}{I\omega p} (1 - \cos pt_0) = \frac{\pi}{2} \end{aligned} \right\} \quad (2.102)$$

From Equation (2.95), the α coordinate of the angular momentum vector is found to be:

$$\alpha_{H_0} = \alpha_0 + \frac{\dot{\theta}_0}{p} = \alpha_0 + \frac{M_n}{I\omega p} \sin pt_0$$

which is the same as the α coordinate of the center of the circle given in Equation (2.102), above. Since it was shown in the preceding section that the angular momentum vector, $\mathbf{H}$, remains in the xy -plane ($\theta = \pi/2$) at all times during torquing, it is concluded that when torquing of the gyroscopic body is stopped, the residual motion is a circular motion of the spin vector about the angular momentum vector on

the spherical surface. From Equations (2.101), it is seen that the amplitude of this motion is dependent upon the position of the figure axis on the cycloidal path at the time torquing is stopped. A maximum amplitude of $2M_n/I\omega p$ will occur if $t = n\pi/p$ and no residual motion is present if torquing stops when $t = 2n\pi/p$, or when the spin vector is coincident with the angular momentum vector. This motion is illustrated in Figure 2-15.

Undamped Motion of Spacecraft Rigid-Body System During Constant Torque

The equations of motion derived on page 36 apply to a single symmetric rigid body under constant torque. Extension of that analysis is now made to determine the effect on spacecraft motion of the nonrotating bodies of the spacecraft.

Let I and I_1 be the axial and transverse moments of inertia of the rotating body of the spacecraft. The angular momenta of this body about the trihedron axes, $OMNS$, of Figure 2-12 are then given by Equations (2.83). To determine the total angular momentum of the spacecraft about these axes, these additional momentum terms must be considered.

1. Upper assembly:

$$\left. \begin{aligned} H_{n_2} &= I_{n_2} \dot{\theta} \\ H_{m_2} &= -I_{m_2} \dot{\alpha} \sin \theta \\ H_{s_2} &= I_{s_2} \dot{\alpha} \cos \theta \end{aligned} \right\} \quad (2.103)$$

2. Instrument:

$$\left. \begin{aligned} H_{n_3} &= I_{n_3} \dot{\theta} \\ H_{m_3} &= -I_{m_3} \dot{\alpha} \sin \theta \\ H_{s_3} &= I_{s_3} \dot{\alpha} \cos \theta \end{aligned} \right\} \quad (2.104)$$

where the subscripts n , m , and s refer to the moment of inertia or angular momentum of the body about the N , M , and S axes, respectively, and the subscripts 2 and 3 refer to the upper assembly and instruments, respectively.

Equations (2.103) and (2.104) are derived assuming that the azimuth servo motor prevents rotation of the upper assembly about

the spin axis and that the elevation servo prevents rotation of the instrument about the M -axis. The inertial term, I_{m_3} , then applies only to the motion of the center of mass of the instrument about the center of gravity of the spacecraft.

The total angular momenta of the composite body are obtained by adding Equations (2.83), (2.103) and (2.104):

$$\left. \begin{aligned} H_n &= (I_1 + I_{n_2} + I_{n_3})\dot{\theta} = I_a\dot{\theta} \\ H_m &= -(I_1 + I_{m_2} + I_{m_3})\dot{\alpha} \sin \theta \\ &= -I_b\dot{\alpha} \sin \theta \\ H_s &= I\dot{\phi} + (I + I_{s_2} + I_{s_3})\dot{\alpha} \cos \theta \\ &= I\dot{\phi} + I_c\dot{\alpha} \cos \theta \end{aligned} \right\} \quad (2.105)$$

Substituting Equations (2.105) into Euler's Equations (2.80) yields:

$$\left. \begin{aligned} -I_b \frac{d}{dt} (\dot{\alpha} \sin \theta) + \theta(I\dot{\phi} + I_c\dot{\alpha} \cos \theta) - \dot{\alpha} \cos \theta(I_a\dot{\theta}) &= M_m \\ I_a\ddot{\theta} + \dot{\alpha} \cos \theta(-I_b\dot{\alpha} \sin \theta) + \dot{\alpha} \sin \theta(I\dot{\phi} + I_c\dot{\alpha} \cos \theta) &= M_n \\ \frac{d}{dt} (I\dot{\phi} + I_c\dot{\alpha} \cos \theta) - \dot{\alpha} \sin \theta(I_a\dot{\theta}) - \dot{\theta}(-I_b\dot{\alpha} \sin \theta) &= M_s \end{aligned} \right\} \quad (2.106)$$

Assuming θ and $\dot{\alpha}$ small compared to $\dot{\phi}$, and dropping terms with products or higher powers of the small velocities, we have

$$\left. \begin{aligned} -I_b\dot{\alpha} \sin \theta + I\dot{\phi}\dot{\theta} &= M_m \\ I_a\ddot{\theta} + I\dot{\phi}\dot{\alpha} \sin \theta &= M_n \\ \frac{d}{dt} (I\dot{\phi} + I_c\dot{\alpha} \cos \theta) &= M_s \end{aligned} \right\} \quad (2.107)$$

If it is assumed that $M_u = M_s = 0$, $\theta = \pi/2$, then $\sin \theta = 1$, $\cos \theta = 0$ and the last of Equations (2.107) shows that $\dot{\phi} = \text{constant}$, say ω . The first two equations become:

$$\left. \begin{aligned} -I_b\dot{\alpha} + I\omega\dot{\theta} &= 0 \\ I_a\ddot{\theta} + I\omega\dot{\alpha} &= M_n \end{aligned} \right\} \quad (2.108)$$

Laplace transforms of Equations (2.108) for the initial conditions, $\alpha(0) = \dot{\alpha}(0) = \dot{\theta}(0) = 0$, $\theta(0) = \pi/2$, are:

$$\left. \begin{aligned} -I_b s^2 f(s) + I\omega s g(s) - I\omega \left(\frac{\pi}{2}\right) &= 0 \\ I_a s^2 g(s) - I_a \left(\frac{\pi}{2}\right) s + I\omega s f(s) &= M_n/s \end{aligned} \right\} \quad (2.109)$$

The solution of Equations (2.109) for $f(s)$ and $g(s)$ is:

$$\left. \begin{aligned} f(s) &= \frac{M_n}{I\omega s^2} - \frac{I}{I\omega p} \frac{p}{s^2 + p^2} \\ g(s) &= \left(\frac{\pi}{2} - \frac{M_n I_b}{I^2 \omega^2}\right) \frac{1}{s} - \frac{M_n I_c}{I^2 \omega^2} \frac{s}{s^2 + p^2} \end{aligned} \right\} \quad (2.110)$$

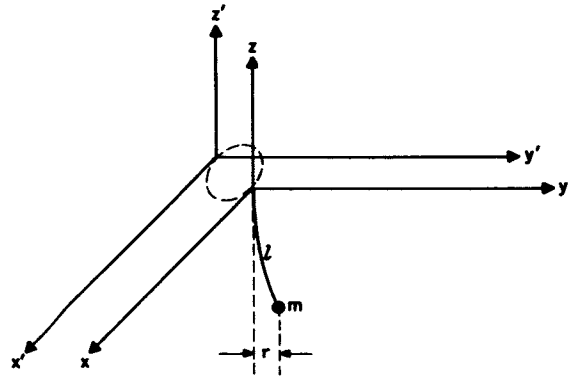
where $p = I\omega/\sqrt{I_a I_b}$.

The inverse transforms of Equations (2.110) are:

$$\left. \begin{aligned} \alpha &= \frac{M_n}{I\omega} t - \frac{M_n}{I\omega p} \sin pt \\ \theta &= \frac{\pi}{2} + \frac{M_n I_b}{I^2 \omega^2} (1 - \cos pt) \end{aligned} \right\} \quad (2.111)$$

Thus, for a composite body with unequal total moments of inertia about the transverse axes, M and N , both the frequency and amplitude of nutational motion are different than for the symmetric body.

Spherical Pendulum Damper



Equations of Motion—Assume that a spherical pendulum is acted upon by a sinusoidal forcing

function that displaces the point of suspension in a circular path in the horizontal plane. If the primed axes are inertial and the unprimed axes move with the point of suspension, the two coordinate systems are related by the equations:

$$\left. \begin{aligned} x' &= x + A' \sin pt \\ y' &= y + A' \cos pt \\ z' &= z \end{aligned} \right\} \quad (2.112)$$

where

$$A' = \frac{M}{I\omega p} r_0 \text{ and } p = I\omega/I_1$$

The kinetic energy is

$$T = \frac{1}{2} m v^2 = \frac{1}{2} m (\dot{x}'^2 + \dot{y}'^2 + \dot{z}'^2)$$

or

$$T = \frac{1}{2} m [(\dot{x} + A' p \cos pt)^2 + (\dot{y} - A' p \sin pt)^2 + \dot{z}^2] \quad (2.113)$$

The potential energy due to the strain energy of bending the pendulum arm wire is, as previously derived:

$$V_1 = \int_0^l \frac{EI}{2\rho^2} dz \cong \frac{EI}{2} \left(\frac{d^2 r}{dz^2} \right)^2 dz \quad (2.114)$$

The potential energy due to the gravity force, mg , is as previously derived:

$$V_2 = W \int_0^l (ds - dz) \cong \frac{1}{2} W \int_0^l \left(\frac{dr}{dz} \right)^2 dz \quad (2.115)$$

If it is assumed that the deflection curve of the wire is approximated by the equation

$$r = r_b \left(1 - \cos \frac{\pi z}{2} \right) \quad (2.116)$$

then

$$V_1 = \frac{EI}{2} \int_0^l \left(\frac{r_b^2 \pi^4}{16 l^4} \cos^2 \frac{\pi z}{2} \right) dz = \frac{EI r_b^2 \pi^4}{64 l^3} \quad (2.117)$$

and

$$V_2 = \frac{W}{2} \int_0^l \frac{r_b^2 \pi^2}{4 l^2} \sin^2 \frac{\pi z}{2} dz = \frac{W r_b^2 \pi^2}{16 l} \quad (2.118)$$

Total potential energy is

$$V = V_1 + V_2 = \frac{1}{2} \left(\frac{EI \pi^4}{32 l^3} + \frac{W \pi^2}{8 l} \right) r_b^2 \quad (2.119)$$

or

$$V = \frac{1}{2} k r_b^2 = \frac{1}{2} k (x^2 + y^2)$$

From equation (2.113) we determine that:

$$\frac{\partial T}{\partial x} = \frac{\partial T}{\partial y} = \frac{\partial T}{\partial z} = 0$$

$$\frac{\partial T}{\partial \dot{x}} = m(\dot{x} + A' p \cos pt)$$

$$\frac{\partial T}{\partial \dot{y}} = m(\dot{y} - A' p \sin pt);$$

$$\frac{\partial T}{\partial \dot{z}} = m \dot{z}$$

$$\frac{d}{dt} \left(\frac{\partial T}{\partial \dot{x}} \right) = m \ddot{x}$$

$$\frac{d}{dt} \left(\frac{\partial T}{\partial \dot{y}} \right) = m \ddot{y} - m A' p^2 \sin^2 pt$$

$$\frac{d}{dt} \left(\frac{\partial T}{\partial \dot{z}} \right) = m \ddot{z} - m A' p^2 \cos pt$$

and from Equation (2.119)

$$\frac{\partial V}{\partial x} = kx; \quad \frac{\partial V}{\partial y} = ky; \quad \frac{\partial V}{\partial z} = 0$$

For viscous damping, the dissipation function is

$$F = \frac{1}{2} c (\dot{x}^2 + \dot{y}^2 + \dot{z}^2) \quad (2.120)$$

and

$$\frac{\partial F}{\partial \dot{x}} = c \dot{x}; \quad \frac{\partial F}{\partial \dot{y}} = c \dot{y}; \quad \frac{\partial F}{\partial \dot{z}} = c \dot{z}$$

The Lagrangian equations of motion are:

$$\left. \begin{aligned} m \ddot{x} - m A' p^2 \sin pt + c \dot{x} + kx &= 0 \\ m \ddot{y} - m A' p^2 \cos pt + c \dot{y} + ky &= 0 \\ m \ddot{z} + c \dot{z} &= 0 \end{aligned} \right\} \quad (2.121)$$

Letting $2n=c/m$ and $p'^2=k/m$, Equations (2.121) become:

$$\left. \begin{aligned} \ddot{x} + 2n\dot{x} + p'^2x &= A' p^2 \sin pt \\ \ddot{y} + 2n\dot{y} + p'^2y &= A' p^2 \cos pt \\ \ddot{z} + 2n\dot{z} &= 0 \end{aligned} \right\} \quad (2.122)$$

Solution of the third of Equations (2.122) is easily determined to be:

$$z = c_1 + c_2 e^{-2nt} \quad (2.123)$$

Thus z gradually changes from some initial value to another fixed value, and vertical motion of the bob is negligible during steady forced vibration. The particular solutions of the first two of Equations (2.122) are:

$$\left. \begin{aligned} x &= A'' \sin(pt - \alpha) \\ y &= A'' \cos(pt - \alpha) \end{aligned} \right\} \quad (2.124)$$

where

$$A'' = A' \frac{p^2/p'^2}{\sqrt{(-p^2/p'^2)^2 + 4n^2 p^2/p'^4}}$$

and

$$\alpha = \tan^{-1} \frac{2np}{p'^2 - p^2}$$

Energy Dissipation Rate—Energy loss/sec = work/cycle $\times$ cycle/sec

$$= -\dot{E} = \frac{c}{2\pi} \int_0^{2\pi} \dot{x}^2 d(pt) \quad (2.125)$$

From Equation (2.125), the energy dissipation rate is determined to be:

$$\begin{aligned} -\dot{E} &= \frac{c}{2\pi} \int_0^{2\pi} (\dot{x}^2 + \dot{y}^2) d(pt) \\ &= A'^2 \frac{cp^2}{(1 - p^2/p'^2)^2 + 4n^2 p^2/p'^4} \end{aligned} \quad (2.126)$$

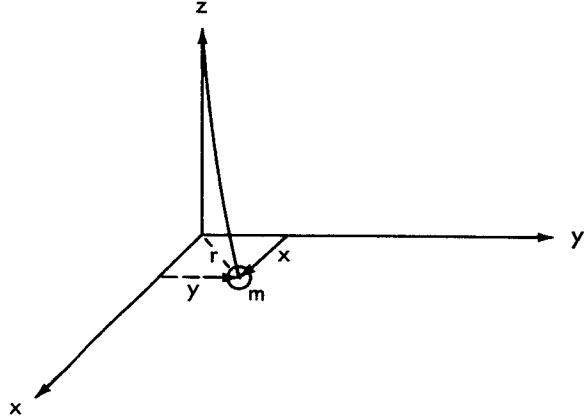
At resonance, this expression becomes:

$$-\dot{E} = A'^2 \frac{cp^4}{4n^2} = A^2 \frac{m^2 p^4}{c} \quad (2.127)$$

Frequency Equations for Spherical Pendulum Damper

In order to use the energy method for determining the natural frequency of vibration of

the spherical pendulum damper, it is first necessary to show that for small vibrations the natural frequency of the damper is the same when operating as a simple pendulum as when performing conical pendulum motion. This is done by considering the free undamped oscillations of the damper during each type of motion.



Lateral Motion—If the pendulum is performing small lateral vibrations in the yz -plane, the kinetic and potential energies of the damper are:

$$\left. \begin{aligned} T &= \frac{1}{2} m \dot{y}^2 \\ V &= \frac{1}{2} k y^2 \end{aligned} \right\} \quad (2.128)$$

where k is determined from the spring force of the wire and the gravity force, mg . The Lagrangian equation of motion is:

$$m\ddot{y} + ky = 0 \quad (2.129)$$

Equation (2.129) is the familiar expression describing free undamped sinusoidal oscillations of frequency, $p = \sqrt{k/m}$.

Rotary Motion—When the pendulum bob is performing small rotary oscillations in the xy -plane, the kinetic and potential energies are:

$$\left. \begin{aligned} T &= \frac{1}{2} m v^2 = \frac{1}{2} m (\dot{x}^2 + \dot{y}^2) \\ V &= \frac{1}{2} k r^2 = \frac{1}{2} k (x^2 + y^2) \end{aligned} \right\} \quad (2.130)$$

Since the spring force and gravity force acting on the damper are unchanged during the two types of motion, the derivation of the constant, k , is the same in Equations (2.130) as in Equations (2.128). The Lagrangian equations may be derived from the energy equations, Equation (2.130).

$$\left. \begin{aligned} m\ddot{y} + ky &= 0 \\ m\ddot{x} + kx &= 0 \end{aligned} \right\} \quad (2.131)$$

Thus, for small vibrations the frequency of the damper in the two cases is identical. The expressions for this frequency for the three different damper environments being considered are now derived using the energy method. The environments to be considered are:

- Pendulum mounted in upright position in gravity field
- Pendulum mounted in inverted position in gravity field
- Gravity-free environment of orbital flight

If the motion of the pendulum is simple harmonic and the potential and kinetic energies are made up of separable parts, the general expression for frequency may be written:

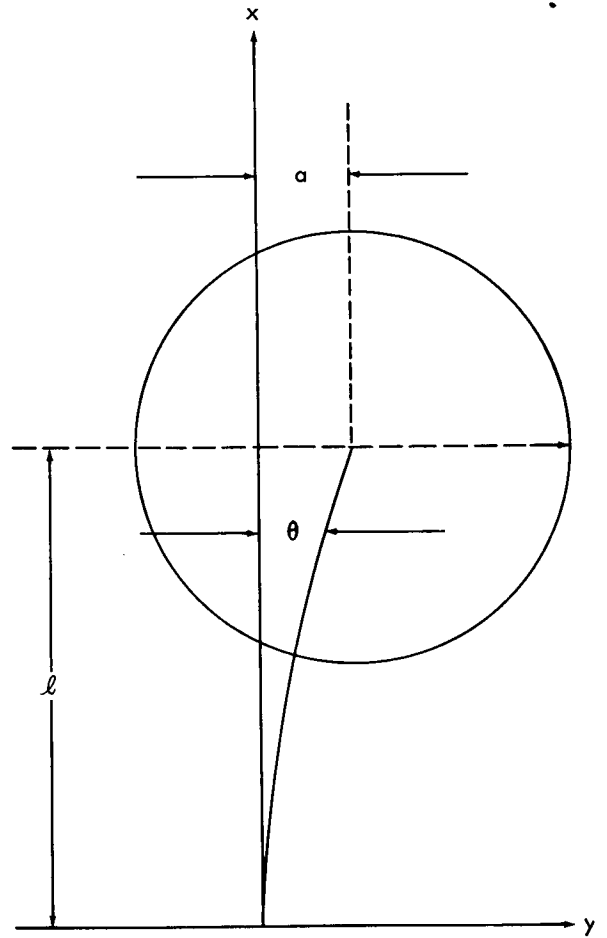
$$p = \sqrt{\frac{\sum_i P_i}{\sum_i K_i}} \quad (2.132)$$

where

P = maximum potential energy
 Kp^2 = maximum kinetic energy

Equation (2.132) follows directly from the law of conservation of energy.

For the pendulum damper in the gravity field, the following energy expressions are derived:



1. *Kinetic energy* is due to the linear and rotational motions of the pendulum bob and spring wire. If w is the constant weight per unit length of the wire and W the weight of the bob, the expression for the kinetic energy due to linear motion of the concentrated load, W , can be shown to be:

$$K_1 p^2 = \frac{1}{2} m v^2 = \frac{1}{2} \left(\frac{W}{g} \right) p^2 y l^2 \quad (2.133)$$

and for the distributed load, w :

$$K_2 p^2 = \frac{1}{2} \frac{w}{g} p^2 \int_0^l y^2 dx \quad (2.134)$$

The kinetic energies due to rotational motion are:

(a) For the concentrated load:

$$K_3 p^2 = \frac{1}{2} m \rho_1^2 \dot{\theta}_{\max}^2 = \frac{1}{2} \frac{W}{g \rho_1^2} \frac{a^2 p^2 \pi^2}{4 l^2} \quad (2.135)$$

where ρ_1 = radius of gyration of bob about its axis of rotation. Derivation of the value of $\dot{\theta}_{\max}$ in Equation (2.135) is as follows:

For small vibrations,

$$\theta = dy/dx; \quad \dot{\theta} = \frac{d}{dt} \left(\frac{dy}{dx} \right)^2 = \left(\frac{d\dot{y}}{dx} \right)^2$$

At any given point along the deflection curve:

$$y(t) = y(x) \sin pt; \quad \dot{y}(t) = y(x) p \cos pt$$

and

$$\dot{y}_{\max} = y(x) p$$

If the equation of the deflection curve is assumed to be:

$$y(x) = a \left(1 - \cos \frac{\pi x}{2} \right)$$

the above expression may be rewritten to:

$$\dot{y}_{\max} = ap \left(1 - \cos \frac{\pi x}{2l} \right)$$

Furthermore, we have

$$\frac{d\dot{y}_{\max}}{dx} = \frac{ap\pi}{2l} \sin \frac{\pi x}{2l}$$

and at $y = y_l$:

$$\frac{d\dot{y}_{\max}}{dx} = \frac{ap\pi}{2l}$$

Then at y_l :

$$\dot{\theta}_{\max}^2 = \left[\frac{d\dot{y}_{\max}}{dx} \right]^2 = \frac{a^2 p^2 \pi^2}{4 l^2}$$

(b) Similarly, for the distributed load:

$$K_4 p^2 = \frac{1}{2} p^2 \int_0^l \frac{w}{g} \rho_2^2 \left(\frac{dy}{dx} \right)^2 dx \quad (2.136)$$

where ρ_2 = radius of gyration of spring wire.

2. *Potential energy* is due to the elastic potential of the spring wire and the gravitational potential of the bob and wire. Maximum potential energy is determined by evaluating these energies when the pendulum is at maximum displacement.

(a) Due to the elastic potential of the spring wire:

$$P_1 = EI \int_0^l \left(\frac{d^2 y}{dx^2} \right)^2 dx \quad (2.137)$$

(b) The gravitational potential of the concentrated weight, W , is:

$$P_2 = \pm \frac{W}{2} \int_0^l \left(\frac{dy}{dx} \right)^2 dx \quad (2.138)$$

where the positive sign is used for the upright pendulum and the negative sign for the inverted position.

(c) The gravitational potential of the distributed weight of the wire is:

$$P_3 = \pm \int_0^l \left[w \int_0^x \frac{1}{2} \left(\frac{dy}{dx} \right)^2 dx \right] dx \quad (2.139)$$

with the sign convention the same as in Equation (2.138).

Substitution of Equations (2.133) through (2.139) into Equation (2.132) yields the frequency equation:

$$p^2 = \frac{EI \int_0^l \left(\frac{d^2 y}{dx^2} \right)^2 dx \pm W \int_0^l \left(\frac{dy}{dx} \right)^2 dx \pm \int_0^l \left[w \int_0^x \left(\frac{dy}{dx} \right)^2 dx \right] dx}{\left(\frac{W}{g} \right) y_l^2 + \left(\frac{W}{g} \right) \rho_1^2 \frac{a^2 \pi^2}{4 l^2} + \frac{w}{g} \int_0^l y^2 dx + \frac{w}{g} \rho_2^2 \left(\frac{dy}{dx} \right)^2 dx} \quad (2.140)$$

For the assumed deflection curve:

$$y = a \left(1 - \cos \frac{\pi x}{2l} \right)$$

the terms in Equation (2.140) are determined to be:

$$EI \int_0^l \left(\frac{d^2 y}{dx^2} \right)^2 dx = \frac{a^2 \pi^4 EI}{32 l^3}$$

$$W \int_0^l \left(\frac{dy}{dx} \right)^2 dx = \frac{a^2 \pi^2 W}{8 l}$$

$$\int_0^l \left[w \int_0^x \left(\frac{dy}{dx} \right)^2 dx \right] dx = \frac{a^2 \pi^2 w}{4} \left(\frac{1}{4} - \frac{1}{\pi^2} \right) \cdot \left(\frac{W}{g} \right) y_l^2 = \frac{W a^2}{g}$$

$$\frac{w}{g} \int_0^l y^2 dx = \frac{a^2 w l}{g} \left(\frac{3}{2} - \frac{4}{\pi} \right)$$

$$\frac{w}{g} \rho_2^2 \int_0^l \left(\frac{dy}{dx} \right)^2 dx = \frac{a^2 \pi^2 w \rho_2^2}{8 g l}$$

These expressions are substituted into Equation (2.140) and simplified to give:

$$p = 34.4 \sqrt{\frac{EI \pm 0.405 W l^2 \pm 0.120 w l^3}{W l^3 + 2.467 W l \rho_1^2 + 0.227 w l^4 + 1.234 w l^2 \rho_2^2}} \quad (2.141)$$

When the weight of the wire is small compared to bob weight, Equation (2.141) can be simplified to the expression:

$$p = 34.4 \sqrt{\frac{EI \pm 0.405 W l^2}{W l^3 + 2.467 W l \rho_1^2}} \quad (2.142)$$

In the gravity-free environment of orbital flight, the term in Equation (2.141) derived from the gravitational potential of the bob is omitted and the frequency equation becomes:

$$p = 1.745 \sqrt{\frac{EI}{m l^3 + 2.467 m l \rho_1^2}} \quad (2.143)$$

where m = mass of the pendulum bob.

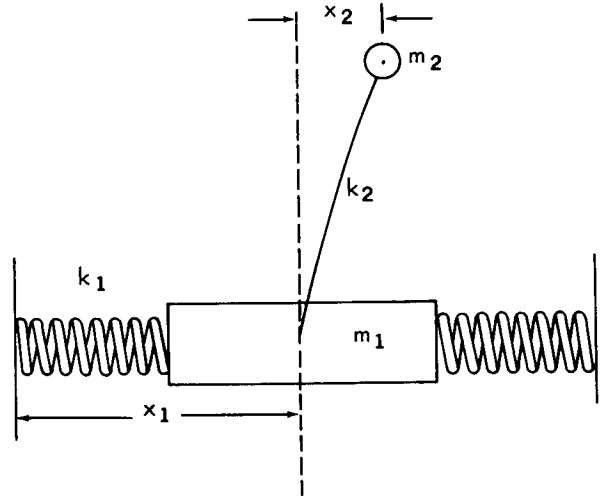
Nutation Damper—Test Platform Vibration Analysis

The equations of motion of the nutation damper-pendulum platform two-body system performing small single-plane vibrations are derived using the method of Timoshenko and Young.[†]

[†]S. Timoshenko and D. H. Young, *Advanced Dynamics*, McGraw Hill, New York, 1948.

Kinetic energy of the system is:

$$T = \frac{1}{2} m_1 \dot{x}_1^2 + \frac{1}{2} m_2 (\dot{x}_1 + \dot{x}_2)^2 \quad (2.144)$$



Potential energy may be written:

$$V = \frac{1}{2} k_1 x_1^2 + \frac{1}{2} k_2 x_2^2 \quad (2.145)$$

If viscous damping is assumed for both masses, the dissipation function is given by the expression:

$$F = \frac{1}{2} c_1 \dot{x}_1^2 + \frac{1}{2} c_2 \dot{x}_2^2 \quad (2.146)$$

In Equations (2.144) through (2.146), the following notation is used:

m_1 = mass of platform
 m_2 = mass of pendulum bob
 $k_1 = m_1 p_1^2$ = spring constant of platform
 where p_1 = natural frequency of vibration of platform
 $k_2 = m_2 p_2^2$ = spring constant of pendulum
 where p_2 = natural frequency of pendulum

c_1 = damping constant of platform
 c_2 = damping constant of damper

Since x_1 and x_2 are not principal coordinates, the following change of variable is made:

$$\begin{aligned} x_1 &= x + y \\ x_2 &= \beta_1 x + \beta_2 y \end{aligned} \quad (2.147)$$

Substituting Equations (2.147) into Equations (2.144) and (2.145), we obtain:

$$T = \frac{1}{2}[m_1 + m_2(1 + \beta_1)^2]\dot{x}^2 + [m_1 + m_2(1 + \beta_1)(1 + \beta_2)]\dot{x}\dot{y} + \frac{1}{2}[m_1 + m_2(1 + \beta_2)^2]\dot{y}^2 \quad (2.148)$$

$$V = \frac{1}{2}(k_1 + k_2\beta_1^2)x^2 + (k_1 + k_2\beta_1\beta_2)xy + \frac{1}{2}(k_1 + k_2\beta_2^2)y^2 \quad (2.149)$$

The cross product terms in Equations (2.148) and (2.149) are eliminated if:

$$\left. \begin{aligned} m_1 + m_2(1 + \beta_1)(1 + \beta_2) &= 0 \\ k_1 + k_2\beta_1\beta_2 &= 0 \end{aligned} \right\} \quad (2.150)$$

If β_1 and β_2 are given by Equations (2.150), we have

$$\left. \begin{aligned} T &= \frac{1}{2}[m_1 + m_2(1 + \beta_1)^2]\dot{x}^2 + \frac{1}{2}[m_1 + m_2(1 + \beta_2)^2]\dot{y}^2 \\ V &= \frac{1}{2}(k_1 + k_2\beta_1^2)x^2 + \frac{1}{2}(k_1 + k_2\beta_2^2)y^2 \\ F &= \frac{1}{2}(c_1 + c_2\beta_1^2)\dot{x}^2 + (c_1 + c_2\beta_1\beta_2)\dot{x}\dot{y} + \frac{1}{2}(c_1 + c_2\beta_2^2)\dot{y}^2 \end{aligned} \right\} \quad (2.151)$$

The Lagrangian equations of motion may be written from Equations (2.151) as follows:

$$\left. \begin{aligned} a_{11}\ddot{x} + b_{11}\dot{x} + c_{11}x + b_{12}\dot{y} &= 0 \\ a_{22}\ddot{y} + b_{22}\dot{y} + c_{22}y + b_{12}\dot{x} &= 0 \end{aligned} \right\} \quad (2.152)$$

where

$$\begin{aligned} a_{11} &= m_1 + m_2(1 + \beta_1)^2; & a_{22} &= m_1 + m_2(1 + \beta_2)^2; \\ b_{11} &= c_1 + c_2\beta_1^2; & b_{22} &= c_1 + c_2\beta_2^2; \\ c_{11} &= k_1 + k_2\beta_1^2; & c_{22} &= k_1 + k_2\beta_2^2; \\ b_{12} &= c_1 + c_2\beta_1\beta_2 \end{aligned}$$

EFFECTS OF EXTERNAL TORQUES ON SPACECRAFT

External torques acting on the spacecraft caused changes in the position of the spin axis

relative to the sun and changes in the spin rate of the wheel section. The following is a discussion of the sources of these external torques and of their effects upon the spacecraft. In this discussion spin rate refers to the wheel spin rate with respect to the sun, roll rate refers to the rate of change of the spin axis orientation about the solar vector, and pitch rate refers to the rate of change of spin axis orientation about a normal to the solar vector and to the spin axis. Reference to Figure 2-16 will clarify the axes of rotation.

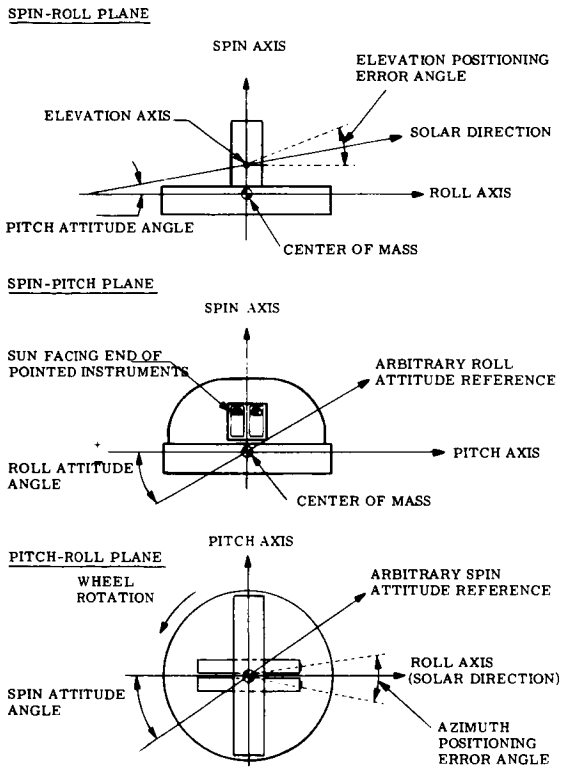


FIGURE 2-16.—OSO I coordinate system.

Aerodynamic Torques

The OSO spacecraft, although in a relatively high orbit, was still subjected to aerodynamic forces. These forces are probably more precisely referred to as molecular forces since the mean free path of the molecules is large compared to the dimensions of the spacecraft. These forces acted to roll the spacecraft about the solar vector and to precess the spacecraft spin axis out of the plane normal to the solar vector.

The OSO upper structure was pointed at the sun throughout the orbit day and rotated with the wheel during the orbit night. The spacecraft wheel rotated throughout the orbit and inspection of Figure 2-17 shows that the x and y components of aerodynamic drag average to zero over an orbit. The usual frictional shear forces associated with a viscous medium were not present, so there existed only a negligibly small nonsymmetrical energy loss due to molecular transport caused by molecules being

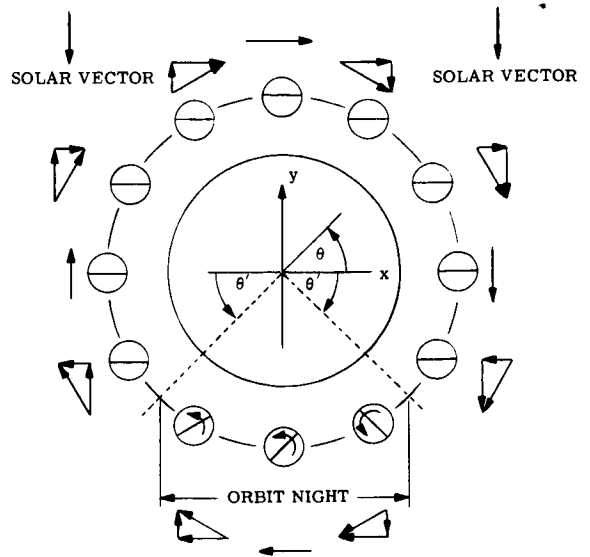


FIGURE 2-17.—Aerodynamic forces during an orbit.

trapped at the surface and escaping a short time later.

The y -component of the force acting upon the upper structure averaged to zero during both the orbit day and orbit night so that there existed no net roll producing torque.

Although the x -components acting upon the upper structure averaged to zero over an orbit, these forces acted upon different configurations: a rotating upper structure during the orbit night, and an upper structure directed at the solar vector during the orbit day.

This net force acting upon the lever arm from the spacecraft center of gravity to the center of pressure of the upper structure produced a torque about the solar vector which acted to precess the spin axis away from the desired pitch position, i.e., normal to the solar vector.

The upper structure was approximately in the form of a semi-circular flat plate. A centroid of area, $\bar{z}$, is defined as

$$\bar{z} = \frac{\int_{A'} z dA'}{\int_{A'} dA'} \quad (2.153)$$

During orbit night, the semi-circular flatplate rotated about the radius perpendicular to the diameter.

Viewed from the direction of the spacecraft velocity vector, the semicircular plate looks like an ellipse with a varying minor axis. The equation of this ellipse is

$$\frac{y^2}{r^2 \cos^2 \beta} + \frac{z^2}{r^2} = 1 \quad z \geq 0 \quad (2.154)$$

where β is the angle from the spacecraft velocity vector to the flatplate normal. The integrands of Equation (2.153) become

$$dA' = \cos \beta \sqrt{r^2 - z^2} dz \quad (2.155)$$

and

$$zdA' = \cos \beta (z \sqrt{r^2 - z^2} dz) \quad (2.156)$$

The integrations over the area give

$$A' = \int_{A'} dA' = \frac{\pi r^2}{2} \cos \beta = A \cos \beta$$

and

$$\int_{A'} zdA' = \frac{2}{3} r^3 \cos \beta$$

and the centroid is

$$z = \frac{4}{3\pi} r \quad (2.157)$$

and is independent of the angle, β .

The average value of the area of the spinning upper structure in this application is

$$\begin{aligned} \bar{A} &= A \left(\frac{1}{\pi/2} \int_0^{\pi/2} \cos \beta d\beta \right) \\ &= \frac{2}{\pi} A \end{aligned}$$

The torque upon the spacecraft during the orbit night is

$$\begin{aligned} T_N &= C_D \frac{\rho}{2} V^2 \bar{A} (\bar{z} + d) \int_{\pi+\theta'}^{2\pi-\theta'} \sin \theta d\theta \\ &= -\rho V^2 A (\bar{z} + d) \left(\frac{4}{\pi} \cos \theta' \right) \end{aligned} \quad (2.158)$$

where

d = distance from the lower edge of the upper structure to the spacecraft center of gravity, in feet

V = spacecraft velocity, in ft/sec

A = upper structure area, in ft²

C_D = coefficient of drag, which is assumed to have the value, $C_D = 2$

and

$$\theta' = \arccos \left(\frac{R}{R+h} \right) \quad (2.159)$$

where

R = radius of the earth

and

h = height of spacecraft above the earth

During the sunlight portion of the orbit, the upper structure area presented to the aerodynamic force is

$$A_s = A \cos \theta \quad (-\theta' \leq \theta \leq \pi/2)$$

and

$$A_s = -A \cos \theta \quad (\pi/2 \leq \theta \leq \pi + \theta')$$

The drag in the x -direction is

$$D_x = D \sin \theta$$

and the torque on the upper structure during the orbit day is

$$\begin{aligned} T_D &= \rho V^2 A (\bar{z} + d) \int_{-\theta'}^{\pi/2} 2 \sin \theta \cos \theta d\theta \\ &= \rho V^2 A (\bar{z} + d) \sin^2 \theta' \\ &= \rho V^2 A (\bar{z} + d) (1 - \sin^2 \theta') \end{aligned} \quad (2.160)$$

The net torque about the solar vector during an orbit is

$$\begin{aligned} T &= T_N + T_D \\ &= \rho V^2 A (\bar{z} + d) \left[-\frac{4}{\pi} \cos \theta' + (1 - \sin^2 \theta') \right] \end{aligned} \quad (2.161)$$

In order to evaluate the net torque, an altitude of 300 nautical miles = 346 statute miles = 556 kilometers was assumed. The air density, from damping calculations on Sputnik I, was

$$\rho_h = (4 \times 10^{-13}) e^{-\left(\frac{h-234}{50}\right)} \text{ gm/cc} \quad (2.162)$$

where

ρ = density at altitude h

h = altitude (kilometers)

The density at $h=556$ kilometers is

$$\begin{aligned}\rho_{556} &= (4 \times 10^{-13}) e^{-6.44} \\ &= 6.4 \times 10^{-16} \frac{\text{gm}}{\text{cc}} = 1.24 \times 10^{-15} \frac{\text{slug}}{\text{ft}^3}\end{aligned}$$

Now

$$\begin{aligned}\rho_{556} V^2 A &= (1.24)(6.125)(5.275 \times 10^{-7}) \text{ lb}_f \\ &= 4.01 \times 10^{-6} \text{ lb}_f\end{aligned}$$

and for

$$\begin{aligned}\theta' &\cong 23^\circ \\ \bar{z} + d &= 1 \text{ ft}\end{aligned}$$

the torque is

$$T = (4 \times 10^{-6})(-0.325) = -1.3 \times 10^{-6} \text{ lb}_f\text{-ft}$$

and the pitch precession is

$$\alpha = \frac{I}{I_s \omega} t \quad (2.163)$$

$$\begin{aligned}|\alpha| &= \frac{(1.3 \times 10^{-6} \text{ lb}_f\text{-ft}) \left(3600 \times 24 \frac{\text{sec}}{\text{day}} \right) (57.3 \text{ deg})}{(26 \text{ lb}_f\text{-ft-sec}^2) \left(2\pi \frac{\text{rad}}{\text{sec}} \right) \text{ rad}} \\ &= 0.04 \frac{\text{deg}}{\text{day}}\end{aligned}$$

The pointed experiments presented an area approximately $\frac{1}{4}$ that of the upper structure at about the same centroidal distance from the spacecraft center of gravity and thus contributed an additional $0.01 \frac{\text{degree}}{\text{day}}$ pitch precession making the total expected about $0.05 \frac{\text{degree}}{\text{day}}$.

Gravity Gradient Torques

Since the gravitational attraction varies inversely as the square of the distance from the center of the earth, the gravitational force acting on those parts of the spacecraft farther away

from the earth was less than the force on those parts nearer the earth. This difference produced a torque. The magnitude of this torque largely determined the amount of control gas required for correction of externally caused motions.

Figure 2-18 shows the coordinate system used in this analysis. Axes x , y , and z form a geocentric ecliptic system with the sun always along the x -axis. Unit vectors in this system are $\mathbf{i}$, $\mathbf{j}$, $\mathbf{k}$. The u , v , w system is a geocentric equatorial system with the u -axis along the equatorial-orbital line of nodes. The x' , y' , z' system is a rotating geocentric system with the spacecraft always on the x' -axis. The unit vectors are $\mathbf{i}'$, $\mathbf{j}'$, $\mathbf{k}'$.

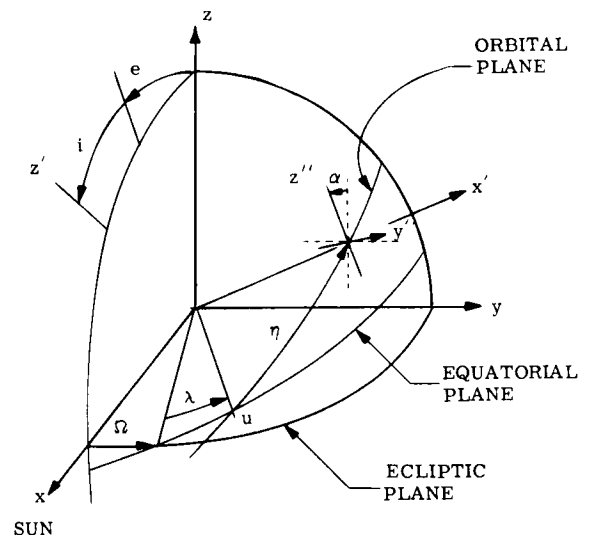


FIGURE 2-18.—Coordinate system.

The OSO I spin axis is stabilized in a position nearly perpendicular to the solar vector. The spin axis is not controlled in the plane perpendicular to the solar vector. The x'' , y'' , z'' system is a spacecraft orientation system with the positive z'' -axis along the positive direction of the spin axis. The unit vectors in this system are $\mathbf{i}''$, $\mathbf{j}''$, $\mathbf{k}''$.

The matrix transformations are

$$\begin{pmatrix} u \\ v \\ w \end{pmatrix} = \begin{pmatrix} \lambda_c \Omega_c - \lambda_s e_c \Omega_s & \lambda_c \Omega_s + \lambda_s e_c \Omega_c & \lambda_s e_s \\ -\lambda_s \Omega_c - \lambda_c e_c \Omega_s & -\lambda_s \Omega_s + \lambda_c e_c \Omega_c & \lambda_c e_s \\ e_s \Omega_s & -e_s \Omega_c & e_c \end{pmatrix} \begin{pmatrix} x \\ y \\ z \end{pmatrix} = (A_{ij}) \begin{pmatrix} x \\ y \\ z \end{pmatrix} \quad (2.164)$$

where

$$\lambda_c = \cos e, \quad e_s = \sin e, \text{ etc.}$$

$$\begin{pmatrix} x' \\ y' \\ z' \end{pmatrix} = \begin{pmatrix} \eta_c & i_c \eta_s & i_s \eta_s \\ -\eta_s & i_c \eta_c & i_s \eta_c \\ 0 & -i_s & i_c \end{pmatrix} \begin{pmatrix} u \\ v \\ w \end{pmatrix} = (B_{ij}) \begin{pmatrix} u \\ v \\ w \end{pmatrix} \quad (2.165)$$

and

$$\begin{pmatrix} x'' \\ y'' \\ z'' \end{pmatrix} = \begin{pmatrix} 1 & 0 & 0 \\ 0 & \alpha_c & \alpha_s \\ 0 & -\alpha_s & \alpha_c \end{pmatrix} \begin{pmatrix} x \\ y \\ z \end{pmatrix} = (c) \begin{pmatrix} x \\ y \\ z \end{pmatrix} \quad (2.166)$$

The unit vector, $\mathbf{i}'$, in the direction of the spacecraft is given by

$$\begin{aligned} \mathbf{i}' &= \sum_n (B_{ij})(A_{jn}) \\ &= \{ \eta_c(\lambda_c \Omega_c - \lambda_s e_c \Omega_s) + \eta_s[i_s e_s \Omega_s \\ &\quad - i_c(\lambda_s \Omega_c + \lambda_c e_c \Omega_s)] \} \mathbf{i} \\ &\quad + \{ \eta_c(\lambda_c \Omega_s + \lambda_s e_c \Omega_c) + \eta_s[-i_s e_s \Omega_c \\ &\quad - i_c(\lambda_s \Omega_s - \lambda_c e_c \Omega_c)] \} \mathbf{j} \\ &\quad + [\eta_c(\lambda_s e_s) + \eta_s(i_s e_c + i_c \lambda_c e_s)] \mathbf{k} \\ &= \left. \begin{aligned} &(A_1 \cos \eta + A_2 \sin \eta) \mathbf{i} \\ &+ (B_1 \cos \eta + B_2 \sin \eta) \mathbf{j} \\ &+ (C_1 \cos \eta + C_2 \sin \eta) \mathbf{k} \end{aligned} \right\} \quad (2.167) \end{aligned}$$

where

$$\left. \begin{aligned} A_1 &= \cos \lambda \cos \Omega - \sin \lambda \cos e \sin \Omega \\ A_2 &= \sin i \sin e \sin \Omega - \cos i (\sin \lambda \cos \Omega \\ &\quad + \cos \lambda \cos e \sin \Omega) \\ B_1 &= \cos \lambda \sin \Omega + \sin \lambda \cos e \cos \Omega \\ B_2 &= -\sin i \sin e \cos \Omega - \cos i \\ &\quad (\sin \lambda \sin \Omega - \cos \lambda \cos e \cos \Omega) \\ C_1 &= \sin \lambda \sin e \\ C_2 &= \sin i \cos e + \cos i \cos \lambda \sin e \end{aligned} \right\} \quad (2.168)$$

Now the torque on the spacecraft due to the gravitational field is

$$\mathbf{L} = - \int \left(\mathbf{r}' \times \frac{g R_c^2}{R^3} \mathbf{R} \right) \rho dv \quad (2.169)$$

where

$\mathbf{L}$ = vector torque on spacecraft
 $\mathbf{r}'$ = vector from spacecraft c.m. to dv
 $\mathbf{p}$ = vector from earth's center to dv
 ρ = density of volume element, dv
 R_c = radius of earth
 g = acceleration of gravity at earth's surface

Let

$$\mathbf{R} = \mathbf{R}_c + \mathbf{r}' \quad (2.170)$$

Where $\mathbf{R}_c$ is a vector from the center of earth to the center of mass of the spacecraft.

The wheel of the spacecraft is continuously rotating, giving the spacecraft gyroscopic stability. The sail with its solar cells and experiments is directed at the sun during the orbit day and, because of bearing friction, rotates with the wheel during the orbit night. The spin axis is maintained nearly perpendicular to the solar vector by means of a pitch gas system.

Let

$$\mathbf{r}' = \mathbf{c}_i - \mathbf{r}_i \quad (2.171)$$

where

$\mathbf{c}_i$ is a vector from the spacecraft center of mass to the center of mass of the part and $\mathbf{r}_i$ is a vector from the center of mass of the part to dv and is expressed in a principal axis system fixed to the part.

Then

$$\begin{aligned} \mathbf{L} &= -gR_c^2 \int_v R^{-3} [(\mathbf{c}_i + \mathbf{r}_i) \times (\mathbf{R}_c + \mathbf{c}_i + \mathbf{r}_i)] \rho dv \\ &= -gR_c^2 \int_v R^{-3} [(\mathbf{c}_i + \mathbf{r}_i) \times \mathbf{R}_c + (\mathbf{c}_i + \mathbf{r}_i) \times (\mathbf{c}_i + \mathbf{r}_i)] \rho dv \\ &= -gR_c^2 \int_v R^{-3} [(\mathbf{c}_i + \mathbf{r}_i) \times \mathbf{R}_c] \rho dv \quad (2.172) \end{aligned}$$

since

$$(\mathbf{c}_i + \mathbf{r}_i) \times (\mathbf{c}_i + \mathbf{r}_i) = 0$$

The quantity R^{-3} in the integrand may be approximated as follows:

$$\begin{aligned} R^{-3} &= (\mathbf{R} \cdot \mathbf{R})^{-3/2} = [(\mathbf{R}_c + \mathbf{c}_i + \mathbf{r}_i) \cdot (\mathbf{R}_c + \mathbf{c}_i + \mathbf{r}_i)]^{-3/2} \\ &= [\mathbf{R}_c \cdot \mathbf{R}_c + 2\mathbf{R}_c \cdot (\mathbf{c}_i + \mathbf{r}_i) + (\mathbf{c}_i + \mathbf{r}_i)^2]^{-3/2} \\ &= R_c^{-3} \left[\frac{\mathbf{R}_c \cdot \mathbf{R}_c}{R_c^2} + \frac{2\mathbf{R}_c \cdot (\mathbf{c}_i + \mathbf{r}_i)}{R_c^2} + \left(\frac{\mathbf{c}_i + \mathbf{r}_i}{R_c} \right)^2 \right]^{-3/2} \\ &\cong R_c^{-3} \left[\frac{\mathbf{R}_c \cdot \mathbf{R}_c}{R_c^2} + \frac{2\mathbf{R}_c \cdot (\mathbf{c}_i + \mathbf{r}_i)}{R_c^2} \right]^{-3/2} \quad (2.173) \end{aligned}$$

for a spacecraft of small size. Now

$$\frac{\mathbf{R}_c}{R_c} = \mathbf{i}'$$

$$\mathbf{i}' \cdot \mathbf{i}' = 1$$

Therefore, the quantity R^{-3} may be expressed

$$R^{-3} \cong R_c^{-3} \left[1 + \frac{2\mathbf{i}' \cdot (\mathbf{c}_i + \mathbf{r}_i)}{R_c} \right]^{-3/2} R_c^{-3} \left[1 - \frac{3\mathbf{i}' \cdot (\mathbf{c}_i + \mathbf{r}_i)}{R_c} \right] \quad (2.174)$$

and

$$\begin{aligned} \mathbf{L} &\cong \frac{gR_c^2}{R_c^3} \int_v [\mathbf{R}_c \times (\mathbf{c}_i + \mathbf{r}_i)] \left[1 - \frac{3\mathbf{i}' \cdot (\mathbf{c}_i + \mathbf{r}_i)}{R_c} \right] \rho dv \\ &= \frac{gR_c^2}{R_c} \left\{ R_c \times \left(\mathbf{c}_i \int_v \rho dv + \int_v \mathbf{r}_i \rho dv \right) - 3 \int_v [\mathbf{i}' \times (\mathbf{c}_i + \mathbf{r}_i)] [\mathbf{i}' \cdot (\mathbf{c}_i + \mathbf{r}_i)] \rho dv \right\} \quad (2.175) \end{aligned}$$

Now

$$\int_v \mathbf{r}_i \rho dv = 0$$

by definition of the center of mass of the part, and

$$\int_v \rho dv = m_i$$

where m_i is the mass of the i^{th} part. Furthermore, we have

$$\sum_i (\mathbf{R}_c \times \mathbf{c}_i) \int_v \rho dv = \mathbf{R}_c \times (m_1 \mathbf{c}_1 + m_2 \mathbf{c}_2) = 0$$

by definition of the center of mass of the parts.

The torque on the spacecraft may be written

$$\mathbf{L} \cong -3\omega^2 \int_v [\mathbf{i}' \times (\mathbf{c}_i + \mathbf{r}_i)] [\mathbf{i}' \cdot (\mathbf{c}_i + \mathbf{r}_i)] \rho dv \quad (2.176)$$

since gR_c^2/R_c^3 is the square of the angular velocity for a circular orbit.

The spacecraft rotates about its spin axis and the equation of transformation between the x_i''', y_i''', z_i''' axes fixed in the spacecraft parts and the x'', y'', z'' system is

$$\begin{pmatrix} x'' \\ y'' \\ z'' \end{pmatrix} = \begin{pmatrix} \beta_c^i & \beta_s^i & 0 \\ -\beta_s^i & \beta_c^i & 0 \\ 0 & 0 & 1 \end{pmatrix} \begin{pmatrix} x''' \\ y''' \\ z''' \end{pmatrix} \quad (2.177)$$

where

β^i can have the value

β^s = sail position angle

or

β^w = wheel position angle

The positional vectors $\mathbf{r}_i$ and $\mathbf{c}_i$ are

$$\begin{aligned} \mathbf{r}_i &= (x''' \beta_c^i - y''' \beta_s^i) \mathbf{i}'' \\ &\quad + (x''' \beta_s^i + y''' \beta_c^i) \mathbf{j}'' + z''' \mathbf{k}'' \end{aligned}$$

and

$$\mathbf{c}_i = c_i \mathbf{k}''$$

Adding these last two equations yields

$$\mathbf{c}_t + \mathbf{r}_t = (x''' \beta_c^t - y''' \beta_s^t) \mathbf{i}'' + (x''' \beta_s^t + y''' \beta_c^t) \mathbf{j}'' + (z''' + c_t) \mathbf{k}'' \quad (2.178)$$

The spacecraft positional vector $\mathbf{i}'$, may be expressed in the $\mathbf{i}''$, $\mathbf{j}''$, $\mathbf{k}''$ system as

$$\begin{aligned} \mathbf{i}' &= (\mathbf{i}' \cdot \mathbf{i}) \mathbf{i} + (\mathbf{i}' \cdot \mathbf{j}) \mathbf{j} + (\mathbf{i}' \cdot \mathbf{k}) \mathbf{k} \\ &= a \mathbf{i} + b \mathbf{j} + c \mathbf{k} \\ &= a \mathbf{i}'' + (b \alpha_c + c \alpha_s) \mathbf{j}'' + (-b \alpha_s + c \alpha_c) \mathbf{k}'' \\ &= a \mathbf{i}'' + J \mathbf{j}'' + K \mathbf{k}'' \end{aligned} \quad (2.179)$$

where

$$\begin{aligned} b &= (\mathbf{i}' \cdot \mathbf{j}) \\ c &= (\mathbf{i}' \cdot \mathbf{k}) \\ a &= (\mathbf{i}' \cdot \mathbf{i}) \\ J &= b \alpha_c + c \alpha_s \\ K &= -b \alpha_s + c \alpha_c \end{aligned}$$

The dot product of $\mathbf{i}'$ and $(\mathbf{c}_t + \mathbf{r}_t)$ is

$$\mathbf{i}' \cdot (\mathbf{c}_t + \mathbf{r}_t) = (a \beta_c^t + J \beta_s^t) x''' + (-a \beta_s^t + J \beta_c^t) y''' + K(z''' + c_t)$$

and the cross-product of $\mathbf{i}'$ and $(\mathbf{c}_t + \mathbf{r}_t)$ is

$$\begin{aligned} \mathbf{i}' \times (\mathbf{c}_t + \mathbf{r}_t) &= \mathbf{i}'' [-K \beta_s^t x''' - K \beta_c^t y''' + J(z''' + c)] + \mathbf{j}'' [K \beta_c^t x''' \\ &\quad - K \beta_s^t y''' - a(z''' + c)] + \mathbf{k}'' [(a \beta_s^t - J \beta_c^t) x''' + (a \beta_c^t + J \beta_s^t) y'''] \end{aligned}$$

The product of the latter two equations is

$$\begin{aligned} [\mathbf{i}' \times (\mathbf{c}_t + \mathbf{r}_t)] [\mathbf{i}' \cdot (\mathbf{c}_t + \mathbf{r}_t)] &= \mathbf{i}'' K [(-a \beta_s^t \beta_c^t - J \beta_s^t) x'''' + (a \beta_s^t \beta_c^t - J \beta_c^t) y'''' + J(z''' + c_t)^2 + \dots] \\ &\quad + \mathbf{j}'' K [(a \beta_c^t + J \beta_s^t) x'''' + (a \beta_s^t - J \beta_c^t) y'''' - a(z''' + c_t)^2 + \dots] \\ &\quad + \mathbf{k}'' [(a \beta_c^t + J \beta_s^t)(a \beta_s^t - J \beta_c^t)(x'''' - y''') + \dots] \\ &= \mathbf{i}'' K \{ a \beta_s^t \beta_c^t (y'''' - x''') + J[(z''' + c_t)^2 - (\beta_s^t x'''' + \beta_c^t y''')] \} + \dots \\ &\quad + \mathbf{j}'' K \{ J \beta_s^t \beta_c^t (x'''' - y''') - a[(z''' + c_t)^2 - (\beta_c^t x'''' + \beta_s^t y''')] \} + \dots \\ &\quad + \mathbf{k}'' [(a \beta_c^t + J \beta_s^t)(a \beta_s^t - J \beta_c^t)(x'''' - y''') + \dots] \end{aligned}$$

where crossproduct terms have not been written.

Because of the choice of axes, $\mathbf{L}$ simplifies upon integration to

$$\begin{aligned} L &= -3\omega^2 (\mathbf{i}'' \cdot \{ K \} \{ a \beta_s^t \beta_c^t (I_z^t - I_y^t) + J[\beta_s^t (I_z^t - I_z^t) + \beta_c^t (I_y^t - I_z^t) + c_t^2 m_t] \} + \mathbf{j}'' \cdot \{ K \} \{ J \beta_s^t \beta_c^t (I_y^t - I_z^t) \\ &\quad - a[\beta_c^t (I_z^t - I_z^t) + \beta_s^t (I_y^t - I_z^t) + c_t^2 m_t] \} + \mathbf{k}'' \cdot [(a \beta_c^t + J \beta_s^t)(a \beta_s^t - J \beta_c^t)(I_y^t - I_z^t)]) \\ &= -3\omega^2 (\mathbf{i}'' \cdot \{ (a \alpha_c - a b \alpha_s) \beta_s^t \beta_c^t (I_z^t - I_y^t) + [(c^2 - b^2) \alpha_s \alpha_c + b c (2 \alpha)_c] [\beta_s^t (I_z^t - I_z^t) + \beta_c^t (I_y^t - I_z^t) + c_t^2 m_t] \} \\ &\quad + \mathbf{j}'' \cdot \{ [(c^2 - b^2) \alpha_s \alpha_c + b c (2 \alpha)_c] \beta_s^t \beta_c^t (I_y^t - I_z^t) + (a \alpha_c - a b \alpha_s) [\beta_c^t (I_z^t - I_z^t) + \beta_s^t (I_y^t - I_z^t) + c_t^2 m_t] \} \\ &\quad + \mathbf{k}'' \cdot \{ [a^2 - (b \alpha_c + c \alpha_s)^2] \beta_s^t \beta_c^t - (a b \alpha_c + a c \alpha_s) (2 \beta^t)_c \} (I_y^t - I_z^t)) \end{aligned}$$

where the superscripts on the moments of inertia indicate that the moments of inertia can be those of the sail or of the wheel. Now let

L_w = torque due to the wheel (orbit day and orbit night)

L_{SD} = torque due to the sail (orbit day only)

L_{SN} = torque due to the sail (orbit night only)

Due to the symmetry of the wheel about the spin axis, $I_x^w = I_y^w$ and the equation for L reduces to

$$L_w = 3\omega^2[(I_s - I_T) - c_s^2 m_s] \cdot \{ \mathbf{i}''(c^2 - b^2)\alpha_s \alpha_c + bc(2\alpha)_c + \mathbf{j}''(ab\alpha_s - ac\alpha_c) \} \quad (2.180)$$

where

I_s = moment of inertia of the wheel about the spin axis

$I_T = I_x^w = I_y^w$ = transverse moment of inertia of the wheel.

During the orbit day, when the sail is directed at the sun and $\beta^s = 0$. The torque due to the sail (orbit day) is

$$\begin{aligned} L_{SD} = 3\omega^2 \{ \mathbf{i}''[(c^2 - b^2)\alpha_s \alpha_c + bc(2\alpha)_c] \\ [I_z^s - I_y^s - c_s^2 m_s] + \mathbf{j}''(ab\alpha_s - ac\alpha_c) \\ (I_z^s - I_x^s - c_s^2 m_s) + \mathbf{k}''(ab\alpha_c \\ + ac\alpha_s)(I_y^s - I_z^s) \} \quad (2.181) \end{aligned}$$

where

I_z^s = sail moment of inertia about the spin axis

I_z^s = sail moment of inertia about the solar vector directed through the sail center of mass

I_y^s = sail moment of inertia about the perpendicular to the solar vector and the spin axis and through the sail center of mass

m_s = mass of the sail

c_s = distance between center of masses of spacecraft and sail.

During the orbit night, servo control power was removed and the sail rotated with the wheel because of bearing, slip ring, and motor brush friction. The torque on the sail during orbit night is

$$\begin{aligned} L'_{SN} = \left(-3\omega^2 \{ \mathbf{i}''(ac\alpha_c - ab\alpha_s)\beta_s^s \beta_c^s (I_z^s - I_y^s) + [(c^2 - b^2)\alpha_s \alpha_c + bc(2\alpha)_c][\beta_s^s (I_z^s - I_x^s) + \beta_c^s (I_y^s - I_z^s) \right. \\ \left. + c_s^2 m_s] \} + \mathbf{j}'' \{ [(c^2 - b^2)\alpha_s \alpha_c + bc(2\alpha)_c]\beta_s^s \beta_c^s (I_y^s - I_z^s) + (ac\alpha_c - ab\alpha_s)[\beta_c^s (I_z^s - I_x^s) \right. \\ \left. + \beta_s^s (I_y^s - I_z^s) + c_s^2 m_s] \} + \mathbf{k}'' \{ [a^2 - (b\alpha_c + c\alpha_s)^2]\beta_s^s \beta_c^s - (ab\alpha_c + ac\alpha_s)(2\beta^s)_c \} (I_y^s - I_z^s) \right) \end{aligned}$$

Averaged over one sail rotation, the torque is

$$L_{SN} = 3\omega^2 \{ [I_z^s - \frac{1}{2}(I_x^s + I_y^s) - c_s^2 m_s] [(c^2 - b^2)\alpha_s \alpha_c + bc(2\alpha)_c] + \mathbf{j}[ab\alpha_s - ac\alpha_c] \} \quad (2.182)$$

In order to obtain the angular rates imparted to the spacecraft by the gravity gradient torques:

$$c^2 - b^2 \leq c^2 + b^2 \leq 1$$

$$bc \leq \frac{\sqrt{2}}{2} \cdot \frac{\sqrt{2}}{2} = \frac{1}{2}$$

$$ac \leq \frac{\sqrt{2}}{2} \cdot \frac{\sqrt{2}}{2} = \frac{1}{2}$$

The torque about the $\mathbf{i}''$ -axis results in a pitch of the spacecraft spin axis toward or away from the solar vector. As a function of the spacecraft roll position

$$(c^2 - b^2)\alpha_s \alpha_c + bc(2\alpha)_c = (1)(0) + \frac{1}{2} \text{ for } \alpha = 0^\circ$$

$$= (1) \left(\frac{1}{2} \right) + 0 = \frac{1}{2} \text{ for } \alpha = 45^\circ$$

$$= (1)(0) + \frac{1}{2}(1) = \frac{1}{2} \text{ for } \alpha = 90^\circ$$

The torque about the j'' -axis results in a roll about the solar vector and

$$ab\alpha_s - ac\alpha_c \leq |ab\alpha_s| + |ac\alpha_c| = \left(\frac{1}{2} \right) (0) + \frac{1}{2} (1) = \frac{1}{2} \text{ for } \alpha = 0^\circ$$

$$= \frac{1}{2} \left(\frac{\sqrt{2}}{2} \right) + \frac{1}{2} \left(\frac{\sqrt{2}}{2} \right) = \frac{\sqrt{2}}{2} \text{ for } \alpha = 45^\circ$$

$$= \frac{1}{2} (1) + \frac{1}{2} (0) = \frac{1}{2} \text{ for } \alpha = 90^\circ$$

The torque about the spin axis was applied to the sail only during the sunlight portion of the orbit, which averaged about $\frac{2}{3}$ of the total orbital period. The servo controlling movement about the spacecraft spin axis resisted the

torque due to gravity gradient by driving against the wheel. The action resulted in a spin-up or spin-down of the wheel (change in angular rate). Now

$$ab\alpha_c + ac\alpha_s \leq |ab\alpha_c| + |ac\alpha_s| = \frac{1}{2} (1) + \frac{1}{2} (0) = \frac{1}{2} \text{ for } \alpha = 0^\circ$$

$$= \frac{1}{2} \left(\frac{\sqrt{2}}{2} \right) + \frac{1}{2} \left(\frac{\sqrt{2}}{2} \right) = \frac{\sqrt{2}}{2} \text{ for } \alpha = 45^\circ$$

$$= \frac{1}{2} (0) + \frac{1}{2} (1) = \frac{1}{2} \text{ for } \alpha = 90^\circ$$

Therefore

$$(c^2 - b^2)\alpha_s + bc(2\alpha)_c = \frac{1}{2}$$

$$(ab\alpha_s - ac\alpha_c) = \frac{1}{2} \sqrt{2}$$

$$(ab\alpha_c + ac\alpha_s) = \frac{1}{2} \sqrt{2}$$

will be used.

For the OSO I spacecraft, the following parameter values were determined:

$$I_z^s = 1.59 \text{ slug-ft}^2$$

$$I_y^s = 2.46 \text{ slug-ft}^2$$

$$I_z^r = 2.70 \text{ slug-ft}^2$$

$$I_s = 23.32 \text{ slug-ft}^2$$

$$I_T = 16.42 \text{ slug-ft}^2$$

$$m_s = 4.12 \text{ slugs}$$

$$m_w = 9.65 \text{ slugs}$$

$$c_s = +0.523 \text{ ft}$$

$$c_w = -0.222 \text{ ft}$$

and

$$I_s - I_T - c_w^2 m_w = 6.42 \text{ slug-ft}^2$$

$$I_z^s - I_y^s - c_s^2 m_s = -0.884 \text{ slug-ft}^2$$

$$I_z^s - I_z^r - c_s^2 m_s = -0.014 \text{ slug-ft}^2$$

$$I_y^s - I_z^s = 0.87 \text{ slug-ft}^2$$

$$I_z^s - \frac{1}{2}(I_z^s + I_y^s) - c_s^2 m_s = -0.449 \text{ slug-ft}^2$$

The torque on the wheel is

$$\mathbf{L}_w = \frac{3}{2} \left(\frac{2\pi}{96 \times 60} \right)^2 (6.42)(\mathbf{i}'' + \sqrt{2}\mathbf{j}'') = 1.14 \times 10^{-5} (\mathbf{i}'' + \sqrt{2}\mathbf{j}'') \text{ lb-ft} \quad (2.183)$$

The torque on the sail during the sunlight portion of the orbit is

$$\mathbf{L}_{SN} = \frac{2}{3} \left(\frac{3\omega^2}{2} \right) (884\mathbf{i}'' + 0.014\mathbf{j}'' + 0.87\mathbf{k}'') = 10^{-5} (1.054\mathbf{i}'' + 0.0236\mathbf{j}'' + 1.47\mathbf{k}'') \text{ lb-ft} \quad (2.184)$$

The torque on the sail during the night portion of the orbit is

$$\mathbf{L}_{SN} = \frac{1}{3} \left(\frac{3\omega^2}{2} \right) (0.449)(\mathbf{i}'' + \sqrt{2}\mathbf{j}'') \left. \vphantom{\frac{1}{3} \left(\frac{3\omega^2}{2} \right)} \right\} (2.185) \\ = 0.0267 \times 10^{-5} (\mathbf{i}'' + \sqrt{2}\mathbf{j}'')$$

and the total torque is

$$\mathbf{L} = 10^{-5} (1.265\mathbf{i}'' + 1.64\mathbf{j}'' + 0.1147\mathbf{k}'') \quad (2.186)$$

The $\mathbf{k}$ component of the torque changed the spin rate of the wheel by $5.91 \times 10^{-3} \left(\frac{\text{rad}}{\text{sec}} \right) / \text{day}$ or a 5 percent change in nominal wheel rate in 26 days. Thus the spin rate time constant was 312 days.

From the formula

$$\theta = \frac{T}{I_s \omega} \quad (2.187)$$

which gives the gyroscopic precession as a function of the applied torque, the wheel moment of inertia, and the wheel angular rate, the pitch and roll rates were determined to be 0.85 deg/day and 1.1 deg/day, respectively.

Radiation Pressure Torques

Suppose a stream of radiation in a vacuum falls normally on the surface of a body. According to Einstein, we know that

$$m = \frac{E}{c^2} \quad (2.188)$$

where

m = mass/volume

E = mean energy/volume

and

c = velocity of light

Then the momentum per volume, p , is given by

$$p = cm = \frac{E}{c}$$

If the radiation impinges upon and is absorbed by the surface, the change in momentum per unit area in one second is

$$cp = E$$

or since a change in momentum per unit area is a force per unit area, the radiation pressure for absorption, P_a , is

$$P_a = E$$

If the incident beam is reflected from the surface, the change in momentum per unit area in one second is

$$2cp = 2E$$

and the radiation pressure for reflection, P_r , is

$$P_r = 2E$$

Suppose a part, call it α , of the incident radiation is absorbed and another part, ρ , is reflected. Now, for an opaque body

$$\alpha + \rho = 1$$

Then the pressure due to the absorbed radiation, $\bar{p}_a$, is

$$\bar{p}_a = \alpha E$$

and the pressure due to the reflected radiation, $\bar{p}_r$, is

$$\bar{p}_r = 2\rho E = 2(1 - \alpha)E$$

and the pressure per unit area becomes

$$P = \bar{p}_a + \bar{p}_r = (2 - \alpha)E \quad (2.189)$$

*For a radiation of mean energy per volume impinging upon the surface with a velocity, v , the energy per unit area in one second, i.e., the solar constant, S , is

$$S = cE$$

and $S = 0.14$ watt/cm². By using the above expression for E , the radiation pressure per unit area for normal incidence may be written

$$P = (2 - \alpha) \frac{S}{c} \quad (2.190)$$

Suppose that the radiation is incident at an angle θ to the surface normal. The energy received per unit area of surface, $E(\theta)$, is

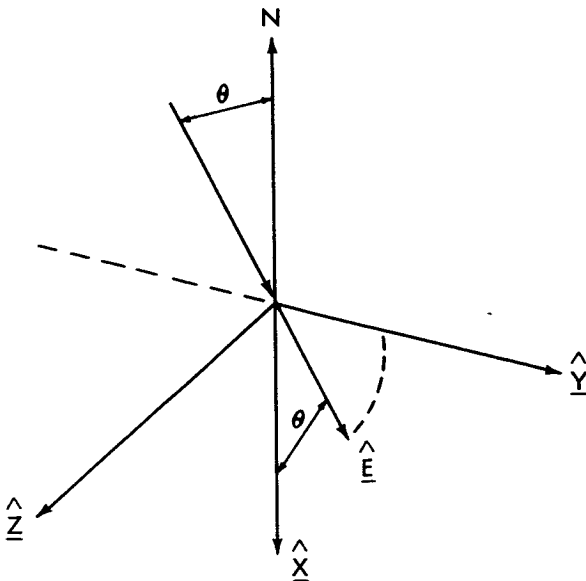
$$E(\theta) = E \cos \theta.$$

For absorption of the incident radiation, the momentum change (or force, $\mathbf{F}_a$) is in the direction of the incident radiation and

$$\mathbf{F}_a(\theta) = E \cos \theta \hat{\mathbf{E}}$$

where $\hat{\mathbf{E}}$ is a unit vector in the direction of the incident energy. The vector $\hat{\mathbf{E}}$ may be resolved in the direction of the inward surface normal and along the surface, and the force per unit area is

$$\mathbf{F}(\theta) = E \cos \theta (\hat{\mathbf{x}} \cos \theta + \hat{\mathbf{y}} \sin \theta) \quad (2.191)$$

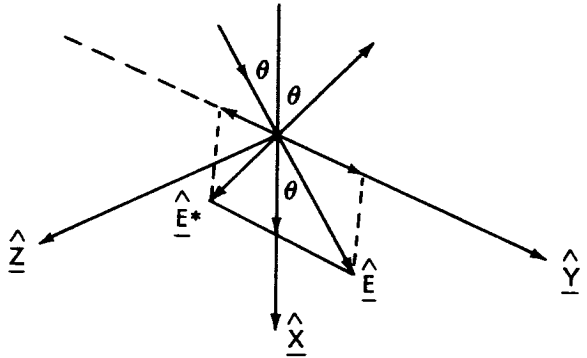


The pressure is the normal component of the force per unit area and

$$P_a(\theta) = \frac{S}{c} \cos^2 \theta \quad (2.192)$$

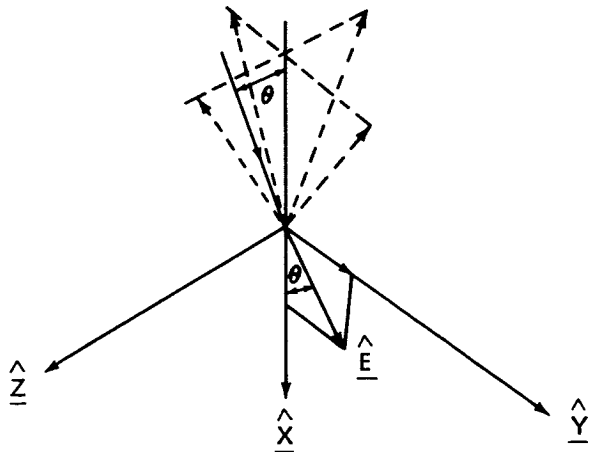
If an incident beam in which the energy density, E , is specularly reflected from a surface at the angle of incidence, θ , the force, $\mathbf{F}_s(\theta) = E \cos \theta (\hat{\mathbf{E}} + \hat{\mathbf{E}}^*)$ where $\hat{\mathbf{E}}^*$ is the image of $\hat{\mathbf{E}}$ in the $\hat{\mathbf{x}}-\hat{\mathbf{z}}$ plane, and

$$\begin{aligned} \mathbf{F}_s(\theta) &= \frac{S}{c} \cos \theta (2\hat{\mathbf{x}} \cos \theta + 0\hat{\mathbf{y}}) \\ &= 2 \frac{S}{c} \cos^2 \theta \hat{\mathbf{x}} \end{aligned}$$



The radiation pressure is

$$P_s(\theta) = 2 \frac{S}{c} \cos^2 \theta \quad (2.193)$$



In the case of the beam, in which the energy density, E , is incident at an angle θ to the surface normal and is diffusely reflected from the surface, it can be seen that y and z components of the reflected energy cancel and that the force in the diffuse case, $F_D(\theta)$, is

$$\mathbf{F}_D(\theta) = \frac{S}{c} \cos \theta \left[x \left(\frac{\gamma \perp}{\gamma} + \cos \theta \right) + y \sin \theta \right] \quad (2.194)$$

where

$\frac{\gamma \perp}{\gamma}$ is the ratio of the force along the normal to the total diffuse force.

Since the surface is diffuse, a surface dS_1 emits radiation in all directions in amounts proportional to its projected area, i.e., $B_1 dS_1 \cos \theta_1$ where B_1 is the brightness. The flux emitted from the element dS_1 , to a second surface element dS_2 at a distance r is $dS_2 \cos \theta_2 / r^2$. Thus, the total flux emitted by dS_1 , that passes through a hemisphere above it is

$$\gamma = \int B_1 dS_1 \cos \theta_1 \left(\frac{dS_2 \cos \theta_2}{r^2} \right) \quad (2.195)$$

If a zone on the hemisphere is taken as dS_2 , then

$$dS_2 = 2\pi r^2 \sin \theta_1 d\theta_1$$

and

$$\begin{aligned} \gamma &= 2\pi B_1 dS_1 \int_0^{\pi/2} \cos \theta_1 \sin \theta_1 d\theta_1 \\ &= B_1 dS_1 \sin^2 \theta_1 \int_0^{\pi/2} \\ &= \pi B_1 dS_1 \end{aligned}$$

The component of the total flux in the direction of the normal is

$$\begin{aligned} \gamma \perp &= \int B_1 dS_1 \cos^2 \theta_1 \left(\frac{dS_2 \cos \theta_2}{r^2} \right) \\ &= 2\pi B_1 dS_1 \int_0^{\pi/2} \cos^2 \theta_1 \sin \theta_1 d\theta_1 \\ &= -\frac{2}{3} \pi B_1 dS_1 \cos^3 \theta_1 \int_0^{\pi/2} \\ &= \frac{2}{3} \pi B_1 dS_1 \end{aligned}$$

and the ratio

$$\frac{\gamma \perp}{\gamma} = \frac{2}{3}$$

Therefore, the force in the diffuse case is

$$\mathbf{F}_D(\theta) = \frac{S}{c} \cos \theta \left[x \left(\frac{2}{3} + \cos \theta \right) + y \sin \theta \right] \quad (2.196)$$

The pressure term due to radiation by the spacecraft involves the factor of the difference in emissivities of the parts of the body facing towards and away from the sun. The difference for the wheel is zero and a small number for the sail, and it is therefore neglected.

The pressure on the sail is

$$\begin{aligned} P_s &= (2 - 0.8)(4.67 \times 10^6)(2.08 \times 10^{-3}) \\ &= 1.17 \times 10^{-7} \text{ lb/ft}^2 \end{aligned}$$

The frontal area of the sail is approximately 5.3 square feet and the force is

$$\begin{aligned} F_s &= 1.17 \times 5.3 \times 10^{-7} \text{ lb} \\ &= 6.2 \times 10^{-7} \text{ lb} \end{aligned}$$

The distance from the spacecraft center of gravity to the c.p. of the radiation pressure is one foot. Therefore, the torque about the pitch axis is

$$T_p = 6.2 \times 10^{-7} \text{ ft/lb}$$

and the roll rate is 0.04 deg/day.

For the wheel, the solar absorptivity, $\alpha_\odot$, is approximately 0.3 and the force on each infinitesimal area is

$$F'_{w,\alpha} = \alpha_\odot \mathbf{F}_\alpha(\theta) = 0.3 \left(\frac{S}{c} \right) \cos \theta \hat{\mathbf{E}} \text{ lb/ft}^2$$

The total force, $F_{w,\alpha}$, on the wheel due to solar absorptivity becomes

$$F_{w,\alpha} = 0.3 \left(\frac{S}{c} \right) A_w \hat{\mathbf{E}}$$

where A_w is the cross-sectional area of the wheel. The cross-sectional area of the wheel is approximately 4.2 square feet, and $F_{w,\alpha}$ becomes

$$\begin{aligned} F_{w,\alpha} &= (0.3)(9.7 \times 10^{-8})(4.2) \\ &= 1.22 \times 10^{-7} \text{ lb} \end{aligned}$$

For the wheel and the case of specular reflection

$$\begin{aligned}
 F_{w,p} &= (0.7) \left(2 \frac{S}{c} \right) \left(\frac{2}{3} A_{cs} \right) \\
 &= \frac{0.7 \times 2 \times 2}{3} \left(\frac{S}{c} \right) A_{cs} \\
 &= \frac{0.7 \times 2 \times 2}{3 \times 0.3} 0.3 \frac{S}{c} A_{cs} \\
 &\cong 3 F_{w,\alpha} \\
 &= 3.66 \times 10^{-7} \text{ lb}
 \end{aligned}$$

For the wheel and the case of diffuse reflection the force is approximately $2.7 F_{w,\alpha}$. Using the larger figure, the total radiation force against the wheel is 4.88×10^{-7} lb, and the roll rate due to the wheel radiation pressure is 0.005 deg/day since the distance between the pressure center and the spacecraft c.g. is 2 inches.

Magnetic Torques

Eddy Currents in Outer Shell of Wheel

For a thin circular cylinder of area resistivity s and radius a spinning on its axis with angular velocity ω in a uniform magnetic field B normal to its axis, the retarding torque per unit length is:†

$$T = -\frac{4\pi\omega s a^3 B^2}{4s^2 + \mu_0^2 a^2 \omega^2} \quad (2.197)$$

or

$$T = -\frac{\pi\omega a^3 \frac{1}{s} B^2}{1 + \frac{\mu_0^2 a^2}{4s^2} \omega^2}$$

or since $s = 1/\sigma d$

$$T = -\frac{\pi\omega a^3 \sigma d B^2}{1 + \frac{1}{4} \mu_0^2 a^2 \sigma^2 d^2 \omega^2} \quad (2.198)$$

Letting $p = \pi a^3 l \sigma d B^2$ and $q = \frac{1}{2} \mu_0 a \sigma d$, then the total torque is

$$T = -\frac{p\omega}{1 + q^2 \omega^2} \quad (2.199)$$

†W. R. Smythe, *Static and Dynamic Electricity*, 2d Edition, London: McGraw-Hill, 1950, Chapter XI, Pg. 417.

For the OSO I spacecraft made with an aluminum shell: $a = 22$ in. $= 0.559$ m; $\sigma = 3.8 \times 10^7$ mho/m; $\mu_0 = 1.25 \times 10^{-6}$ henry/m

$$d_{\max} = 0.090 \text{ in.} = 2.28 \times 10^{-3} \text{ m}$$

$$q = \frac{1}{2} (1.25 \times 10^{-6}) (0.559) (3.8 \times 10^7) (2.28 \times 10^{-3}) = 0.030 \text{ sec}$$

$$\omega_0 = 0.5 \text{ cp} = 3.14/\text{sec} \quad q\omega_0 = 0.094$$

As a first approximation, since $q^2 \omega^2$ is negligible compared to 1, drop the second term in the denominator in equation to get:

$$T = -p\omega \quad (2.200)$$

Also since

$$T = I_0 \dot{\omega},$$

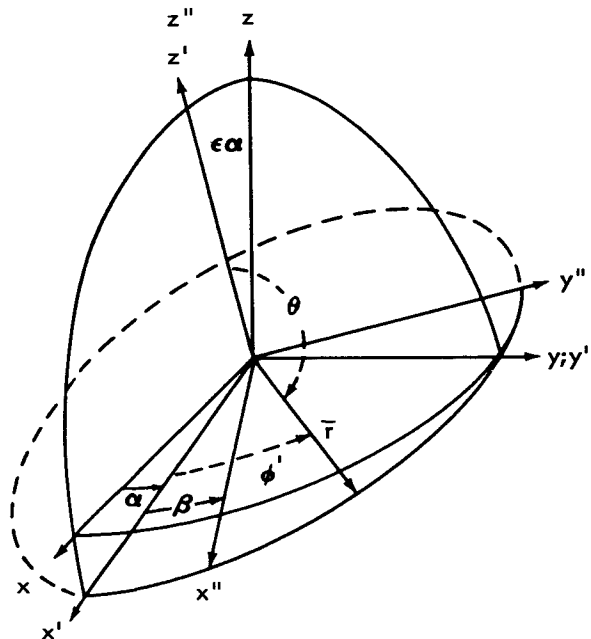
$$\dot{\omega} + \frac{p}{I_0} \omega = 0$$

and solving for the initial condition $\omega(0) = \omega_0$ yields

$$\omega = \omega_0 e^{-t/\tau} \quad (2.201)$$

where $\tau = I_0/p$ is the time constant for the decay rate of the spin.

Since $p = \pi a^3 l \sigma d B^2$ is a function of the normal component of the magnetic field with respect to the spin axis, the average value of the normal component during one orbit must be determined.



For the coordinate system shown in the figure above, the transformation equations are:

$$\begin{pmatrix} x'' \\ y'' \\ z'' \end{pmatrix} = \begin{pmatrix} \cos \beta & \sin \beta & 0 \\ -\sin \beta & \cos \beta & 0 \\ 0 & 0 & 1 \end{pmatrix} \begin{pmatrix} \cos \alpha & 0 & -\sin \alpha \\ 0 & 1 & 0 \\ \sin \alpha & 0 & \cos \alpha \end{pmatrix} \begin{pmatrix} x \\ y \\ z \end{pmatrix} \quad (2.202)$$

$$\begin{pmatrix} x'' \\ y'' \\ z'' \end{pmatrix} = \begin{pmatrix} \cos \beta \cos \alpha & \sin \beta & -\sin \alpha \cos \beta \\ -\sin \beta \cos \alpha & \cos \beta & \sin \alpha \sin \beta \\ \sin \alpha & 0 & \cos \alpha \end{pmatrix} \begin{pmatrix} x \\ y \\ z \end{pmatrix} \quad (2.203)$$

$$\begin{aligned} x'' &= \cos \beta \cos \alpha x + \sin \beta y - \sin \alpha \cos \beta z \\ y'' &= -\sin \beta \cos \alpha x + \cos \beta y + \sin \alpha \sin \beta z \\ z'' &= \sin \alpha x + \cos \alpha z \end{aligned} \quad (2.204)$$

In terms of polar coordinates, we have

$$\begin{aligned} x' &= r \cos \phi' \\ y' &= r \sin \phi' \\ z' &= 0 \end{aligned} \quad (2.205)$$

and the relation between the primed and unprimed axes may be written

$$\begin{pmatrix} x' \\ y' \\ z' \end{pmatrix} = \begin{pmatrix} \cos \alpha & 0 & -\sin \alpha \\ 0 & 1 & 0 \\ \sin \alpha & 0 & \cos \alpha \end{pmatrix} \begin{pmatrix} x \\ y \\ z \end{pmatrix} \quad (2.206)$$

or taking the inverse of Equation (2.206) yields

$$\begin{pmatrix} x \\ y \\ z \end{pmatrix} = \begin{pmatrix} \cos \alpha & 0 & \sin \alpha \\ 0 & 1 & 0 \\ -\sin \alpha & 0 & \cos \alpha \end{pmatrix} \begin{pmatrix} x' \\ y' \\ z' \end{pmatrix}$$

and

$$\left. \begin{aligned} x &= \cos \alpha x' + \sin \alpha z' \\ y &= y' \\ z &= -\sin \alpha x' + \cos \alpha z' \end{aligned} \right\} \quad (2.207)$$

Substitute Equations (2.205) into (2.207) to obtain

$$\left. \begin{aligned} x &= r \cos \alpha \cos \phi' \\ y &= r \sin \phi' \\ z &= -r \sin \alpha \cos \phi' \end{aligned} \right\} \quad (2.208)$$

The potential of the magnetic field, $V = +V_0 \cos \theta' / r^2 = +V_0 z / r^3$ or

$$V = +V_0 z (x^2 + y^2 + z^2)^{-3/2} \quad (2.209)$$

Since the magnetic field is very nearly a dipole field, it can be obtained by taking the negative gradient of the potential.

$$\begin{aligned} B_x &= -\frac{\partial V}{\partial x} = -V_0 z \left[-\frac{3}{2} (x^2 + y^2 + z^2)^{-5/2} \cdot 2x \right] = 3 \left(\frac{V_0}{r^5} \right) xz \\ B_y &= -\frac{\partial V}{\partial y} = -V_0 z \left[-\frac{3}{2} (x^2 + y^2 + z^2)^{-5/2} \cdot 2y \right] = 3 \left(\frac{V_0}{r^5} \right) yz \\ B_z &= -\frac{\partial V}{\partial z} = -V_0 \left\{ r^{-3} + z \left[-\frac{3}{2} (x^2 + y^2 + z^2)^{-5/2} \cdot 2z \right] \right\} = V_0 \left(\frac{1}{r^3} - \frac{3z^2}{r^5} \right) \\ &= -V_0 \left(\frac{x^2 + y^2 + z^2 - 3z^2}{r^5} \right) = \left(\frac{V_0}{r^5} \right) (2z^2 - x^2 - y^2) \end{aligned} \quad (2.210)$$

The average value of B^2 normal to the spin of the satellite is:

$$B_{\text{eff}}^2 = \frac{1}{2\pi} \int_0^{2\pi} d\phi' (\mathbf{S} \times \mathbf{B}) \cdot (\mathbf{S} \times \mathbf{B}) \quad (2.211)$$

where $\mathbf{S}$ is a unit vector in the direction of spin. Also since

$$(\mathbf{S} \times \mathbf{B}) \cdot (\mathbf{S} \times \mathbf{B}) = (\mathbf{S} \cdot \mathbf{B})(\mathbf{B} \cdot \mathbf{B}) - (\mathbf{S} \cdot \mathbf{B})(\mathbf{S} \cdot \mathbf{B}) \text{ and because } \mathbf{S} \cdot \mathbf{S} = 1; \mathbf{S} \cdot \mathbf{B} = B_y'', \text{ Equation (2.211) becomes:}$$

$$B_{\alpha\alpha}^2 = \frac{1}{2\pi} \int_0^{2\pi} d\phi' (\mathbf{B} \cdot \mathbf{B} - \mathbf{B}_y''^2) \quad (2.212)$$

$\mathbf{B} \cdot \mathbf{B} = B_x^2 + B_y^2 + B_z^2$ and from Equations (2.204) we have

$$B_y''^2 = (-\sin \beta \cos \alpha B_x + \cos \beta B_y + \sin \alpha \sin \beta B_z)^2$$

Evaluating the integrand of Equation (2.212) yields

$$\begin{aligned} \mathbf{B} \cdot \mathbf{B} - \mathbf{B}_y''^2 = & (1 - \sin^2 \beta \cos^2 \alpha) B_x^2 + (1 - \cos^2 \beta) B_y^2 + (1 - \sin^2 \alpha \sin^2 \beta) B_z^2 \\ & + 2 \sin \beta \cos \beta \cos \alpha B_x B_y + 2 \sin \alpha \cos \alpha \sin^2 \beta B_x B_z - 2 \sin \alpha \sin \beta \cos \beta B_y B_z \end{aligned} \quad (2.213)$$

Substituting Equations (2.208) into (2.210) yields

$$\left. \begin{aligned} B_x &= \left(\frac{V_0}{r^5} \right) 3xz = \left(\frac{V_0}{r^5} \right) (-3r^2 \sin \alpha \cos \alpha \cos^2 \phi') = -3 \left(\frac{V_0}{r^3} \right) \sin \alpha \cos \alpha \cos^2 \phi' \\ B_y &= \left(\frac{V_0}{r^5} \right) 3yz = \left(\frac{V_0}{r^5} \right) (-3r^2 \sin \alpha \sin \phi' \cos \phi') = -3 \left(\frac{V_0}{r^3} \right) \sin \alpha \sin \phi' \cos \phi' \\ B_z &= \left(\frac{V_0}{r^5} \right) (2z^2 - x^2 - y^2) = \left(\frac{V_0}{r^5} \right) (2r^2 \sin^2 \alpha \cos^2 \phi' - r^2 \cos^2 \alpha \cos^2 \phi' - r^2 \sin^2 \phi') \\ &= \left(\frac{V_0}{r^3} \right) (3 \sin^2 \alpha \cos^2 \phi' - 1) \end{aligned} \right\} \quad (2.214)$$

Substitute Equations (2.214) into (2.213) to obtain the result

$$\begin{aligned} \mathbf{B} \cdot \mathbf{B} - \mathbf{B}_y''^2 = & \left[\left(\frac{V_0}{r^3} \right)^2 1 + 3 \sin^2 \alpha \cos^2 \phi' - 9 \sin^2 \alpha \sin^2 \phi' \cos^2 \phi' + \sin^2 \beta \sin^2 \alpha (6 \cos^2 \phi' \right. \\ & \left. - 1 + 9 \sin^2 \phi' \cos^2 \phi' - 9 \cos^4 \phi') + 6 \sin \beta \cos \beta \sin^2 \alpha (3 \sin \phi' \cos^3 \phi' \sin \phi' \cos \phi') \right] \end{aligned} \quad (2.215)$$

Equation (2.215) is now used to evaluate Equation (2.212).

$$\begin{aligned} B_{\alpha\alpha}^2 &= \frac{1}{2\pi} \left(\frac{V_0}{r^3} \right)^2 \left\{ \phi + \left[3 \sin^2 \alpha \left(\frac{\phi}{2} + \frac{\sin^2 \phi}{4} \right) \right] - \left[9 \sin^2 \alpha \left(\frac{\phi}{8} - \frac{\sin^4 \phi}{32} \right) \right] \right\} \\ &+ \sin^2 \beta \sin^2 \alpha \left(3\phi + \frac{3 \sin^2 \phi}{2} - \phi + \frac{9\phi}{8} - \frac{9 \sin^4 \phi}{32} - \frac{27\phi}{8} - \frac{9 \sin^2 \phi}{4} - \frac{9 \sin^4 \phi}{32} \right. \\ &\quad \left. + 6 \sin \beta \cos \beta \sin^2 \alpha \right) \left(-\frac{3 \cos^4 \phi}{4} - \frac{\sin^2 \phi}{2} \right) \Big|_0^{2\pi} \\ &= \frac{1}{2\pi} \left(\frac{V_0}{r^3} \right)^2 \left[\phi + \frac{3}{8} \phi \sin^2 \alpha - \left(\frac{1}{4} \phi \sin^2 \beta \sin^2 \alpha \right) \right] \Big|_0^{2\pi} \end{aligned}$$

or

$$\begin{aligned} B_{\alpha\alpha}^2 &= \left(\frac{V_0}{r^3} \right)^2 \left[1 + \frac{3}{8} \sin^2 \left(1 - \frac{2}{3} \sin^2 \alpha \right) \left(1 - \frac{2}{3} \sin^2 \beta \right) \right] \\ &= B_{1g}^2(\alpha, \beta) \end{aligned} \quad (2.216)$$

where $B_1 = V_0/R^3$ is the value of the geomagnetic field at the magnetic equator a distance R from the origin.

B_1^2 can be written:

$$B_1^2 = (V_0/R_e^3)(R_e/R)^6 \\ = B_0^2(1+x)^6$$

or

$$B_1^2 = B_0^2/h(x) \quad (2.217)$$

where

$R_e = 40,000/2\pi km$ is the radius of the earth,
 $A = R - R_e$ is the height of the orbit above sea level,

$x = A/R_e$, and B_0 , the magnitude of the geomagnetic field at sea level at the magnetic equator, is 3.1×10^{-5} webers/m².

To determine the time constant for the decay rate of spin, we make the following calculations:

$$p = \pi a^2 \sigma d l B_1^2$$

$$a = 0.559 \text{ m}/\sigma = 3.8 \times 10^7 \text{ mho/m}; d = 2.38 \\ \times 10^{-3} \text{ m}; l = 0.254 \text{ m}$$

$$B_1^2 = B_0^2 g(\alpha, \beta)/h(x)$$

$$B_0 = 3.1 \times 10^{-5} \text{ webers/m}^2$$

For average values used in Rosenstock§ of $\sin^2 \alpha = 0.35$; $\sin^2 \beta = 0.5$

$$g(\alpha, \beta) = 1 + \frac{3}{8} (0.35) \left(1 - \frac{2}{3} \cdot \frac{1}{2}\right) \\ = 1 + (0.375)(0.667) = 1.0875$$

The average height of orbit = 300 nautical miles = 555.5 km; therefore, we have

$$x = (5\pi \times 10^{-5})(555.5) = 0.08726$$

$$h(x) = (1.08726)^6 = 1.65195$$

$$B_1^2 = \frac{1.0875}{1.65195} (3.1 \times 10^{-5})^2 = 0.65831 (9.61 \times 10^{-10}) \\ = 6.326 \times 10^{-10}$$

$$p = \pi (0.559)^3 (0.254) (3.8 \times 10^7) (2.28 \times 10^{-3}) \\ (6.326 \times 10^{-10}) = 7.640 \times 10^{-6} \text{ kgm}^2/\text{sec}$$

$$I_0 = 20.73 \text{ slug ft}^2 = 667.506 \text{ lb} \cdot \text{ft}^2 = 62.01 \text{ lb m}^2 \\ = 28.13 \text{ kgm}^2$$

and

$$\tau = \frac{I_0}{p} = 3.68 \times 10^6 \text{ sec} \approx 43 \text{ days}$$

Eddy Currents in the Rib Plates

To determine the effect on the spin rate of the torque exerted on the flat plates that divide the rotating body into nine separate compartments, the derivation given in Winch^{||} for the average power loss due to eddy currents is used to obtain the expression

$$p = \frac{l^2 h^2 b}{64 \rho} B_m^2 \omega^2 \quad (2.218)$$

and total torque is given by the relation

$$T = -p/\omega = -\frac{l^2 h^2 b}{64 \rho} B_m^2 = -k\omega \quad (2.219)$$

where

$$k = \frac{l^2 h^2 b}{64} \beta_m^2$$

For (9) 16-inch panels, we calculate the values

$$l = 16 \text{ in.} = 0.4064 \text{ m}; l^2 = 0.165 \text{ m}^2$$

$$h = 10 \text{ in.} = 0.254 \text{ m}; h^2 = 0.0645 \text{ m}^2$$

$$b = 0.060 \text{ in.} = 1.524 \times 10^{-3} \text{ m}$$

$$\rho = 2.63 \times 10^{-8} \text{ ohm/m}$$

$$B_m^2 = 6.326 \times 10^{-10} \text{ webers}^2/\text{m}^4$$

$$K = \frac{9(0.165)(0.0645)(1.524 \times 10^{-3})(6.326 \times 10^{-10})}{64(2.63 \times 10^{-8})} \\ = 5.486 \times 10^{-8}$$

and the time constant for spin rate decay due to this factor is

$$\tau = \frac{28.13}{5.49} \times 10^8 = 5.12 \times 10^8 \text{ sec} = 5,926 \text{ days}$$

Eddy Currents in Inner Cylinder

The time constant due to torque on the inner cylindrical shell of the spacecraft is evaluated as follows:

§H. B. Rosenstock, Memorandum on "The Effects of the Earth's Magnetic Field on the Spin of a Satellite," Optics Div., Naval Research Laboratory, Washington 25, D.C.

||R. P. Winch, *Electricity and Magnetism*, New York, Prentice Hall, 1955, pp. 550-52.

$$a=6 \text{ in.}=0.1524 \text{ m}; a^3=3.54 \times 10^{-3} \text{ m}^3; b=10 \text{ in.}=0.254 \text{ m}; d=0.1 \text{ in.}=2.54 \times 10^{-3} \text{ m}$$

$$p=\pi a^3 \sigma d l B_1^2 = \pi (3.54 \times 10^{-3}) (3.8 \times 10^7) (2.54 \times 10^{-3}) (0.254) (7.252 \times 10^{-10})$$

$$= (1.97725 \times 10^{-7}) \text{ kg m}^2/\text{sec}$$

and

$$\tau = \frac{2.813}{1.977} \times 10^8 = 1.423 \times 10^8 \text{ sec} = 1647 \text{ days}$$

Summary of Eddy Current Damping Torques

Adding the torques acting on nine flat panels and the inner cylinder, we have

$$p = (1.97725 + 0.5486) \times 10^{-7} = 2.5259 \times 10^{-7}$$

$$\tau = \frac{2.813}{2.526} \times 10^8 = 1.114 \times 10^8 \text{ sec} = 1,289 \text{ days}$$

Adding the effect of both cylindrical shells and the flat panels yields

$$p = (7.640 + 0.1977 + 0.0548) \times 10^{-6} = 7.893 \times 10^{-6}$$

$$\tau = \frac{28.13}{7.89} \times 10^6 = 3.57 \times 10^6 \text{ sec} = 41 \text{ days}$$

The cylinder of the wheel was the only source of damping torque not of negligible magnitude when considering a six-month orbital life. The only effective way to reduce this damping torque was to insulate the compartment cover plates and the rim panels to break up eddy current paths. It was not practical to insulate the rim panels since they were structural members. The cover plates were not difficult to insulate from the rest of the structure, so this was done.

A slow-down time constant of between 41 and 187 days was anticipated considering all torque effects present. This required a spin-up gas expenditure about every four days in the extreme case. (See the discussion of Chapter 4 on the gas system.) Spin-up cycling at four-day intervals would have used up the spin gas in a little over a year; well in excess of the six-month life supply requirement.

Magnetic Damping Due to Hysteresis Losses

Ferromagnetic materials were used in the OSO I spacecraft. Hysteresis losses will occur

whenever a ferromagnetic material rotates in the earth's magnetic field. The spacecraft wheel was continuously rotating and, therefore, the hysteresis losses were constantly present. The ferromagnetic materials in the upper section contributed their losses only during the dark portion of the orbit when the upper structure rotated with the wheel.

From Figure 11-26, page 509 of Bozorth,** the hysteresis loss per cycle is given for various ferromagnetic materials. Taking the value for 45 Permalloy and for a maximum induction of $B=0.23$ gauss, the power loss per cycle due to hysteresis is approximately 1 erg/cm³.

The average power loss in the wheel, p_w , is

$$p_w = (1 \text{ erg/cm}^3/\text{cycle})(V_w \text{ cm}^3)(0.5 \text{ cycle/sec})$$

$$= V_w/2 \text{ ergs/sec}$$

$$= \frac{V_w}{2} \times 10^{-7} \text{ watts}$$

where

V_w = volume of ferromagnetic materials in the wheel, in cm³

Since the spacecraft was in the dark portion of the orbit a maximum of 35 percent of the orbital time, the maximum effective value for the average power loss in the upper structure, p_u , due to hysteresis was

$$p_u = (0.35) \frac{V_u}{2} \times 10^{-7} \text{ watts}$$

where

V_u = volume of ferromagnetic materials in the upper structure, in cm³

This average power loss value is now compared with the power losses due to eddy cur-

**R. M. Bozorth, *Ferromagnetism*, Princeton, N.J.; D. Van Nostrand Co., Inc., 1951.

rents in the rotating wheel of the spacecraft. In Equation (2.200), the torque exerted on the spacecraft due to eddy currents in the cylindrical shell is given as

$$T = -k\omega$$

where

$$k = \pi a^3 l \sigma dB_{\perp}^2$$

The instantaneous power loss is related to the torque as follows:

$$p = -\frac{dE}{dt} = -\frac{d}{dt} \left(\frac{1}{2} I \omega^2 \right) = -(I\dot{\omega})\omega = -T\omega$$

or

$$T = -\frac{p}{\omega}$$

Therefore

$$k\omega = \frac{p}{\omega}$$

or

$$p = k\omega^2$$

For the cylindrical shell, $k = 8.77 \times 10^{-6}$ kg m²/sec and

$$p = \frac{1}{2} (\pi)^2 (8.77 \times 10^{-6}) = 8.66 \times 10^{-5} \text{ watt}$$

This is the average power loss because in computing k , an average value of B^2 was used.

Thus, if the time constant for the decay of spin rate due to eddy currents in the cylindrical shell is

$$\tau_E = 3.68 \times 10^6 \text{ sec} = 43 \text{ days}$$

the time constant due to hysteresis losses is

$$\tau_H = \frac{866}{\frac{V_w}{2} + 0.35 \frac{V_u}{2}} \tau_E = \frac{1732 \times 43}{V_w + 0.35 V_u} \text{ days}$$

For the OSO I, V_w and V_u are estimated to be 100 cm³ and 25 cm³, respectively, and therefore we determine that

$$\tau_H = \frac{1732}{109} (43) = 688 \text{ days}$$

Therefore, the effect of hysteresis losses in ferromagnetic materials would not be significant when compared to the eddy current losses were the eddy current paths not broken up. The

688-day time constant is, however, significant compared to the actual eddy current losses remaining due to the rib plates and inner cylinder conducting paths.

Magnetic Coupling Torques

Torques Due to Permanent Magnets in Pointed Instruments.

The open-end photomultiplier in the Goddard pointed experiment used 8 Alnico permanent magnets for magnetic focusing. These magnets were arranged parallel to each other and were mounted with their poles aligned with the pitch axis of the spacecraft. The torques produced by the interaction of their magnetic moment with the earth's field had either spin or roll components. The combined pole strength of these 8 magnets was not known, but they were obviously quite strong. It was doubtful that the spin rate increase or the extra gas requirement for stabilization in pitch that would result from their effect could be tolerated.

Fortunately, a solution to the problem was to mount another photomultiplier (or the part housing the magnets) on the upper structure with the pole directions of the magnets reversed. This canceled the effects of the first unit both in pitch and spin at all times. There were no location restrictions on the second unit since the moment arm of the couple was the length of the magnets and was independent of location on the upper structure. Thus, as shown in Figure 2-19, we have that

$$T_z = -T'_z$$

$$T_x = -T'_x$$

$$T_y = T'_y = 0$$

regardless of the direction of H.

Spin decay due to hysteresis and eddy current damping in these magnets at night when the instrument is spinning should be negligible.

Since a second unit of magnets probably would not exactly match the magnetic moment of those in the photomultiplier, calculations were made to determine how much residual torque could be tolerated.

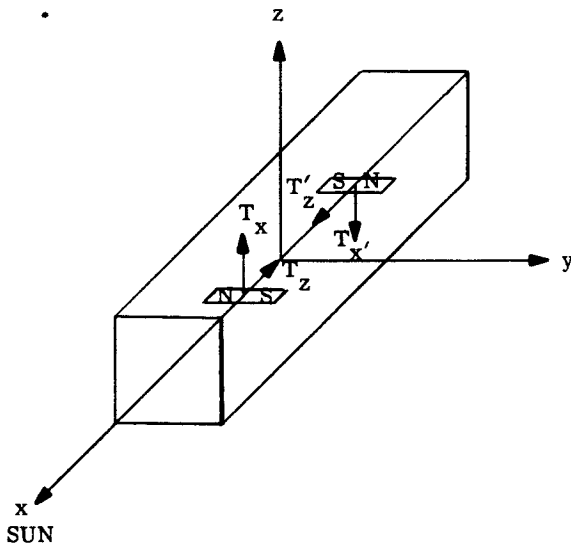


FIGURE 2-19.—Torques resulting from interaction between bar magnets and the axial components of the earth's magnetic field.

First, however, it is worth while to point out that the spin component of this residual torque will average out to near zero around one spacecraft orbit. For an orbit inclined 33° to the earth's equator, H will oscillate 33° back and forth through the y - z plane (Figure 2-19) regardless of the direction of the component of H lying in that plane. If the geographical and geomagnetic poles were coincident this oscillation would be exactly $\pm 33^\circ$, but since they vary by several degrees some accumulative torque is realized. The net accumulation is small, however, in comparison to that in roll, and from the restriction below, can be neglected.

In determining the maximum allowable residual torques of the magnets in roll, an arbitrary statement that no more than 20 percent of the total gas available for torquing in roll can be used to buck out this torque.

The following data is required:

Specific impulse of N_2 , $I_{sp} = 65 \frac{\text{lb}_f \text{ sec}}{\text{lb}_w}$

Total weight of available N_2 , $W = 4.4 \text{ lb}$

Thrust of each jet, $F_j = 0.05 \text{ lb}_f$

Torque of each jet, $T_j = 0.05 \text{ lb}_f \cdot 2 \text{ ft} = 0.10 \text{ lb-ft}$

Total impulse available from N_2 gas,

$$I_T = I_{sp} \times W = 286 \text{ lb}_f \text{ sec}$$

Total thrust time available,

$$t_j = \frac{I_T}{F} = 5,720 \text{ sec}$$

Total torque impulse available,

$$(T, t_j)_T = 572 \text{ lb-ft-sec}$$

If 20 percent of this amount is available to counter magnetic torques, then

$$(T, t_j)_{\text{avail}} = 114.4 \text{ lb-ft-sec}$$

Assume that the roll rate is uniform and slow compared to the orbital rate, yet is fast enough for the spacecraft to make a complete rotation in a few days. Then, since torques of either sense consume gas, the average magnetic torque is the sinusoidal average of $\frac{1}{2}$ cycle, or 63.7 percent of maximum magnetic torque. Equal magnetic torques and gas jet torques produce the same precession rate in pitch, so

$$0.637 T_m t - (T, t_j)_{\text{available}} = 114.4 \text{ lb-ft-sec}$$

or

$$t = \frac{2.433 \times 10^9}{T_m} \text{ sec}$$

or

$$t = \frac{2.82 \times 10^4}{T_m} \text{ days}$$

where T_m is the maximum magnetic torque in dyne-cm.

Thus, if $t = 6 \text{ months} = 180 \text{ days}$

$$T_m = \frac{2.82 \times 10^4}{180} \cong 160 \text{ dyne-cm.}$$

The upper structure was magnetically balanced to less than 100 dyne-cm unbalance torque at the surface of the earth.

Torques Due to Permanent Magnets in Rotating Section

Some of the wheel experiments contained small permanent magnets. The effects of these rotating magnets on spacecraft dynamics were investigated. Two different magnet orientations needed to be considered:

1. magnetic axis parallel to spin axis
2. magnetic axis normal to spin axis

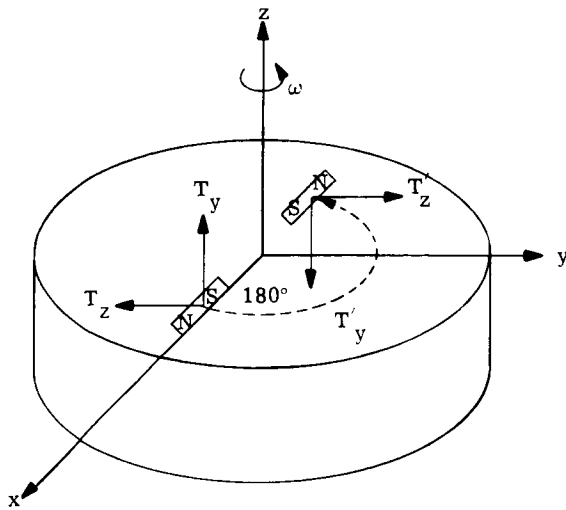


FIGURE 2-20.—Torques produced by a magnet mounted normal to the spin axis.

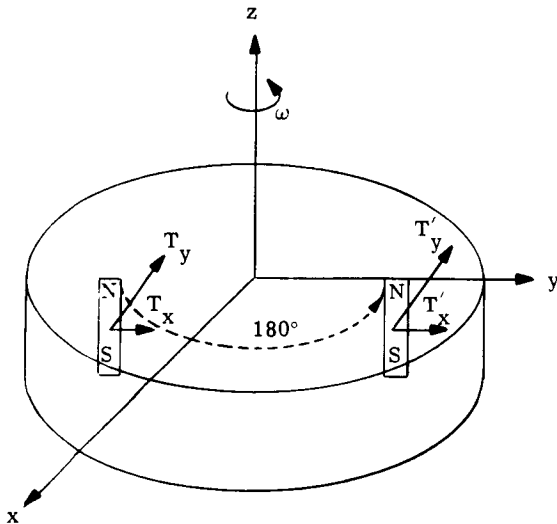


FIGURE 2-21.—Torques produced by a magnet mounted parallel to the spin axis.

Since the wheel was constantly rotating, constant alignment of a magnet with the pitch or roll axis was not possible. For a magnet mounted with the magnetic axis not aligned with the spin axis nor exactly normal to it, the torque produced would have had both components.

When the magnetic axis of the magnet was normal to the spin axis, the net torque exerted by the magnet on the spacecraft was zero.

This is shown in Figure 2-20. After 180° of rotation the torque due to the earth's field acting either along or normal to the spin axis is reversed in direction. In the figure, T_y is the reaction of the magnet to the earth's field when the magnetic north pole is in the $+y$ direction, etc.

From Figure 2-21 note that the torque produced by a magnet mounted parallel to the spin axis does not change direction due to spin and is therefore cumulative. If in Figure 2-21 the sun is always in the $+y$ direction, the earth's field vector oscillates back and forth through the y - z plane in the manner previously described, and the major cumulative component of this torque is a roll torque producing precession in pitch. As the spacecraft rolls, this torque is 90° out of phase with that produced by the magnets mounted on the pointed instrument, but it still requires pitch jet gas to counter its effect. Therefore, this torque must be added to the residual torque of the other magnets in satisfying the value for T_m given above.

Solar Array Current Loops.

Any current loops in the spacecraft solar array would have produced a coupling torque tending to precess the spacecraft about the roll axis. To compensate for this effect, the solar array was wired as follows. The two roughly quarter-circular sections of the array were mounted symmetrically with respect to the spin axis and were wired as mirror images of each other such that the net coupling torque is zero. The rectangular center panel lead wires were looped to form compensating loops for the panel cell wiring to also produce a net zero torque from the center panel.

Orbital Motion of Earth

Although the motion of the earth about the sun did not introduce any external torques on the satellite, it did introduce a need for pitch axis correction. If the spacecraft were oriented with the spin axis located in the ecliptic plane, this would have required a pitch correction of approximately 1 degree per day. For the spin axis perpendicular to the ecliptic plane, no pitch correction was required. To obtain a

reasonable number, consider the spin axis to be oriented at an angle of 45° with respect to the ecliptic plane. In this case, the necessary pitch correction would be 0.707 degrees per day.

Vehicle Separation Effects

Separation of the spacecraft from the third-stage engine was accomplished by means of a separation spring located in a cavity in the interface plane. The spring force did not act purely along the spacecraft spin axis but imparted a force component to the spacecraft in a plane perpendicular to the spin axis.

To investigate the effect of the side force, Equations (2.86) are used with the torques set equal to zero. These equations become

$$-I_1\ddot{\alpha} + I\omega\dot{\theta} = 0 \quad I_1\ddot{\theta} + I\omega\dot{\alpha} = 0 \quad (2.220)$$

where ω is constant. Dividing Equations (2.220) by I_1 we have

$$\ddot{\alpha} - p\dot{\theta} = 0 \quad \ddot{\theta} + p\dot{\alpha} = 0 \quad (2.221)$$

where

$$p = \frac{I\omega}{I_1}$$

The Laplace transforms of these equations are

$$s^2\alpha(s) - p s\theta(s) = s\alpha(0) + \dot{\alpha}(0) - p\theta(0) \\ p s\alpha(s) + s^2\theta(s) = p\alpha(0) + s\theta(0) + \dot{\theta}(0) \quad (2.222)$$

The solution of Equations (2.222) for $\alpha(s)$ is

$$\alpha(s) = \frac{\alpha(0)}{s} + \frac{1}{s^2 + p^2} \dot{\alpha}(0) + \frac{p}{s(s^2 + p^2)} \dot{\theta}(0) \quad (2.223)$$

and for $\theta(s)$ is

$$\theta(s) = \frac{\theta(0)}{s} + \frac{1}{s^2 + p^2} \dot{\theta}(0) - \frac{p}{s(s^2 + p^2)} \dot{\alpha}(0) \quad (2.224)$$

and, the time functions are

$$\alpha(t) = \alpha(0) + \dot{\alpha}(0) \frac{\sin pt}{p} + \frac{\dot{\theta}(0)}{p} (1 - \cos pt) \quad (2.225)$$

and

$$\theta(t) = \theta(0) + \dot{\theta}(0) \frac{\sin pt}{p} - \frac{\dot{\alpha}(0)}{p} (1 - \cos pt) \quad (2.226)$$

Equations (2.225) and (2.226) show that the same axis initial position error is unchanged; the same axis initial rate error has an average value of zero. The cross axis initial rate error also appears and has an average value equal to the initial rates divided by p .

Summary of the Effects of External Torques

Many of the interactions produced torques that were subtractive. Therefore, the totals listed in Table 2-1 are probably extreme, and the average anticipated values in most cases were considerably less.

TABLE 2-1.—Summary of External Torque Effects

Torque	Pitch error rate (deg/day)	Roll error rate (deg/day)	Spin decay time constant (days)
Gravity gradient.....	0.85	1.1	312
Aerodynamic.....	.05	-----	-----
Magnetic.....	-----	-----	300
Ferromagnetic.....	-----	-----	688
Eddy current.....	-----	-----	41
Radiation pressure.....	-----	.005	-----
Earth orbital motion.....	71 ave.	-----	-----
Total if all additive...	1.6	1.1	30

CONTROL OF TORQUE EFFECTS

Spacecraft Pitch Control

To control the pitch angle, torques were applied by means of gas jets about the N -axis (see Figure 2-2) to cause precessional motion. The spacecraft pitch control circuit and gas system, described in detail in Chapter 4, provided this control to keep the pitch axis aligned to within 3° of the solar direction. Spacecraft dynamics associated with this system are discussed below.

In Equations (2.85) θ is set equal to $\pi/2$ and M_m and M_p are set equal to zero to obtain the equations

$$\left. \begin{aligned} I_1\ddot{\alpha} - I\omega\dot{\theta} &= 0 \\ I_1\ddot{\theta} + I\omega\dot{\alpha} &= M_n \\ I\dot{\omega} &= 0 \end{aligned} \right\} \quad (2.227)$$

where ω has been substituted for $\dot{\phi}$ and from the last equation we note that ω is a constant. Differentiation of the first of Equations (2.227) results in

$$\ddot{\alpha} - p\ddot{\theta} = 0 \quad (2.228)$$

where

$$p = \frac{I\omega}{I_1}$$

Substituting the value for $\ddot{\theta}$ from the second of Equations (2.227) into Equation (2.228) gives

$$\ddot{\alpha} + p^2\dot{\alpha} = \frac{p^2}{I\omega} M_n$$

Taking the Laplace transform, with initial values zero, gives

$$\dot{\alpha}(s) = \frac{p^2}{I\omega} M_n \frac{1}{s(s^2 + p^2)} \quad (2.229)$$

or

$$\dot{\alpha}(t) = \frac{M_n}{I\omega} (1 - \cos pt) \quad (2.230)$$

and

$$\alpha(t) = \frac{M_n}{I\omega} \left(t - \frac{\sin pt}{p} \right) \quad (2.231)$$

Substituting Equation (2.230) into the second of Equations (2.227) yields

$$\begin{aligned} \ddot{\theta}(t) &= \frac{M_n}{I_1} - p \frac{M_n}{I\omega} (1 - \cos pt) \\ &= \frac{M_n}{I_1} \cos pt \end{aligned} \quad (2.232)$$

Integration of Equation (2.232) yields

$$\dot{\theta}(t) = \frac{M_n}{I_1} \cdot \frac{\sin pt}{p} = \frac{M_n}{I\omega} \sin pt \quad (2.233)$$

and

$$\theta(t) = \frac{n}{2} + \frac{M_n}{I\omega p} (1 - \cos pt) \quad (2.234)$$

Therefore, the application of a torque about the N -axis causes an average pitch velocity of $\dot{\alpha} = M_n/I\omega$, while the average roll velocity ($\dot{\theta}$) is zero.

For a torque about the M -axis (pitch axis) Equations (2.227) become

$$\ddot{\alpha} - p\ddot{\theta} = -M_m/I_r$$

$$\ddot{\theta} + p\dot{\alpha} = 0$$

$$p = \text{constant} \quad (2.235)$$

These equations may be combined to form the expression

$$\ddot{\theta} + p^2\dot{\theta} = \frac{p^2}{I\omega} M_m \quad (2.236)$$

By comparing Equation (2.236) with Equation (2.229) it is seen that the solution for θ is given by Equation (2.231) with α replaced by θ , and the solution for α is given by Equation (2.234) with θ replaced by α .

Therefore, the application of a torque about the M -axis (pitch axis) caused an average roll velocity of $\dot{\theta} = M_m/I\omega$, while the average pitch velocity ($\dot{\alpha}$) was zero. The roll position of the spacecraft was not controlled.

Chapter 3

STRUCTURAL DESIGN AND FABRICATION

DESIGN CRITERIA

The configuration of the Orbiting Solar Observatory as presented in the initial proposal is shown in Figure 3-1. At that time it contained all of the basic elements of the final design except the external arms which carried the spin gas supply bottles.

As the various design criteria, i.e., power requirements, functions of the spacecraft, size and shape of the payload envelope, and load inputs from the launch vehicle became known, the present OSO I configuration gradually evolved.

Function

The function of the spacecraft structure was twofold: to carry all experiments, controls, and components safely through the launch phase, and to serve as a part of the control system when placed in orbit.

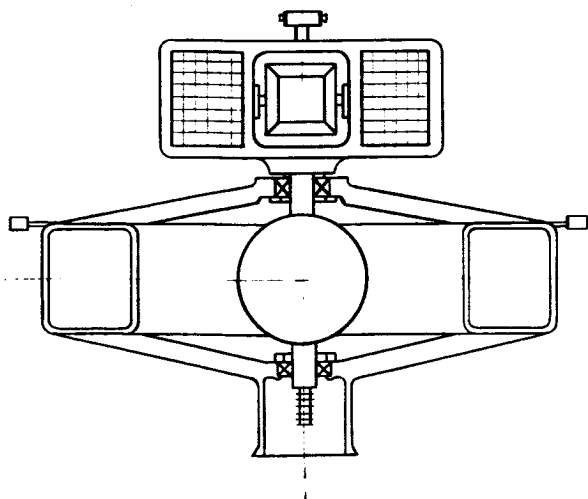


FIGURE 3-1.—Proposal stage configuration.

Dimensions and Geometry

Figure 3-2 illustrates the second stage of evolution. This layout was completed in early 1960, prior to the first experimenters' meeting. At the time, dimensional layouts of the Delta shroud had been received as had the preliminary vibration input specifications and the steady-state acceleration specification. At this point, the over-all weight was to be 350 pounds and the pointed experiments were to be not more than 80 pounds.

Figure 3-2 shows all of the basic elements contained in Figure 3-1 with some change. The maximum diameter of the wheel was known to be 44 inches and the pointed experiments were to be 4 inches wide by 8 inches high and no longer than 39 inches. The 4-by-8 inch dimensions were compatible with the Aerobee pointing control experiment space in the anticipation that OSO I pointed experiments might

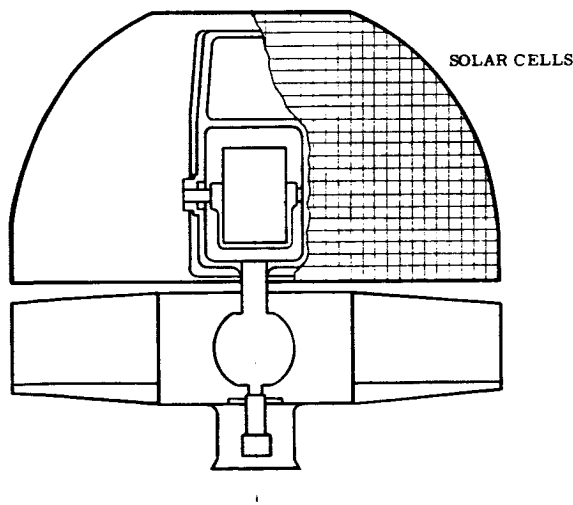


FIGURE 3-2.—Second configuration.

be tested in Aerobee shots. The length was determined by laying out the necessary components. Starting at the third stage mounting interface the components were stacked until the position of the pointed experiments was determined in relation to the shroud curvature.

At that time it was known that the power requirements were much greater than originally thought, and the solar cell array was shown to require almost all available space within the Delta shroud.

The major changes were the division of the wheel into compartments, and the use of rib and deck structure in the design of the wheel. Spin gas requirements were determined and the storage bottles selected. Three containers were required to store the gas, so compartmentation was made a multiple of three. Nine compartments were chosen because of the deck span of each compartment at the outer edge. This configuration provided ample compartment space and good structural rigidity.

At that time very little was known about the weights and centers of gravity of the various experiments and components to be mounted in the wheel. The Delta was more adaptable to a spherical payload than a wheel configuration. It was found that the principal moments of inertia of the configuration shown in Figure 3-2 were very nearly those of a sphere, but it was determined from dynamics considerations that the ratio of spin axis moment of inertia to any transverse axis moment of inertia had to be at least 1:2. It was decided to add extension arms to obtain this ratio. The gas storage bottles were the logical mass to put on the arms because of their size and weight.

Figure 3-3 shows the final OSO I structural configuration. The basic elements of the original concept were all present; the wheel was roughly a torus around the azimuth shaft and precession gas bottle, the pointed experiments were in a cast yoke, and the solar cell array panel was attached to the yoke. The main differences were the addition of the extension arms and the nine-sided wheel.

The changes in shape from the design shown in Figure 3-2 to that in Figure 3-3 were the result of further knowledge of the components which were fitted into the wheel and onto the

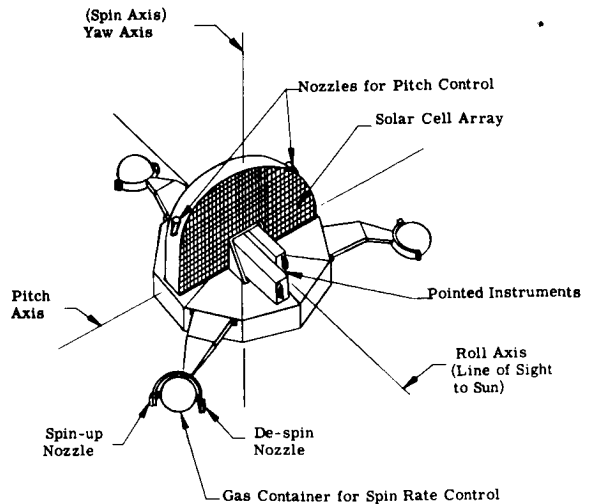


FIGURE 3-3.—Final configuration.

elevation casting, and of further analysis of loads and stresses to be sustained during launch. The flat wheel-perimeter segments resulted from the addition of the extension arms and further investigation of fabrication problems. The wheel deck structure was of stamped, convoluted sheet metal for weight reduction. Castings mounting the azimuth shaft bearings were refined for load-carrying ability in view of the increasing weight requirement and higher vibration input specifications.

Load Inputs

The thrust and vibration loads were specified by Douglas Aircraft Company in their Payload Restraints Document, and later modified by NASA. The first stage had a peak steady-state thrust acceleration of 12.1 g, the second stage 2.8 g, and the third stage 6 g.

WHEEL STRUCTURE

General

The wheel was a torus centered about the azimuth shaft and precession gas bottle, including the base casting and the upper bearing casting, but not the bearings or arms.

The wheel was composed of a center casting, upper and lower center castings, nine decks, nine ribs, nine rim panels, and the upper and lower cover panels as shown in Figure 3-4. The base was specified by the Douglas Payload Restraints Document to provide compatibility

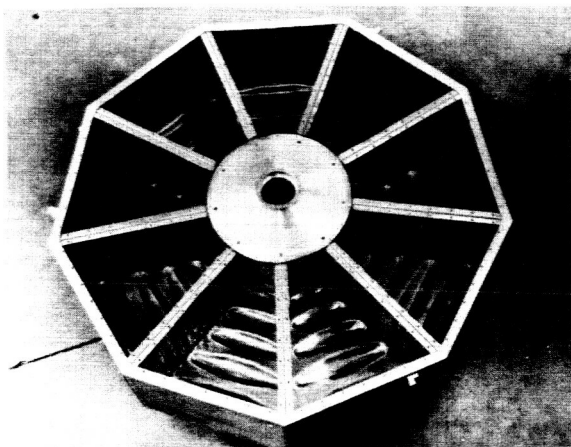


FIGURE 3-4.—Basic wheel assembly.

with the third-stage interface mounting surface. The third-stage rocket and the spacecraft were clamped with a vee-band. The wheel height was established by the longitudinal dimension required of slip rings, bearings, and precession gas bottle. The cover panels were used primarily for thermal shielding. It was necessary that each cover be electrically insulated to reduce the eddy current losses.

Loads and Stresses

The critical flight conditions for structural design occurred at the time dynamic pressure multiplied by booster angle of attack was a maximum, and at the time of Thor main engine burnout. These conditions are referred to as maximum $q\alpha$, and burnout, respectively. Maximum transverse loadings occurred at the maximum $q\alpha$ condition and maximum axial loads at the burnout condition. Limit loading inputs at the Delta vehicle-spacecraft interface for these conditions are as follows:

Maximum $q\alpha$ Condition:

<i>Axial</i>	
Steady-state	= +0
	-2.8 g
<i>Vibration</i>	
Sinusoidal (50-500 cps) *	= ± 7.1 g
Random	= 0.047 g ² /cps
<i>Transverse</i>	
<i>Vibration</i>	
Sinusoidal (50-500 cps) *	= ± 1.4 g
Random	= 0.047 g ² /cps

*It is assumed that the spacecraft is vibration critical in the 50-500 cps range.

Burnout Condition:

<i>Axial</i>	
Steady-state	= +0
	-12.1 g
<i>Vibration</i>	
Sinusoidal (50-500 cps)	= ± 7.1 g
Random	= 0
<i>Transverse</i>	
<i>Vibration</i>	
Sinusoidal (50-500 cps)	= ± 1.4 g
Random	= 0

It was necessary to estimate magnification and attenuation factors to arrive at final load factors on the spacecraft components. By using the conservative assumption that the lowest primary structural resonant frequency is at 100 cps and that the magnification factor is 2.5, final loads on the spacecraft components and structure became:

Maximum $q\alpha$ Condition:

<i>Axial</i>	
Steady-state	= +0
	-2.8 g
<i>Vibration</i>	
Sinusoidal	= ± 17.8 g
Random	= ± 5.1 g
<i>Total</i>	
	= +22.9
	-25.7 g
<i>Transverse</i>	
<i>Vibration</i>	
Sinusoidal	= ± 3.5 g
Random	= ± 5.1 g
<i>Total</i>	
	= ± 8.6 g

Burnout Condition:

<i>Axial</i>	
Steady-state	= +0
	-12.1 g
<i>Vibration</i>	
Sinusoidal	= ± 17.8 g
Random	= 0
<i>Total</i>	
	= +17.8 g
	-29.9 g
<i>Transverse</i>	
<i>Vibration</i>	
Sinusoidal	= ± 3.5 g
Random	= 0
<i>Total</i>	
	= ± 3.5 g

For design purposes, ultimate loadings were assumed to be 1.5 times the above values.

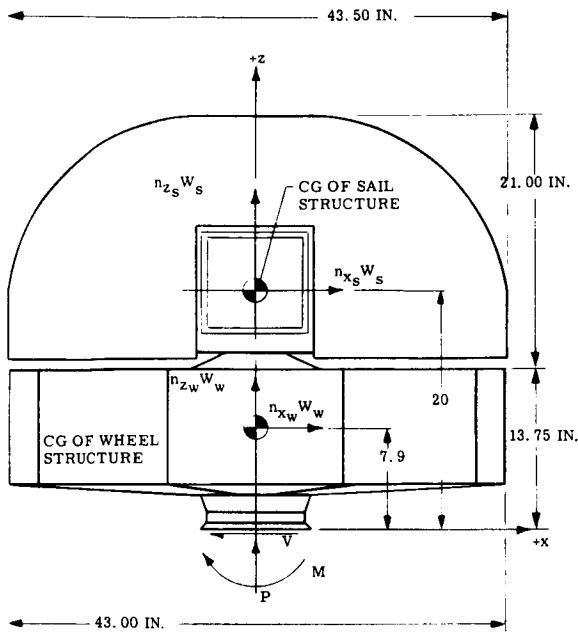


FIGURE 3-5.—Geometry and loading on OSO I.

Spacecraft Geometry

Figure 3-5 is a sketch of the spacecraft showing the general configuration and loading while in the boost phase. The symbols used are defined as follows:

W_s = Weight of oriented portion of spacecraft

W_w = Weight of wheel portion of spacecraft

n_x = Lateral load factor

n_z = Vertical load factor

V = Shear at spacecraft-launch vehicle interface

P = Axial load at spacecraft-launch vehicle interface

M = Bending moment at spacecraft-launch vehicle interface

Loads on Spacecraft

By utilizing geometry data from Figure 3-5 and the load factor data from the preceding

pages, the following equations were derived for the spacecraft-launch vehicle interface loads. Results of the equations appear in the table following.

$$\sum F_z = 0$$

$$M_{z_s} W_s + n_{z_w} W_w + P = 0 \quad (3.1)$$

$$P = -n_{z_s} W_s - n_{z_w} W_w \quad (3.2)$$

$$\sum F_x = 0$$

$$n_{x_s} W_s + n_{x_w} W_w - V = 0 \quad (3.3)$$

$$V = n_{x_s} W_s + n_{x_w} W_w \quad (3.4)$$

$$\sum M = 0$$

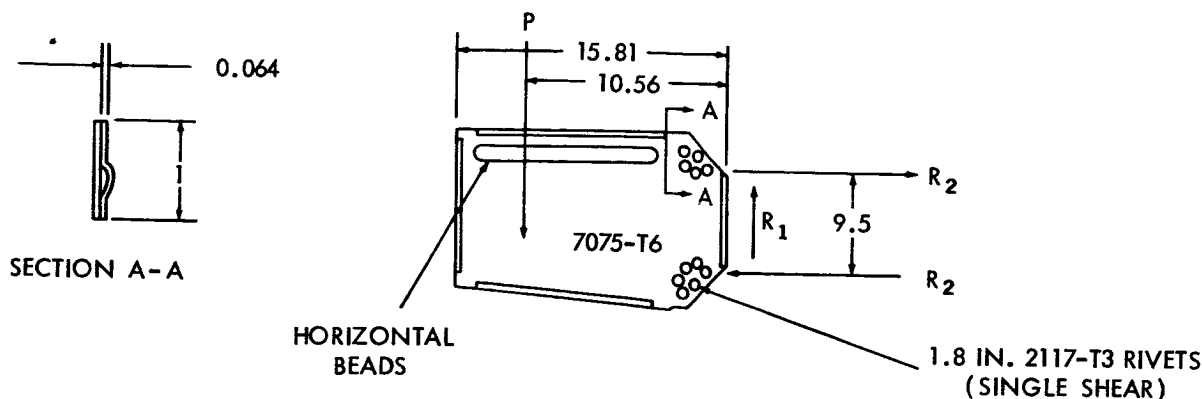
$$20 n_{x_s} W_s + 7.9 n_{x_w} W_w + M = 0 \quad (3.5)$$

$$M = -20 n_{x_s} W_s - 7.9 n_{x_w} W_w \quad (3.6)$$

Item	Units	Condition	
		Max. $q\alpha$	Burnout
W_w	Lb-----	315	315
W_s	Lb-----	145	145
n_{z_s}	-----	-25.7	-29.9
n_{z_w}	-----	8.6	3.5
n_{x_s}	-----	-25.7	-29.9
n_{x_w}	-----	8.6	3.5
P	Lb limit----	11822	13754
V	Lb limit----	3956	1610
M	In.-lb limit--	-46340	18860

Wheel Ribs

Because the wheel had to be spin balanced, it can be assumed that axial loads were distributed equally to each of the nine ribs. It can be further assumed that the load center of mass is two-thirds of the distance from the center wheel casting to the outer rib radius.



$$\sum M=0$$

$$-10.56P + 9.5R_2 = 0$$

$$R_2 = \frac{10.56}{9.5} P$$

$$P = \frac{310}{9} (n_{zw}) = \frac{310}{9} (29.9 \times 1.5) = 1545 \text{ lb ult.}$$

$$R_2 = 1720 \text{ lb ult.}$$

$$R_1 = P = 1545 \text{ lb ult.}$$

The rib is critical for burnout condition.

Effective area for reacting R_2 is Section A-A.

$$\text{Area} = 1 \times 0.064 = 0.064 \text{ in.}^2$$

$$F_{cy} = 62000 \text{ psi (MIL-HDBK-5)}$$

$$f_c = \frac{P}{A} = \frac{1720}{0.064} = 26900 \text{ psi ult.}$$

Assuming that the beaded panels do not buckle, we determine the margin of safety to be

$$\text{M.S.} = \frac{62000}{26900} - 1 = 1.3$$

R_2 must be transferred to the center wheel casting by five $\frac{1}{8}$ " 2117-T3 rivets in single shear.

$$\text{Load/Rivet} = \frac{R_2}{5} = \frac{1720}{5} = 344 \text{ lb ult.}$$

$$\text{Allow. Rivet Shear} = 388 \text{ lb (MIL-HDBK-5)}$$

$$\text{M.S.} = \frac{388}{344} - 1 = 0.13$$

Bearing in rib (0.064 thick)

$$F_{bru} = 105,000 (e/d = 1.5)$$

$$\text{Bearing Allow./Rivet} = 810 \times 1.05 = 850 \text{ lb ult.}$$

$$\text{M.S.} = \frac{850}{344} - 1 = 1.47$$

Shear Load R_1 :

This load is carried through 18 $\frac{1}{8}$ " rivets in single shear.

$$P_{\text{rivet}} = \frac{1545}{18} = 86 \text{ lb}$$

Web Shear:

$$q = \frac{R_1}{9.5} = \frac{1545}{9.5} = 163 \text{ lb-in. ult.}$$

Because of the web beading, the web is not critical.

Decks

The decks were formed from 7075 sheet aluminum, 0.050 inch thick, and heat treated to a T6 condition. There were 5 depressions formed, varying in depth from $\frac{1}{4}$ inch at the wide end to $\frac{1}{8}$ inch at the narrow end. A typical deck is shown in Figure 3-6.

Rim Panels

The rim panels were brake-formed of 6061-T6 aluminum, 0.090-inch thick, with a $\frac{1}{8}$ -inch bend radius. They served the function of holding the ribs laterally, and supporting the wide edge of the decks. The three arm compartments received the full reaction force of the arms during arm extension. A typical rib panel is shown in Figure 3-7.

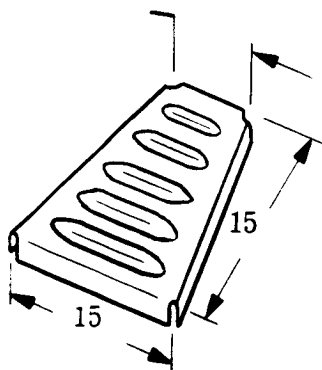


FIGURE 3-6.—Typical deck.

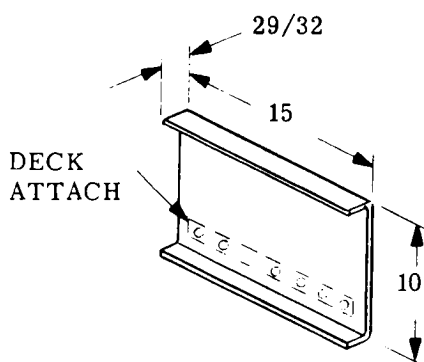


FIGURE 3-7.—Typical rib panel.

The main loads induced in the rim panels were caused by the arms during extension, and to a lesser extent from vibrations when the arms were folded. Peak P_1 and P_2 loads (see Figure 3-8) occurred during arm extension and were on the order of 600 pounds and 300 pounds, respectively. These loads occurred simultaneously for a short duration.

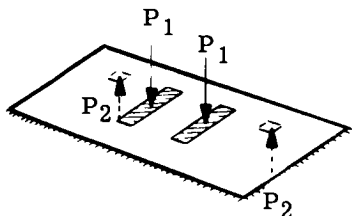


FIGURE 3-8.—Loading on rim panel during arm extension.

Wheel Center Casting Assembly

Figure 3-9 is a free-body diagram of the center casting assembly showing the expected loadings. This casting had to react all loadings from the oriented portion of the spacecraft and from the wheel ribs and deck assembly. In arriving at final loads, the weights and load factors shown on page 3-12 were used in conjunction with the ultimate load factor of 1.5. P_s , R_{us} , and R_{1s} are the loads produced on the

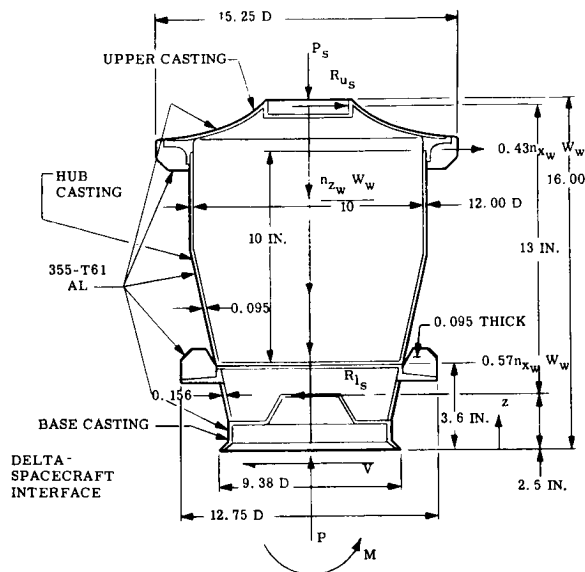


FIGURE 3-9.—Wheel center casting assembly.

casting by the oriented portion of the spacecraft. Critical condition is maximum $g\alpha$.

$$P_s = 145 \times 25.7 \times 1.5 = 5590 \text{ lb ult.}$$

$$P = 11822 \times 1.5 = 17730 \text{ lb ult.}$$

$$V = 3956 \times 1.5 = 5930 \text{ lb ult.}$$

$$M = 46340 \times 1.5 = 6950 \text{ in.-lb ult.}$$

$$R_{us} - R_{1s} = M_z W_s = 8.6 (145) (1.5) = 1870 \text{ lb ult.}$$

$$13 R_{us} = 17.5 n_z W_s$$

$$R_{us} = \frac{17.5 (8.6) (145) 1.5}{13} = 2520 \text{ lb ult.}$$

$$R_{1s} = 2520 - 1870 = 650 \text{ lb ult.}$$

$$P_s = M_z W_s = 25.7 (145) (1.5) = 5590 \text{ lb ult.}$$

$$0.43 n_{zw} W_w = (0.43) (8.6) (315) (1.5) = 1750 \text{ lb ult.}$$

$$0.57 n_{zw} W_w = 2320 \text{ lb ult.}$$

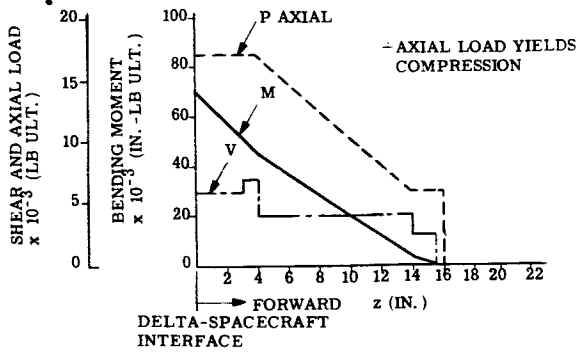


FIGURE 3-10.—Ultimate cross-sectional loads on wheel center casting assembly.

Figure 3-10 shows ultimate cross-sectional loads on the casting. These loads are used in the following stress analysis of the casting.

The equivalent axial load on the casting assembly shell is:

$$P_{eq} = P_{axial} + \frac{2M}{R} \quad (3.7)$$

X	P_{axial} lb	M in.-lb	R in.	P_{eq} lb ult.
0	17730	69590	4.69	47406
3.6	17730	47490	4.69	38834
9	11000	55000	6.0	29333
13.6	5590	4790	6.0	7187

Maximum Stress on Base Casting:

$$f_c = \frac{P_{eq}}{2\pi R t} = \frac{47406}{2\pi(4.69)(.156)} = 10300 \text{ psi ult.}$$

Casting is Compression Critical:

$$F_{cy} = 31000 \text{ psi}$$

$$\text{M.S.} = \frac{31000}{10300} - 1 = 2.0$$

Maximum Stress on Hub Casting:

$$f_c = \frac{P_{eq}}{2\pi R t} = \frac{38834}{2\pi(4.69)(0.095)} = 13900 \text{ psi ult.}$$

$$\text{M.S.} = \frac{31000}{13900} - 1 = 1.23$$

Casting was not buckling critical because of stiffening from rib flanges running the length of the casting.

Fabrication

The dies for hydroforming of decks and ribs were designed by Ball Brothers Research Corporation and procured from a subcontractor. The raw materials and dies were then furnished to the manufacturer of the hydroformed parts. This procedure resulted in BBRC responsibility for proper sizing and fitting of parts. BBRC also controlled the necessary die rework for springback correction and tolerance correction. Considerable liaison and design work were necessary in this task. Heat treatment for these parts to BBRC specification was also procured from a subcontractor. The resulting uniformity of parts was excellent as was control of tolerance.

Patterns for castings were subcontracted using BBRC designs. Castings were also obtained from a subcontractor and were then machined at BBRC. Assembly jigs and fixtures were designed and built at BBRC, as was all of the tooling required. After the first trial fit of stamped parts, the dies were reworked for tolerance correction and sizing. This was done between the time of assembly of the structural test model and the prototype model. The structural model required some shimming and handwork which was largely eliminated on the prototype model.

Fasteners presented some problems. Originally, POP rivets were specified throughout, and were used in the structural model. These break a pin inside the rivet while swage-forming the head. The broken pin inside the rivet cannot be extracted and rattles loosely inside the rivet. All rivets were changed to solid aluminum rivets for prototype and flight models. To reduce the weight, aluminum screws were specified for mounting the cover panels, but it was found that the heads twisted off when used with locking nuts. These were replaced with stainless steel screws.

Other minor fabrication problems were encountered, but the over-all wheel assembly was straightforward and relatively simple. No redesign of the structure was required after

tests, however minor changes in the locations of wiring holes were required.

UPPER STRUCTURE

General

The upper structure of the spacecraft consisted of: (1) the azimuth casting which was mounted on the azimuth shaft; (2) the elevation casting, which carried the pointed experiments and was mounted on bearing supported shafts within the azimuth casting; (3) the sail structure which supported the solar cell array; (4) the azimuth shaft; (5) the elevation motor and bearings; (6) the nutation damper.

The upper structure contained the motorized servo elevation drive system and the precession gas control system, including the precession gas jets. It also supported the upper structure electronic packages and controls. Figure 3-11 is a photo of the sail structure.

Because the moment of inertia of the upper structure added to the spacecraft transverse moment of inertia and contributed nothing to the spin moment of inertia, it was necessary to keep this moment of inertia as small as possible relative to the spacecraft center of gravity. This meant that all large masses were to be mounted as near to the bottom of the upper structure as possible. However, the servo elevation control of the pointed experiments had to permit at least ± 5 degrees of movement of the experiments about the elevation axis. The elevation axis, or pivot point, had to be chosen so that in the depressed position the

experiment field of view would not be occluded by the extended arms nor would the experiment physically strike the wheel structure.

A necessary design requirement for the upper structure was that all of the moving parts be locked during the launch phase. Both the elevation and azimuth motions of the upper structure were locked with squib-actuated pin pullers.

An open framework was used for the solar cell support structure. This was made of 0.025-inch thick aluminum sheet stampings riveted together, and formed a light rigid unit readily adaptable to mounting components as well as solar cells. The sail structure form was determined by the cross-sectional area available in the vehicle shroud.

Since it was necessary to locate the precession jet nozzles as far from the spacecraft center of gravity as possible to develop the greatest torque, they were located at the periphery of the sail within the open framework.

The pointed experiment mounting method and the mounting of the solar detector eye blocks on the pointed experiments were problems which had to be solved early in the design phase in order to provide design criteria and specifications for the experimenter groups. The sail structure design had to await correlation of the total spacecraft power requirements and the solar cell output capability before it could be completed.

The azimuth casting supported several electronic components and control system sub-assemblies. Since the design of the casting was predicated upon final configuration of these components and subassemblies, the final configuration of the casting was the last piece of structural hardware design completed.

The azimuth casting bolted to the azimuth shaft. The shaft was supported by bearings in the wheel assembly as illustrated in Figure 3-12. Axial motion of the shaft was restrained at the upper bearing, and the lower bearing controlled radial play only. The shaft was surrounded by a precession gas storage bottle. The bottle was threaded on the shaft at the upper end and was free to slide on O-ring seals for normal strain deformation at the lower end. A hole was provided in the shaft to allow flow of the gas up

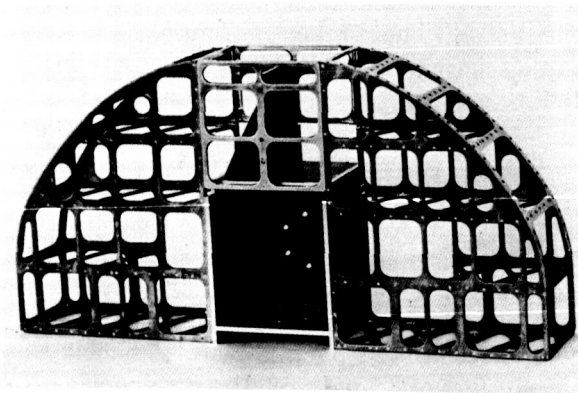


FIGURE 3-11.—Basic sail structure.

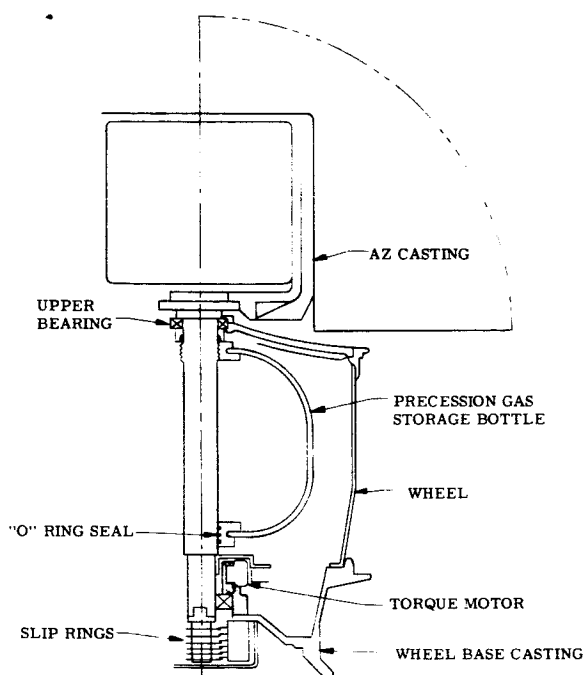


FIGURE 3-12.—Azimuth casting and shaft.

through the shaft to the gas distribution tubing on the azimuth casting. The azimuth drive motor and the slip rings were located at the lower end of the shaft, as shown.

Loads and Stresses

The load inputs were the same as those for the wheel structure. The weights involved are as follows:

Upper section including azimuth shaft items.....	70 lb
Design weight for pointed experiments.....	75 lb
Total section.....	145 lb

Azimuth Bearings

The entire thrust axis load of the upper structure including experiments had to be reacted by the upper azimuth shaft bearing. This amounted to 5590 pounds ultimate. New Departure, a division of General Motors Corporation, publishes a value for "quiet" running limit rating, defined as the safe thrust load on a bearing not running. An excess load over this rating will produce small Brinell marks in the races, and will cause the bearing to become

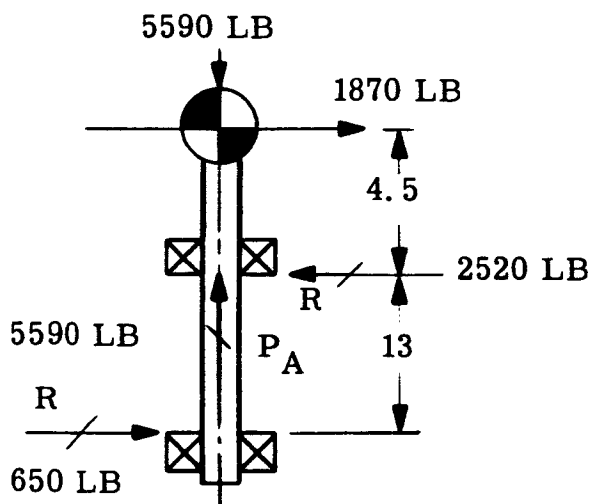


FIGURE 3-13.—Transverse bearing loads.

noisy when subsequently rotated under lighter loads.

A single-row radial, nonnotched-filled, size 3308, 40 mm bore, ball bearing was selected because its size was correct for the space available and because it has a quiet rating of 6830 pounds. The resulting thrust margin of safety for this bearing is 0.22. The radial rating is determined by the geometry shown in Figure 3-13. The upper bearing required a rating of 2520 pounds ultimate, and the size 3308 bearing is rated at 4480 pounds, for a margin of safety of 0.78.

The lower bearing was selected for size and rating, and a single-row, nonloading groove bearing of extra light series, number R-18 was selected. This bearing has a radial rating of 1520 for a margin of safety of 1.34. The bearings were both 52100 steel, and metal ball retainers were replaced with Rulon retainers and given a BBRC Vac-Kote treatment for vacuum operation.

Azimuth Shaft

The only difficulty experienced with the azimuth shaft during qualification tests of the prototype spacecraft was the low torsional resonance response of the shaft. With the azimuth drive torque motor at the lower end of the shaft, the shaft acted as a torsional spring. The spring rate of the shaft used was

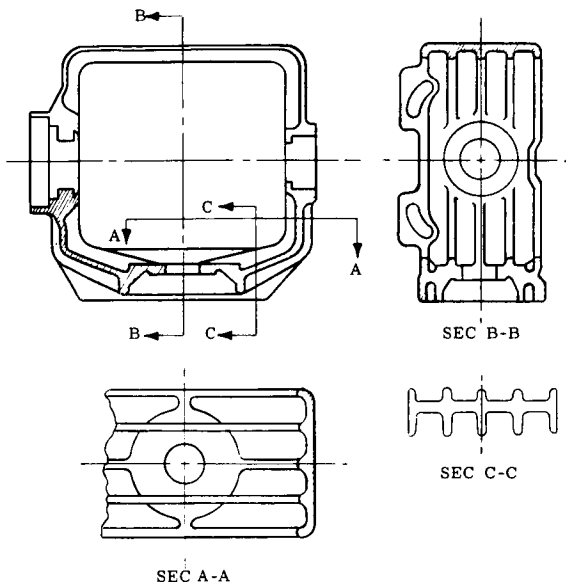
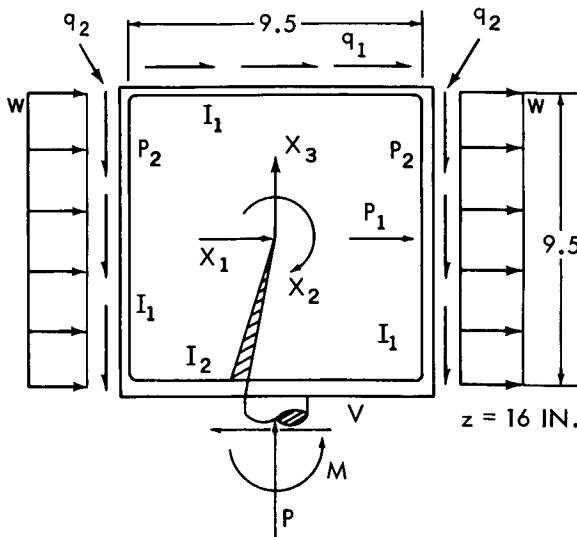


FIGURE 3-14.—Azimuth casting sections.



• MATERIAL: TENS 50

FIGURE 3-15.—Free body diagram of the azimuth casting.

calculated to be 108,000 lb-in/rad, and, with a moment of inertia of the upper section of 1.8 slug-ft², the torsional resonance was calculated to be at 11.4 cycles per second. This resonance caused forced oscillations in the azimuth drive servo system. The gain of the servo system had to be reduced to eliminate the problem. The other obvious solution would

have been to initiate a major redesign and locate the motor at the top of the shaft. This was done on the following OSO program. It was too late in the S-16 program to allow time for this redesign.

Azimuth Casting

The azimuth casting (see Figure 3-14) is critical for axial and transverse loadings resulting from the maximum $q\alpha$ condition. The casting is loaded by the sail assembly and equipment trunnion-mounted to the casting. Assumed loadings are as shown in Figure 3-15.

$$\frac{\partial U}{\partial X_1} = 0 = \int \frac{M \frac{\partial M}{\partial X_1} d_s}{EI_s} \quad (3.8)$$

$$\frac{\partial U}{\partial X_2} = 0 = \int \frac{M \frac{\partial M}{\partial X_2} d_s}{EI_s} \quad (3.9)$$

$$\frac{\partial U}{\partial X_3} = 0 = \int \frac{M \frac{\partial M}{\partial X_3} d_s}{EI_s} \quad (3.10)$$

Center of Gravity of Sail Structure Without Azimuth Shaft:

Approx. shaft weight = 20 lb

Sail structure weight = 145 - 20 = 125 lb

c.g. of azimuth shaft from Delta-spacecraft separation plane is:

$Z_{\text{shaft}} = 11$ in. approx.

c.g. of complete upper structure is:

$Z = 20$ in.

$125Z_{\text{sail}} = 145(20) - (11)$

$Z_{\text{sail}} = 21.44$ in.

From free body diagram (Figure 3-15) and above c.g. data, the following calculations are made:

$$-M + n_x W_s = 0$$

$$M = 125(8.6) \dagger (1.5)(21.44 - 16.5) = 7953 \text{ in-lb ult.}$$

$$P - n_z W_s = 0$$

$$P = (25.7)(125)(1.5) = 4820 \text{ lb ult.}$$

$$-V + n_x W_s = 0$$

$$V = 125(0.6)(1.5) = 1610 \text{ lb ult.}$$

† Load factor (see page 73).

• Weight of elevation casting and pointed experiments=75 lb

Weight of remaining sail structure less azimuth shaft=125-75=50 lb

$$P_1 = (76)(8.6) \dagger (1.5) = 967 \text{ lb ult.}$$

$$P_2 = \left(\frac{75}{2}\right) (25.7) \dagger (1.5) = 1446 \text{ lb ult.}$$

$$q_2 = \frac{(50)(25.7) \dagger (1.5)}{2(9.5)} = 101 \text{ lb-in.}$$

$$\sum M_0 = 0$$

$$(2)(9.5)(4.75)w + (9.5)^2 q_1 + 4.75 P_1 - M = 0$$

$$90.25w + 90.25q_1 = 3360$$

$$w + q_1 = 37.2 \quad (3.11)$$

$$\sum F_z = 0 \text{ (sail and solar cells only)}$$

$$(2)(9.5)w + 9.5q_1 = 50(8.6) \dagger (1.5)$$

$$2w + q_1 = 67.9 \quad (3.12)$$

From Equations (3.11) and (3.12) we determine that:

$$w = 30.7 \text{ lb/in. ult.}$$

$$q_1 = 6.5 \text{ lb/in. ult.}$$

Solution of Equations (3.8), (3.9), and (3.10), using values for I_1 and I_2 of 0.0212 in.⁴ and 0.26 in.⁴, respectively, yields values for X_1 , X_2 , and X_3 :

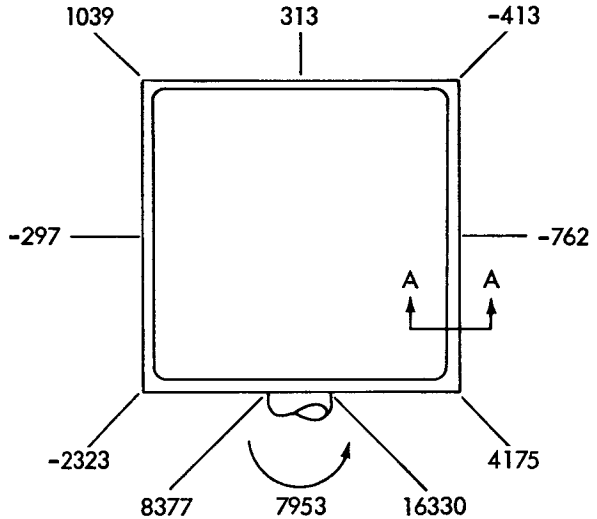
$$X_1 = -500 \text{ lb ult.}$$

$$X_2 = 11457 \text{ in.-lb ult.}$$

$$X_3 = 2255 \text{ lb ult.}$$

†Load factor (see page 73).

Final ultimate bending moments (in.-lb) are shown in the diagram below.



NOTE. Positive moments yield compression in outer fiber.

The casting is critical in bending on the vertical leg (Section A-A, see Figure 3-14).

Bending moment=4175 in.-lb ultimate
Tension ultimate for TENS 50=40,000 psi
Under plastic bending of the material at Section A-A, the maximum allowable bending moment is:

$$\text{Allow B.M.} = 4500 \text{ in.-lb}$$

$$\text{M.S.} = \frac{4500}{4175} - 1 = 0.077$$

The structural model was built and tested with the azimuth casting configuration shown in Figure 3-14. Strain gages were utilized on both the azimuth and elevation castings during steady-state acceleration tests. The maximum stress obtained was found on the outer edge of a rib near a sail attach point. The remaining gages located on webs showed negligible stress levels.

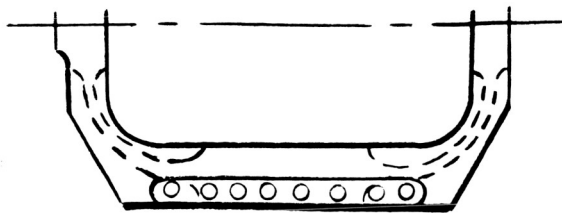


FIGURE 3-16.—Flange with holes added.

Between the fabrication of the structural model and the fabrication of the prototype model of the casting, a design change was made adding eight holes for electrical components in the outer bottom flange of the casting as shown in Figure 3-16. Four holes were $\frac{5}{16}$ in. in diameter, and four were $\frac{1}{4}$ in. in diameter. The first indication of trouble was found after prototype model vibration tests, when a crack was found in the casting from a hole to the outer edge of the flange.

At this point, the flight model assembly was fully assembled and wired. With this fact in mind, it was decided to use the design regardless of the crack. The justification of this decision was based on two factors:

1. *Vibration input levels on the prototype were one and one-half times expected flight levels. During actual flight the superimposed thrust was unidirectional, loading the flange in question in compression and reducing the tension levels which caused the crack to occur.*

2. *Three additional sinusoidal vibration tests were performed on the prototype model, plus a random vibration test, all with the same casting, and the casting still did not fail.*

After vibration tests of the flight model spacecraft, the cover over the flange-mounted components was removed, and the paint on the casting between the holes and the edge of the flange was removed. No cracks were found.

Elevation Casting

The elevation casting (Figure 3-17) was designed to carry an 80-lb instrument load. Strain gage measurements were made during structural model acceleration tests and indicated very low stresses for the 49 pounds of dummy experiments.

During servo checkout tests conducted with dummy experiments on the prototype model,

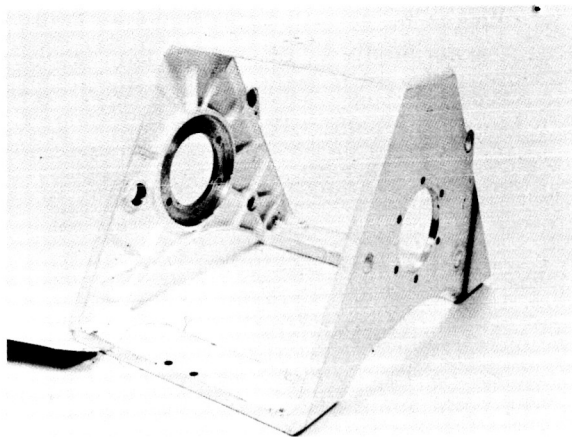


FIGURE 3-17.—Elevation casting.

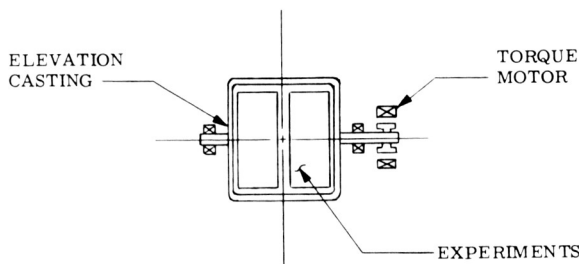


FIGURE 3-18.—Torsional loading of the elevation casting.

it was found that a low torsional frequency mode existed in the elevation casting. From the front, the casting forms a box, with one dummy instrument mounted to each side of this box (see Figure 3-18). Whenever the torque motor was driving, the dummy instrument on the opposite side would lag and then oscillate. The torsional spring rate of the casting was measured and then the casting was redesigned by relocating and adding ribs. No depth of section could be added without redesigning the azimuth casting or reducing the size of the experiments. The stiffness was increased by a factor of 3. The problem still existed and the dummy experiments had to be tied together through damping material during servo tests, or the servo gain had to be reduced.

When the prototype experiments were received and installed, it was found that the amplitude of the oscillation was far less than it had been with the dummy experiments since

the motion was now well damped by the experiments themselves. The problem was much reduced, and the eventual solution was to slightly reduce servo gain at the expense of somewhat reduced pointing accuracy.

Elevation Bearings

The elevation bearings were selected using the same criteria as the azimuth bearings, with different loads. These bearings carry the elevation casting and the pointed experiments for a total of 84 pounds.

These bearings (New Departure type 3305) also were disassembled and treated with the BBRC Vac-Kote process prior to installation.

Sail Structure

The sail structure was a semi-truss type structure of thin sheet aluminum. From front and back it resembled a truss, but from the sides it did not. It was loaded with its own weight of 17.8 pounds, with 5.5 pounds of solar cells, and with some components of low mass.

The structural model sail assembly was tested independent of the spacecraft and only one significant modification resulted. When vibrated in the thrust axis, deflections of transverse tie members were such that the cross axis motion exceeded the thrust motion. To overcome this condition, stiffeners were added.

Originally, it was decided that the solar cells would be mounted in sections on separate subpanels, which in turn mount on the sail structure. The main advantage of this scheme lay in ease of replacement of a small group of cells. The subpanel modules were approximately four by six inches, and were press-formed with a pattern to increase stiffness.

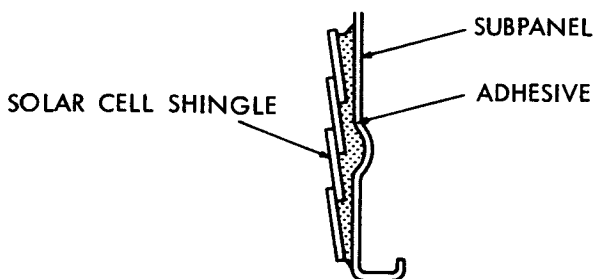


FIGURE 3-19. Solar cell sub-panel cross-section (original design).

The first set of solar cells was installed by applying the adhesive to the panels first, and then placing the shingles of cells in place. The result was that the stamped depressions in the subpanels were filled with the adhesive (see Figure 3-19).

When the units were subjected to thermal changes, the greater mass of adhesive in the depressions expanded more than the thin layers, thus breaking the solder joint between cells and causing open circuit failures. Efforts to correct this condition and salvage the cells were unsuccessful, and the subpanel system had to be abandoned. In its place, sheets of honeycomb material with a smooth, flat surface were used. As a further precaution, the method of bonding the cells was changed to that of placing the adhesive on the cell shingles first. The number of fasteners between the structure and the panels was reduced, and Teflon bushings were used with the fasteners to allow relative motion. The honeycomb panels were tested in vibration on the sail structure. This design proved acceptable, and no further problems occurred.

Nutation Damper

The nutation damper consisted of a pendulum bob submerged in a damping fluid and enclosed by an aluminum case. Nutation caused a point on the upper section to be translated in a circular path in a plane perpendicular to the spin axis.

The reaction of the pendulum-type damper to the nutational input was to absorb energy as heat through the viscous action of the fluid in the damper. The design parameters were the forcing function frequency, the mass of the bob, the natural frequency of the bob and wire pendulum, and the damping constant of the fluid.

The damper was derived analytically and then tested on a special test fixture and on the air bearing table mockup. When proven satisfactory, the design was finalized for flight hardware with the result as shown in Figure 3-20.

Caging the bob for the launch phase presented a difficult problem. Several methods were tried, but during tests the pendulum wire was permanently deformed. Serrating the caging

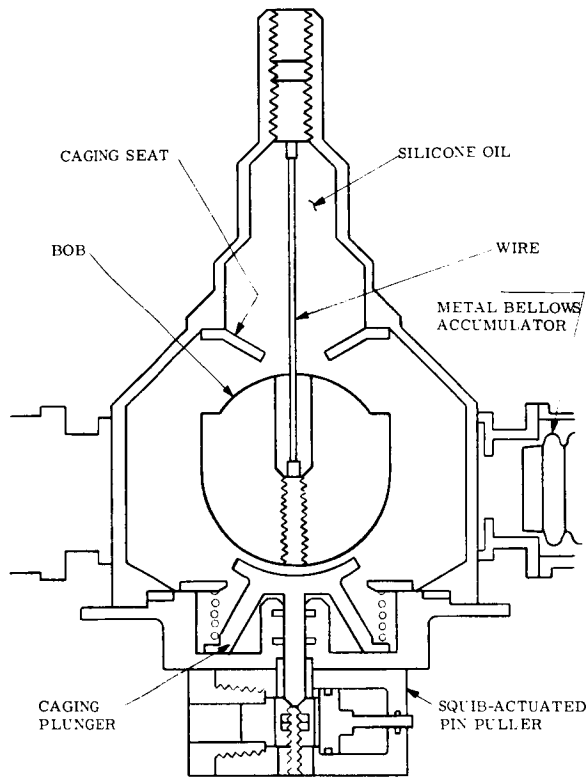


FIGURE 3-20.—Nutation damper assembly.

seat and adding three needle pickup points to the plunger caging surface solved the problem.

Earlier versions of the assembly utilized rolling rubber diaphragms, or Bellophrams. The final version used a piston with O-rings for caging and metal bellows for accumulators. The rubber diaphragms were abandoned when it was realized that they were permeable to gases and would permit bubbles to form in the liquid. The metal bellows act as volume compensators during temperature changes.

Fabrication

Sail Structure

The sail structure was fabricated from stampings of 0.025-inch-thick 2025 aluminum sheet and heat treated to a T-4 condition. The dies were designed by BBRC and built elsewhere. The sheet stampings were entirely riveted together. No particular problems were encountered during fabrication or assembly of the units.

Castings—Elevation and Azimuth

The elevation casting design was completed in time to allow sufficient time for pattern making, casting pour, and heat treat at local sources. The azimuth casting, being last out of design, was rushed for pattern and foundry time. Furthermore, the pattern was very complicated and difficult to fabricate since all surfaces on the casting were cored. Even after heat treated castings were on hand, considerable machining time was required.

Fabrication of Other Items

Fabrication of the azimuth shaft, elevation shafts, and various fittings was straightforward and presented no difficulty. The fabrication of the nutation damper was complicated by the need for pressure-tight, aluminum-welded seams. During assembly of the nutation damper the blind adjustment for caging the pendulum without deforming the pendulum wire, and the vacuum filling with silicone oil without leaving any bubbles, were time consuming and difficult.

COMPONENT MOUNTING

The design requirement for mounting components in the wheel structure and on the pointing section was treated as part of the overall structural design. There were five experiments in the compartments of the spacecraft wheel, all of various size and weights. These had to be balanced against each other and against the four compartments containing telemetry equipment and control items, to keep the wheel in both static and dynamic balance about the spin axis. To do this required a continuous effort throughout the program from the time the weight of the first component was in until the last experiment was weighed and center of gravity determined. To maintain dynamic balance, both components and experiments had to be raised off the wheel compartment decks and brackets had to be designed to mount them at the necessary heights.

For light weight and ruggedness, a basic sheet metal zee bracket design was utilized wherever feasible (Figure 3-21). Buckling

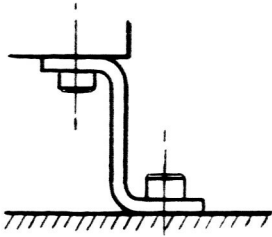


FIGURE 3-21.—Zee bracket mounting.

loads and eccentric loading for these brackets were computed. Either three or four such brackets were used for each item, so that lateral support was provided regardless of the direction of vibration input. Where only three were used, it was found that the angles relative to each other were critical. Some high amplifications of displacement were found due to rocking modes of vibration, and redesign was necessary to eliminate this condition. The zee brackets, being different sizes for different mounting applications, served to decouple the experiments and equipment from each other, since each mounted item had a different resonant frequency. In fact, each bracket supporting an item could be made to have a different resonant frequency to overcome the disadvantage of high amplification. Typical zee brackets are shown in Figure 3-22.

The tape recorders were the only items which were mounted on isolation mounts. The tape recorders were designed to meet the wheel component specification of 15.6 g from 50 to

1000 cps of sinusoidal vibration. However, during structural model spacecraft tests, it was found that the wheel had a resonance at 120 cps, with a transmissibility of 3 to 5, with a prototype input specification of 10.7 g from 50 to 500 cps. It was decided that the tape recorders could not take this level and would be isolation-mounted so that they would never be subjected to any inputs greater than the component specifications. It was realized that the mounts would permit relative movement between balancing operations in a gravity field and performance in a zero g field, but calculations showed that the tape recorders' motions would cause an error less than 25 percent of the allowable balancing error.

In general, vibration mounts are not designed to perform under variable superimposed thrust loads. Characteristic curves by the Lord Manufacturing Company for their BTR mountings showed a potential for this type loading, and so they were tested. A dummy test setup was made of the tape recorders (see Figure 3-23), using six mounts with their elastic centers intersecting as near the center of gravity of the package as space would allow. This required three mounts on the bottom and three around the upper edge. A long, relatively soft spring was placed on top to duplicate the thrust loading. Different size mounts were tested, but all of them bottomed out and put shock loads on the simulated recorders. Springs were added under-

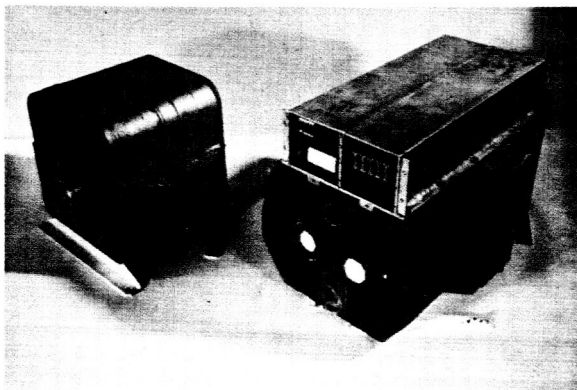


FIGURE 3-22.—Typical zee bracket mountings.

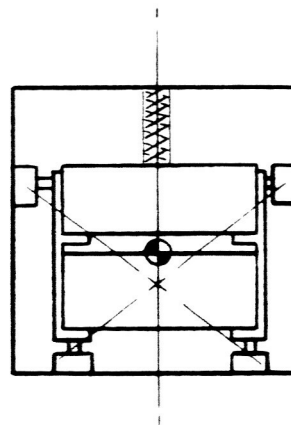


FIGURE 3-23.—Tape recorder vibration isolation test setup.

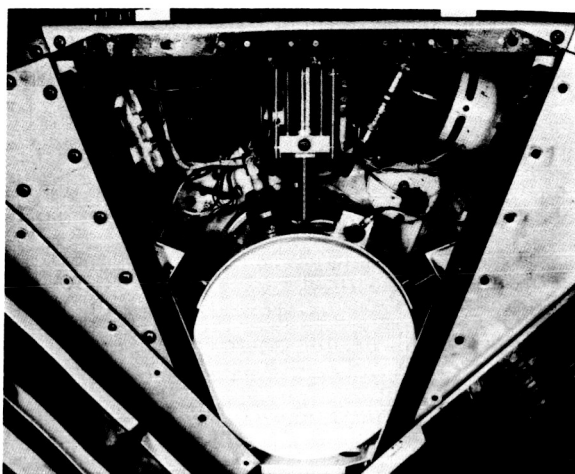


FIGURE 3-24.—Tape recorder mounting and arm damper.

neath the recorders to reduce the differential loading. These were calculated to provide a 3 g lift against the 12 g thrust, for a differential of only 9 g. More lift could not be added because the mounts crept with time under load. This design eliminated the bottoming of the mounts both with and without the thrust spring, and was incorporated in the final design. Figure 3-24 shows the tape recorders mounted in the wheel compartment.

ARMS AND ARM DAMPERS

General

The prime consideration in the design of the extendable arms was the achievement of the proper ratio of the spin moment of inertia to the transverse moment of inertia. The length of the arms for the OSO I was selected on the basis of preliminary calculations of these moments. The final measured ratio was 1.35 to 1.

Even though the arms could be released as soon as the shroud separated from the launch vehicle, it was preferable that they not be released until after termination of third stage thrust so that they would not need to withstand acceleration forces in the extended position. Therefore, the time of extension was selected to occur after third stage burnout and before third stage separation. With the arms extended, there was ample clearance for the third stage bottle separation.

After third stage burnout, the spacecraft and third stage engine case had an angular velocity of $120 \text{ rpm} \pm 10 \text{ percent}$. An added benefit from extending the arms was the reduction of spin rate due to the increase of the spin moment at the time the arms were extended.

It was decided early in the OSO I program that the three arms should be physically tied together so that if any one failed to fully extend, the others would remain in symmetry, and the dynamic balance of the spacecraft would be maintained and pointing control achieved. Calculations indicated that with the extension torques involved, centripetal force alone was more than sufficient to extend the arms and, in fact, the arms must be restrained during extension. This fact led directly to the concept of a hydraulic damper system which was series-connected to each of the three arms. A folding linkage was placed on each arm to position the arms when extended. (See Figure 3-25.) The linkage provided a convenient way to lock the arms down in the stowed position and provided a positive position lock for the extended position.

The design was further modified when it was decided to utilize the arm linkage as a part of the spacecraft antenna system. The linkage had to be electrically insulated from both the arm and the wheel, and had to provide an adequate mechanical lock for ground operation.

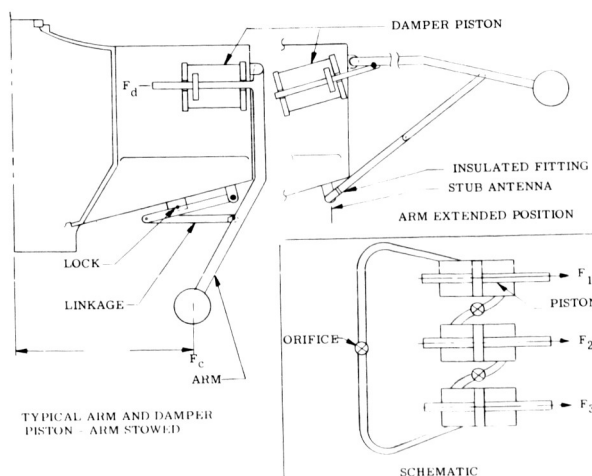


FIGURE 3-25.—Arm and arm damper system.

It was necessary to join the three gas bottles to a common manifold so that equal weight could be maintained in each bottle thus preserving the dynamic balance of the spacecraft.

Provision was made to run a high pressure (3,000 psi) and two low pressure (60 psi) gas lines through the hinge joint of the arm. The low pressure lines supplied the spinup and despin gas jets which had been placed at the ends of the arms to provide maximum torquing.

The problem of a rotating high pressure joint was solved by simply twisting a length of 1/8-inch diameter copper tubing in torsion through the 90 degree rotation. A test unit was built and a piece of tubing was cycled for 400 bends with 3,000 psi internal pressure, without failure. It was estimated that the arms on the flight model would be cycled as many as 40 times, but not more than 10 times with full pressure. The low pressure lines were simply flexible tubing run through slots into the arm hinge end.

Loads and Stresses

The maximum loads on the arms that occurred during arm extension will be discussed first. Later, the loads due to vibration inputs and thrust will be reviewed.

The typical arm and arm damper assembly in Figure 3-25 shows the geometry of an arm in the stowed position. The forces acting on the arm at the instant of release are:

$$\text{Centrifugal force } F_c = Mr\omega^2 \quad (3.13)$$

$$\text{Damper force } F_d = PA \quad (3.14)$$

where

M = Mass of arm and gas bottle

r = Radial distance from spin axis to center of gravity, inches

ω = Initial angular velocity of the wheel in radians per second

P = Fluid pressure within the piston cylinder, lb/in.²

A = Piston area, in.²

From conservation of angular momentum for the spacecraft about the spin axis we may write

$$I_s\omega = I_2\omega_2 \quad (3.15)$$

where

ω_2 = instantaneous velocity as the arms extend

I_s = initial spin moment of inertia

I_2 = instantaneous moment of inertia as the arms extend

The differential equation of motion for this system may be represented by the following equation:

$$I_a\ddot{\theta} = M_c = M_d \quad (3.16)$$

where

I_a = moment of inertia of the arm assembly about its support hinge point

θ = angle of rotation of the arm from the stowed position

$\dot{\theta}$ = angular velocity of the arm

$\ddot{\theta}$ = angular acceleration of the arm

M_c = moment exerted about the arm hinge due to centrifugal forces

M_d = moment exerted about the arm hinge due to the damper

In the differential equation above the terms M_c and M_d are both nonlinear in θ and ω_2 , which are functions of time. This equation does not lend itself to closed-form analytical solution.

To control the time duration of extension, an orifice had to be provided between each damper cylinder, and the size of the orifice determined. For a piston of specified size and stroke, the volume of fluid to be moved is known. The problem reduces to a fluid flow determination, where

$$Q = 0.61A \frac{2P}{\rho} = \text{flow, ft}^3/\text{sec} \quad (3.17)$$

0.61 = Constant for a broomy flow orifice

A = Area of orifice to be found, ft²

P = Pressure, lb/ft²

ρ = Mass density, slug/ft³

When alcohol was used for the hydraulic fluid, a 0.019-inch diameter orifice was required to allow for the volume flow in 1.50 seconds. This was a practical value which could be realized by a simple drilled hole.

Since the dampers form a closed hydraulic system, thermal cycling could cause internal pressure changes which could limit the design values. The differential expansion between the fluid and the container system was calculated and converted to pressure by using the compressibility factor for alcohol. Assuming that the system was filled at low temperature and the maximum rise was 35°C, the pressure was found to be 5000 psi.

In establishing the design load for the hydraulic system, there was one other factor investigated. As mentioned, the system was to provide a mechanical tie between the arms in case one arm hung up. If one arm did hang up, there would be no fluid flow through the orifices, and under static conditions the pressure would become twice the potential maximum seen by any one cylinder, because the arm which was hung up would exert no pressure on the system. Since the maximum pressure per cylinder was calculated to be 1750 psi, this design criterion is less than that for thermal expansion. Thus, 5000 psi was established as the design value for the hydraulic damper system. An arm damper cylinder may be seen in Figure 3-24.

Arm Stress During Extension

Since the damper piston was very closely coupled to the hinge of the arm, and the arm section was heaviest in this region, the arm very closely approximated a cantilever beam during extension. Maximum moment occurred when the arm was extended approximately 30 degrees open, and was normally 143 ft-lb. Section moduli were computed for various points along the arm and stresses were calculated to range from 12,000 psi to 5660 psi. Based on a yield strength for 2024-T4 aluminum, this design provided a margin of safety of 2.0 to 7.1. The arm was fabricated in two sections and bolted together (see Figure 3-26). The section was

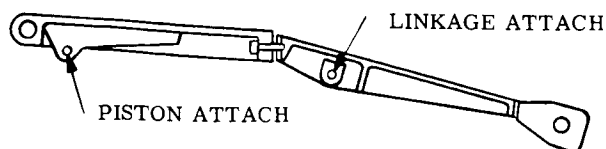


FIGURE 3-26.—Typical arm construction.

heavier at the bolts, and this was where the low flexure stress occurred. The remainder of the arm was very nearly a constant stress section.

The safety margin for the arm flexure stress was raised for several reasons. First, the calculated loads were based on nominal torques at nominal spin rates. The torque increase with wheel spin velocity was not linear. A 10 percent velocity increase raised torque by 21 percent. This reduced the margin of safety to 1.47. Second, the two arm sections were machined from plate stock, and a fairly loose dimensional tolerance was desirable.

The lug that was bolted to the arm web and tied to the piston had a margin of safety of 2.1 in shear and 1.6 in tension. The piston clevis pin which mated with the lug was 4130 cold-drawn steel having a margin of safety of 2.5 in shear and 4.7 in tension. Other detail parts were stressed similarly.

Vibration Stress

For vibration, the arm was again treated as a simple beam and the transverse input of 11.8 g random vibration used for load criteria (Figure 3-27). Checking various points along the arm,

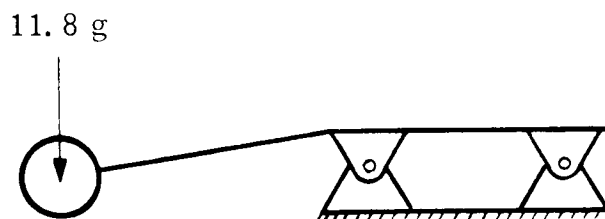


FIGURE 3-27.—Equivalent beam load.

the maximum flexure stress was found to be 10,000 psi, resulting in a margin of safety of 3.6. Using the endurance limit, the margin was reduced to 1.8, based on 5×10^8 cycles. However, this level of input is reached only during random transverse testing for four minutes in each axis, and not enough cycle reversals occur to justify this rating.

The vibrational loads are directly transferred to the pin connections on the linkage. The stresses were calculated and the minimum margin of safety involved was found to be 4.3.

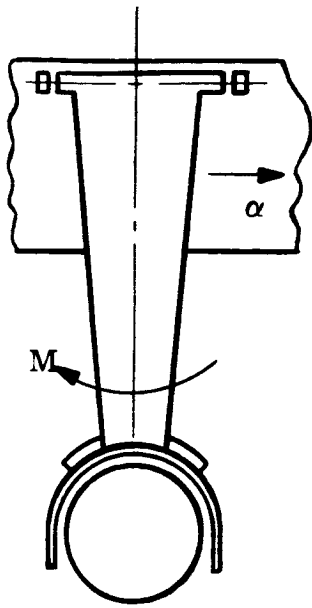


FIGURE 3-28.—Spin acceleration loading.

Spin Acceleration Stresses

During spin acceleration of the third stage booster, the arm hinges were loaded in shear as shown in Figure 3-28. The spin acceleration was specified by Douglas Aircraft Company to be radians per second per second. The margin of safety for the hinge pins in shear was calculated to be 5.0.

Hydraulic System Stresses

The tie bolts were 303 stainless, the cylinder was 304 stainless, the end caps were 2025-T4 aluminum, and the piston was 416 stainless steel.

Based on a maximum load of 5000 psi internal pressure, and the yield strength of the materials involved, the detail parts had the following margins of safety:

Tie bolts.....	1.28 (tension)
Cylinder.....	1.07 (burst)
Aluminum ends...	12.2 (shear about O-rings)

Fabrication

Considerable difficulty was encountered in the fabrication of the arms, resulting in a major design change.

The original design utilized aluminum tubing as structural members and had an all-welded construction. Assembly weld jigs were made

and the first set of arms was procured from an external source. To maintain the required strength, these arms were heat treated after welding.

The first set was tested on a dummy wheel model on a spin table. It showed that the web area where the damper piston attached to the arm was weak. The web was reinforced and further tests proved successful. Additional welded arms were ordered from suppliers and when received were found to be warped and out of tolerance. A new source was tried with the same result. It was found that warpage occurred primarily in the heat treating operation.

At this point, the design was changed to an arm fully machined from plate stock. In redesigning, the same section modulus and weight were maintained for a different geometry. Fabrication proceeded without further trouble, and the arms produced were uniform and within tolerance specification. Further tests were made to fully qualify the latter design.

No difficulty arose in the fabrication of other parts of the system except for the insulated tie member between the linkage and the wheel. First, an insulated bushing system was tried, but this allowed too much electrical capacitance between the charged linkage and the grounded wheel. Then a fitting in the form of a tee was made from laminated fiberglass board. This, however, was not strong enough in shear across the laminations.

Finally, a design was incorporated using fiberglass epoxy laminations where the laminations were not in shear (Figure 3-29). A heated, closed metal mold was used to produce the section desired. The electrical and mechanical performance was satisfactory.

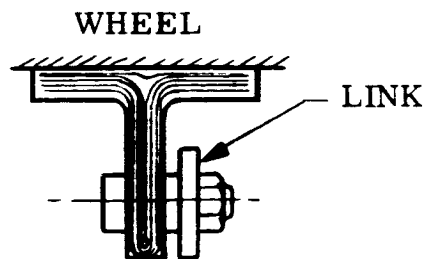


FIGURE 3-29.—Laminated tee section.

Chapter 4

CONTROL SYSTEMS

INTRODUCTION

The basic approach to the problem, that of providing a stabilized solar-oriented space platform from which scientific instruments can continuously monitor the sun, was to utilize the inherent gyroscopic stability of a rotating body and to orient a separate structure which could be shaft-mounted to the spinning body. The separate structure could then be oriented to the sun by means of a gimballed axis arrangement. The control problems then were:

1. Control the rate of spin of the rotating body and the orientation of the spin axis of the rotating body with respect to the sun.
2. Control the gimballed axis orientation of the oriented structure with respect to the sun.

Problem (1) was solved with gas control systems for spin-rate maintenance and for spin-axis orientation. Problem (2) was solved with electrical servo-mechanisms, one for each of the two gimballed axes, deriving orientation information directly from the sun. The following paragraphs describe in a general way how the control systems performed the orientation task. The remainder of this section describes in detail the control system parameters, design, and mechanization.

The OSO I spacecraft utilized the entire vehicle as a controlled platform. Coarse elevation or pitch control of the platform was accomplished by the ON/OFF pitch gas control system. By exhausting nitrogen gas through nozzles mounted on the stabilized upper section, torques of either sense were produced about the roll axis. These torques, which were applied when the spacecraft upper structure was oriented, precessed the spacecraft spin

axis about the pitch axis. The pitch gas control system maintained the spin axis perpendicular to the solar vector within about three degrees. No control about the roll axis (sun vector) was provided, but rates about this axis were slow due to the rigidity of the gyro.

Fine elevation positioning of the pointed instruments was accomplished by an electrical servo called the *elevation servo*. The elevation servo motor was mounted on the casting which supported the pointed instruments.

Prior to third-stage engine firing, the entire third stage and spacecraft with arms in the *stowed* condition were spun up to approximately 120 rpm by a system of small rocket motors. This spin-up started the operation of the spacecraft sequencing timer at *zero* time.

At approximately 100 seconds (sequencer time), and after third-stage engine burnout, the arms were released and, due to centripetal force, raised and locked in the extended position. Shortly after this time, the spacecraft was separated from the third stage.

At 200 seconds the spin rate of the spacecraft, with the wheel and upper section mechanically latched together, was reduced to approximately 30 rpm by the action of the despin jets, located at the ends of the arms. After completion of this despin, the despin system was permanently deenergized.

Turn-on eyes, a set of six eyes mounted on six of the nine sides of the wheel, were used to actuate electrical equipment each time the spacecraft emerged from the earth's shadow. The same eyes were used to turn the electrical servos and the pointed experiments off during the satellite night.

This spin rate was maintained within $\pm 5\%$ of nominal value by spin-up jets located at the

ends of the extended arms. The jets were actuated by a signal from the electronic *spin control system* which computed the instantaneous period of rotation of the wheel relative to the sun. The primary sensors for this system were two *spin eyes* connected in parallel and mounted on the rotating wheel. The spin eye output was a narrow pulse generated as the eye swept past the sun once per wheel rotation.

At 400 seconds (sequencer time), the azimuth release squibs and the nutation damper release squibs were fired and power was applied to the azimuth servo.

The azimuth servo actuating the azimuth DC torque motor drove against the wheel to despin the upper section and point it at the sun. The azimuth servo position error signals originated from two types of detectors. These were coarse eye detectors and fine eye detectors.

There were four coarse eye detectors, each having a 90-degree field of view, giving a full 360 degrees of position control about the spin axis. They were mounted to the solar array support structure and had a stable null perpendicular to this structure and an unstable null in the antisolar direction. Their field of view was limited to 15 degrees above and below a plane perpendicular to the spin axis. This limiting of the coarse eye field of view eliminated the periodic signals caused by sunlight reflecting into the coarse eyes from the three rotating spin gas bottles and reduced any reaction that could be caused by the earth's albedo. This also means that the spacecraft could not acquire the sun if its spin axis exceeded 15 degrees from a plane perpendicular to the solar vector.

The composite coarse eye amplitude curve was such that its first derivative provided a favorable despin signal for approximately 80 percent of an upper structure rotation. As the upper section spin rate was reduced to zero, the coarse eyes point the upper section to within 2 to 3 degrees of the solar vector. This acquisition of the sun had to be accomplished every spacecraft dawn.

Coarse pointing of the upper section permitted the operation of two additional systems: the azimuth fine servo control system and the pitch control system.

The pitch control eyes were detection units for the pitch control system. This block of four eyes was mounted on the side of the stabilized upper structure facing the sun. An 8-second time delay was incorporated into the pitch control circuitry to insure that the upper section was pointed at the sun before using pitch gas. For these conditions, and with the spacecraft spin axis exceeding 3 degrees from the normal to the solar vector, one of these eyes turned on the appropriate jet to precess the spacecraft back toward the correct position. Two of the eyes were needed for this function, one of either sense. The other two eyes turned off the jets whenever the spacecraft attitude had been overcorrected one degree in the opposite direction.

An unstable condition could have resulted without a time delay in the pitch control system. This could have occurred for a situation in which the upper structure was pointed at the sun and the spacecraft had a large nutation or wobble angle, such that the eyes would call for pitch correction during only part of the wobble period. The gas was called for when the spacecraft spin axis was at its maximum error relative to the solar vector. If no gas was used and if the nutation damper should have failed to work, the maximum error would have remained the same. With the nutation damper working properly, the wobble or nutation amplitude was reduced to zero and no problem existed. If pitch gas was expended only near the portion of maximum error, the amplitude of any wobble or nutation was increased for each impulse of pitch gas and the spacecraft spin axis nutated in an ever-widening cone. The time delay, as noted above, prevented pitch gas being expended unless the upper section was oriented, and furthermore prevented an unstable condition if the nutation damper should have failed.

At the time that the coarse azimuth servo had pointed the upper section to within 2 to 3 degrees of the solar vector, a disabling eye mounted on the pointed instruments actuated a relay that turned the coarse control off, leaving the azimuth positioning servo with fine eye control only.

The disabling eye had a field of view of ± 6 to ± 8 degrees in both the azimuth and ele-

vation directions. A 1- to 2-second time delay was used in conjunction with the disabling circuit to prevent transients in the coarse servo. These transients would have resulted if the servo were disabled for a short time each time the upper structure faced the sun during acquisition.

The coarse servo was disabled during the fine pointing mode in order to eliminate the errors caused by the coarse eyes seeing the earth and those errors caused by differences in null position between the coarse and fine eyes.

There were two azimuth fine eye detectors with a field of view of ± 10 degrees. The eye amplitude curve away from the null position had been shaped to minimize hunting during acquisition when large initial position errors and velocities were encountered.

The azimuth fine servo positioned the pointed instruments to within one arc minute of the center of the sun. Long-term accuracy was better than 2 minutes of arc.

After the nutation damper was unlocked at 400 seconds (spacecraft sequencer time) and the upper structure was oriented at the sun, the nutation damper damped out nutation due to the wobble caused by the burned-out third stage, the unsymmetrical thrust of the separation spring, and the unsymmetrical motion of the arms as they were released and swung out. A nutation of several degrees, which could have been caused by these effects, would have caused high power usage by the servo system and therefore had to be quickly removed.

The nutation damper was provided to remove these undesirable motions. The damper worked on the principle that if energy is removed from a freely rotating body, the body tends toward a rotation about the axis of maximum moment of inertia. This axis was made to coincide with the azimuth shaft axis by careful balance before launch.

The nutation damper was mounted to the oriented upper section on the azimuth shaft axis and consisted of a tuned pendulum moving in silicone oil. The frequency of the pendulum determined by the mass of the bob and the spring constant and length of the wire, was tuned to coincide with the nutation frequency. The damping constant was small so the bob

moved through large amplitudes for small nutations. The damper was very effective for nutations small enough that the bob did not hit the walls of the container (less than $\frac{1}{2}$ degree). For these small amplitudes the damping time constant was about 10 seconds. At greater amplitudes the damping rate appears to have been a constant of approximately one degree per minute.

Another cause of nutation during the regular orbiting condition was the unbalance of the pointed instruments due to the movement of internal photodetector scanning mechanisms. The detectors were placed at some specific point for ground checkout balance of the spacecraft. However, during orbit operation, the detectors moved from the balance position. No nutation occurred during the time the instrument was pointed at the sun, but as soon as the pointing control turned off at night and the upper structure began to spin, a dynamic unbalance occurred that caused a wobble. When the servo controls pointed the experiments at the sun the next spacecraft day, the wobble was then converted to a nutation which had to be damped out.

At 800 seconds (spacecraft sequencer time) the elevation gimbal was released and the elevation servo pointed the instrument in elevation.

The fine eyes for the elevation servo were mounted to the pointed instruments and were the same type as the azimuth fine eyes. There were no elevation coarse eyes since the precession control system provided the coarse elevation control.

The Orbiting Solar Observatory is shown in Figures 4-1 and 4-2. Figures 4-3 through 4-8 show the launch sequence events and schematic representations of the various subsystems as they were integrated into the spacecraft

ELECTRICAL CONTROL SYSTEMS

Servo Analysis

There were two independent servo systems, one for azimuth control of the upper structure and one for elevation control of the oriented instruments.

The azimuth servo operated about the spin axis of the spacecraft and was used to hold

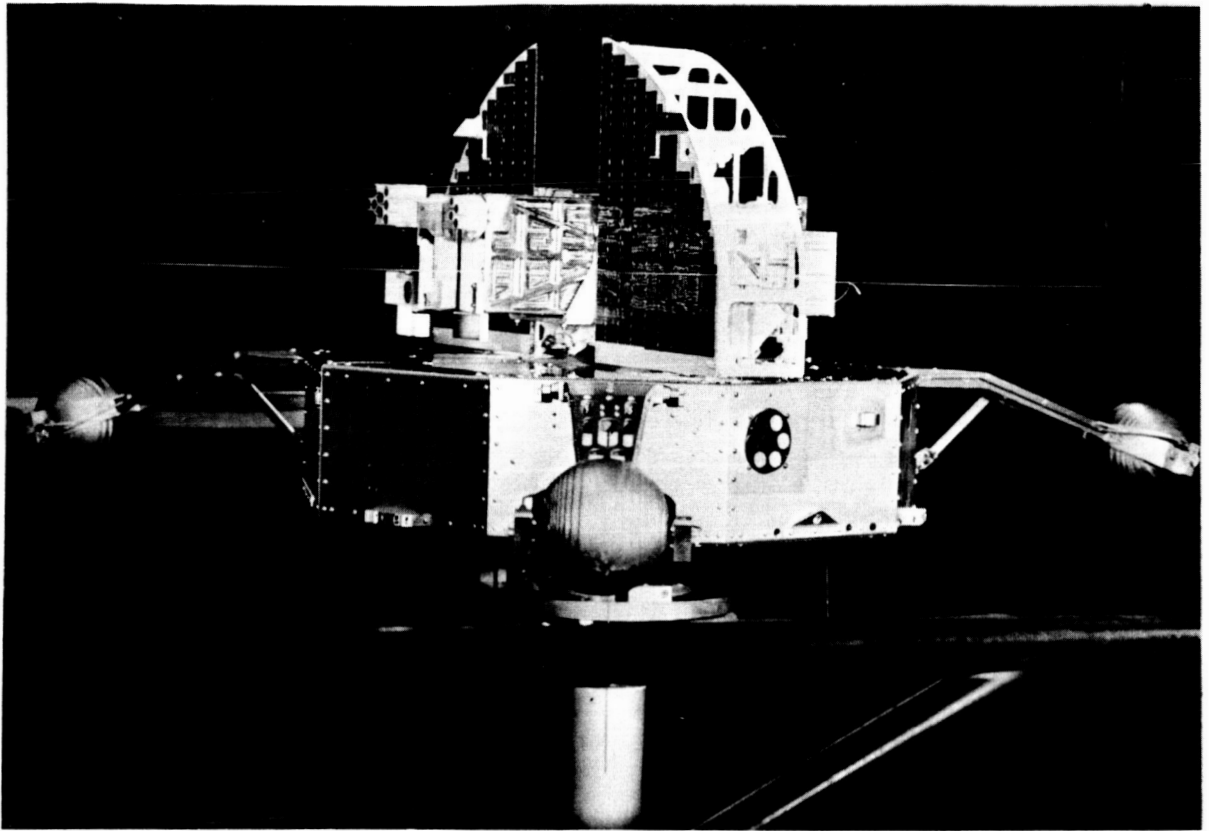


FIGURE 4-1.—Orbiting Solar Observatory (OSO I).

the oriented upper section fixed in space by driving against the rotating wheel. The elevation servo had only to rotate the instruments

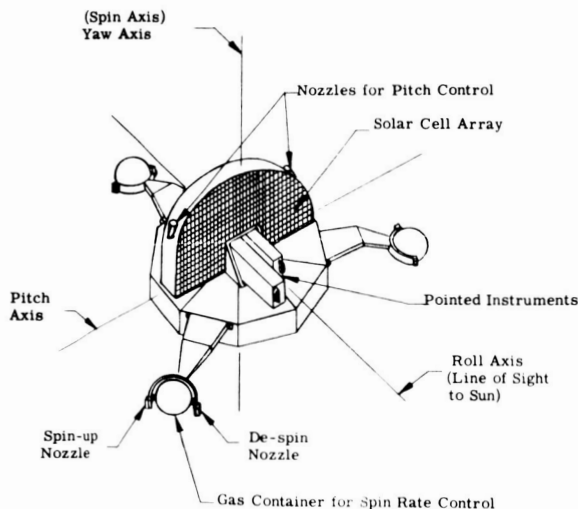


FIGURE 4-2.—Orbiting Solar Observatory systems.

a few degrees in a plane containing the solar vector and the spin axis. The azimuth fine servo and the elevation servo were almost identical.

An azimuth coarse servo was required about the azimuth shaft since the upper section spun up at night. This meant that the azimuth servo had to be able to acquire the sun from a spinning start. The coarse mode servo was required to give proper error signals from all possible sun angles about the azimuth axis and to bring the upper section into the operating region of the fine detectors.

The azimuth coarse servo was designed to acquire the sun from a spinning start with the spacecraft spin axis ± 15 degrees from the normal to the solar vector. This value was selected because of the ± 10 -degree tolerance assigned to spin-axis orientation at orbit injection.

OSO I FLIGHT SEQUENCE

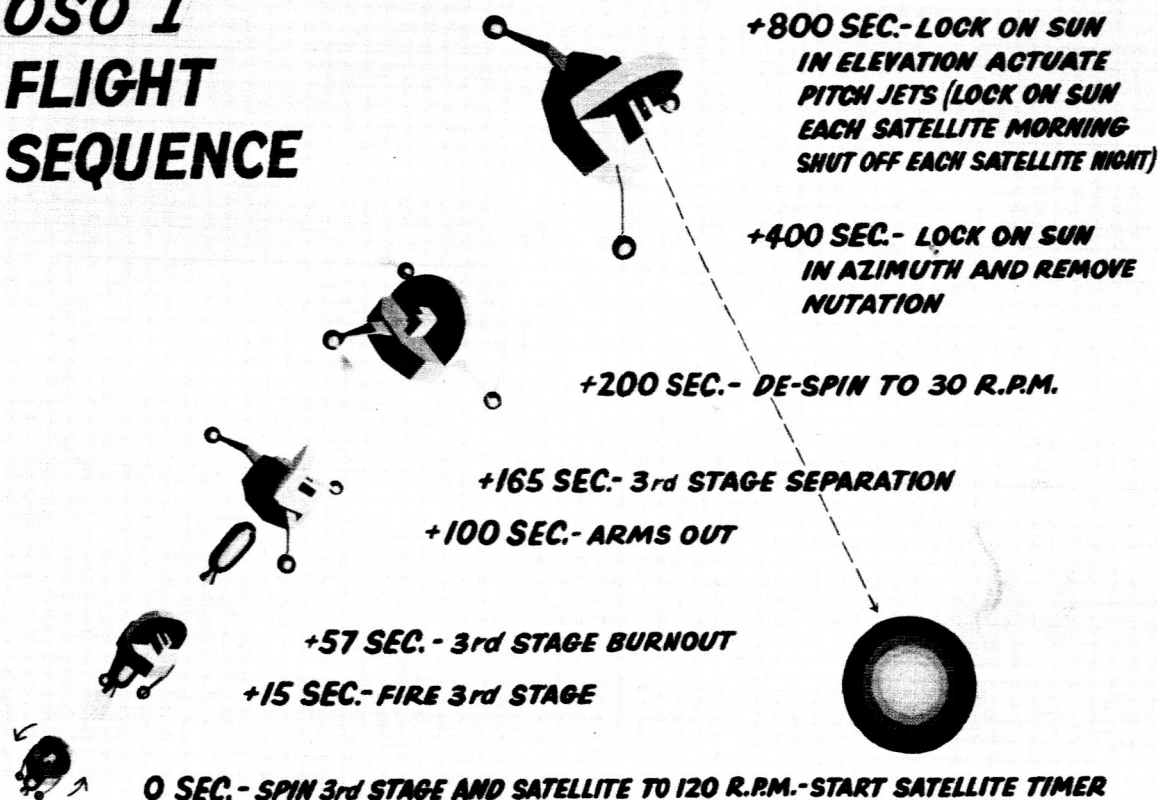


FIGURE 4-3.—OSO I flight sequence.

The coarse servo eyes, therefore, had to have a 360-degree field of view about the spin axis and a ± 15 -degree field of view perpendicular to the spin axis. The field of view of the unshielded coarse eye was ± 90 degrees in the latter direction. Shields were installed to permit an unobstructed view of the sun at an angle of ± 15 degrees and to occult the sun at angles greater than ± 25 degrees. This arrangement permits the coarse eyes to see the sun throughout the design range and limits their view of the earth to the maximum possible extent.

The object of this, of course, was to make the sun the brightest object to the coarse eyes so that the sun would be acquired and not the earth. Even though this problem arose only

during the first acquisition immediately following third-stage separation, it was still serious. If the earth were acquired, the pitch control system would precess the spacecraft spin axis to a position perpendicular to the earth's local vertical. The spacecraft spin axis would be precessed so far out of perpendicularity to the solar vector that acquisition of the sun at the following dawn would be highly unlikely.

The azimuth fine eyes had a field of view limited to ± 10 degrees. Beyond this range, the inputs to the azimuth system consisted entirely of the signals from the coarse eyes.

At the start of acquisition both the fine and coarse servos were in operation. The coarse servo despun the upper structure and oriented it at the sun to within ± 3 degrees (within the

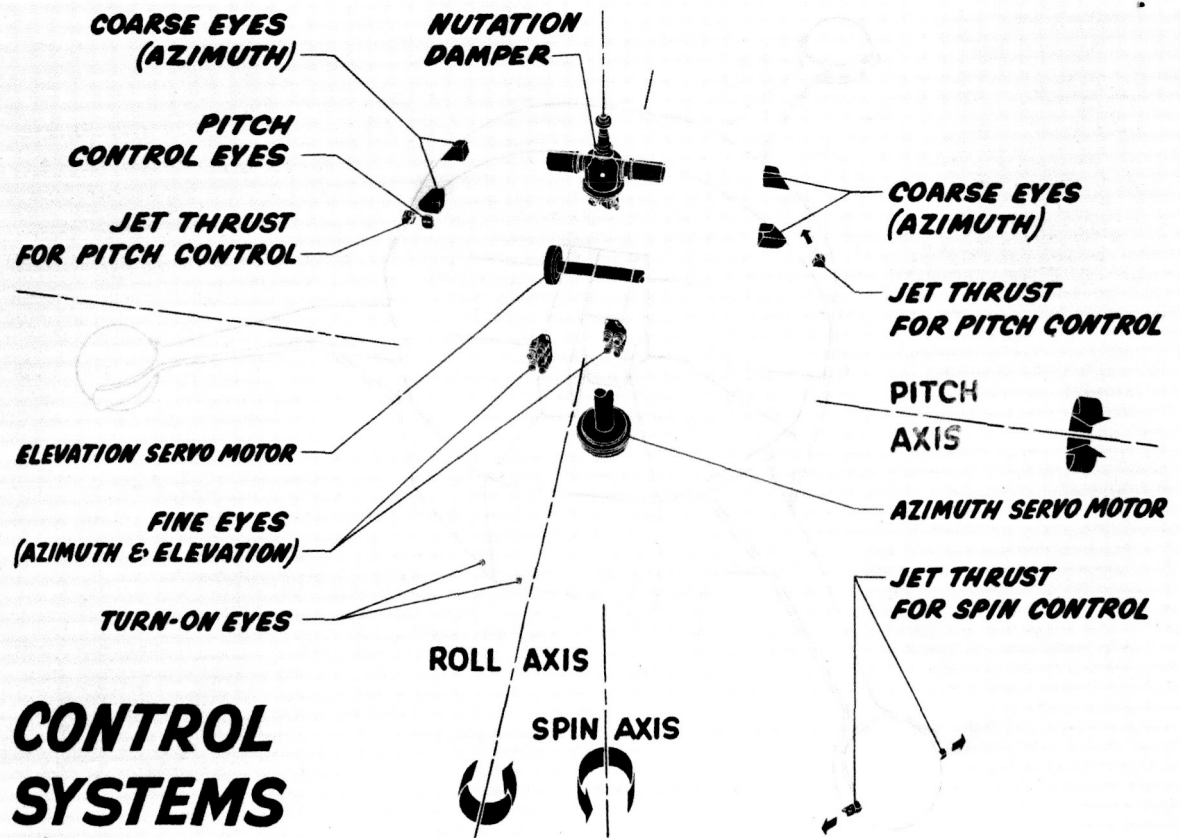


FIGURE 4-4.—OSO I control systems.

fine eye field of view). If the upper structure maintained this orientation for a period of 3 seconds, the input signal from the coarse eyes was disconnected. This action was accomplished by a time delay circuit activated by a target eye having a ± 6 - to 8-degree field of view in azimuth. The fine azimuth servo then pointed the experiment package to within ± 10 degrees of the sun.

Disconnection of the coarse eyes eliminated pointing errors which might have resulted from coarse eye unbalance due to stray light from the earth or reflections from the spacecraft.

The elevation servo was required to work over a maximum range of ± 5 degrees because of the active control of the spin axis orientation. The fine elevation detectors have a field of view of ± 10 degrees which eliminated the requirement for an elevation coarse control mode.

The mechanical constants of primary importance in the servo design were friction, moment of inertia, and the motor damping.

The moment of inertia of the upper structure about the spin axis of the spacecraft included the moment of inertia of the experiment package which could rotate about a pitch axis perpendicular to the spin axis. For a rotation of γ about this pitch axis, the contribution of the experiment package to the moment of inertia about the spin axis was reduced by the factor $\cos^2 \gamma$. Since the pitch rotation was limited to ± 5 degrees and the square of the cosine of 5 degrees is 0.993, the moment of inertia about the spin axis was constant to within 1 percent.

The friction present was the coulomb friction due to bearings, slip ring, and motor brushes. This friction was relatively constant in magnitude, and its sign opposite to that of the angular

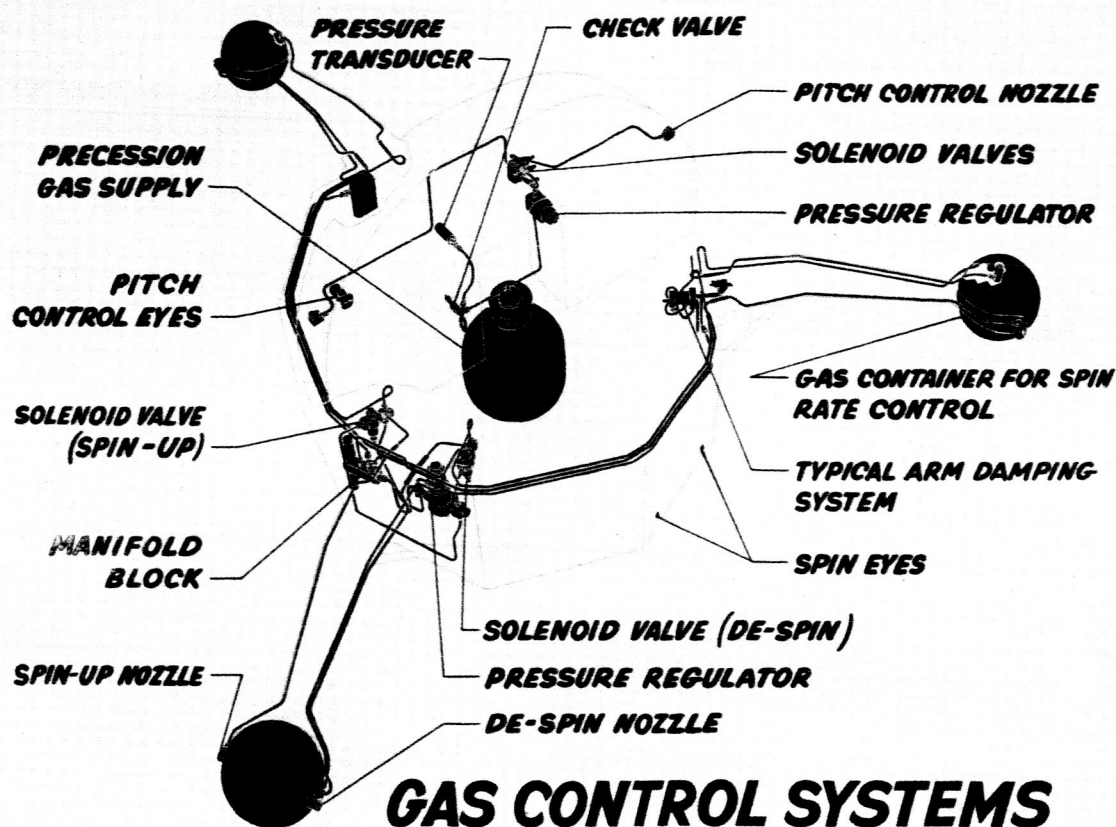


FIGURE 4-5.—Gas control systems.

velocity. The value of the coulomb friction for the azimuth system was approximately 0.05 lb-ft.

A torque motor was selected as the actuator because it could be connected directly to the azimuth shaft, thus eliminating the need for gearing with the attendant problems of backlash and increased friction. The torque motor also had a high efficiency.

A 1.8-lb-ft (364-oz-in.) motor was selected. The motor, with the spacecraft voltage available, produced a maximum of about 0.54 lb-ft of torque. It provided a *dynamic* range (i.e., the ratio of full output torque to friction torque) of 10 and was a convenient size mechanically. The time to acquire the sun with this motor

operating with a full output despin torque and a duty factor of 0.6 was

$$t = \frac{I_s}{T} \dot{\theta}_0$$

$$= \left[\frac{3 \text{ lb-ft-sec}^2}{(0.54 \times 0.6) \text{ lb-ft}} \right] \left[\pi \frac{\text{rad}}{\text{sec}} \right]$$

$$\cong 30 \text{ sec}$$

In writing the overall transfer function, it is assumed that the eyes and the amplifiers were not frequently sensitive in the frequency range of interest. The frequency range of interest was below 100 cps, and the frequency response of the solar-cell eyes and amplifiers was flat

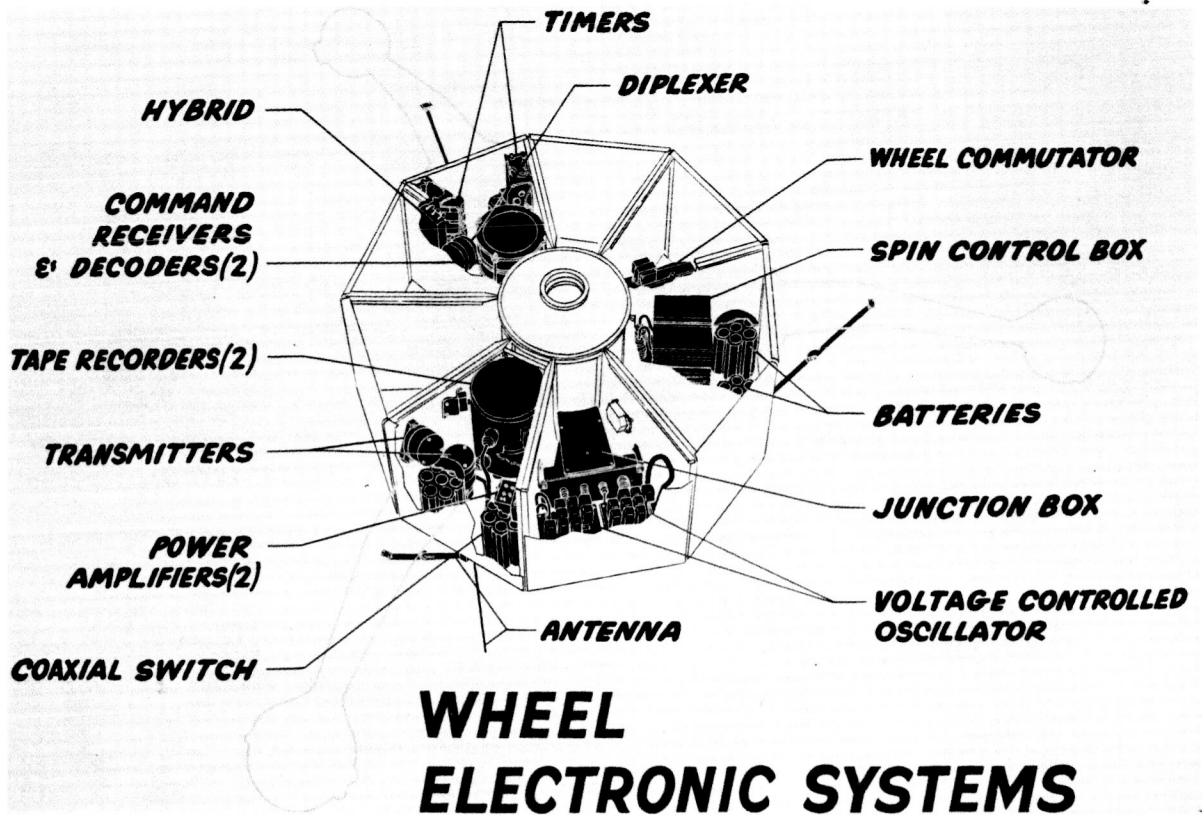


FIGURE 4-6.—Wheel electronics systems.

for frequencies several orders of magnitude greater than 100 cps.

To obtain the open-loop transfer function, we note that the equation of motion for a body rotating about an axis with coulomb friction opposing the velocity of the body is given by

$$I_s \frac{d^2}{dt^2} (\phi_0) = -K_d \frac{d}{dt} (\phi_0) \quad (4.1)$$

where

I_s = moment of inertia of the body about the axis of revolution

ϕ_0 = angular position of the body

K_d = motor damping constant

In an effort to control the motion of the body, a control torque, $T_c(t)$, and gain, K , are introduced,

where

$$T_c(t) = \int_{0-t}^t E_i(t-\tau) W(\tau) d\tau \quad (4.2)$$

and

$E_i(t)$ = the input to a filter

$W(t)$ = the weighting function of the filter

With the addition of a control torque, Equation (4.1) becomes

$$\left(I_s \frac{d^2}{dt^2} + K_d \frac{d}{dt} \right) \phi_0 = K \int_{0-t}^t E_i(t-\tau) W(\tau) d\tau \quad (4.3)$$

The Laplace transform of Equation (4.3) is

$$\begin{aligned} s(I_s s + K_d) \phi_0 &= K \sum [W(t)] \sum [E_i(t)] \\ &= c(s) \phi_{in}(s) \end{aligned} \quad (4.4)$$

where $c(s)$ is the transfer function of the compensating networks needed to stabilize the system and $\phi_{in}(s)$ is the Laplace transform of the input to the compensating network.

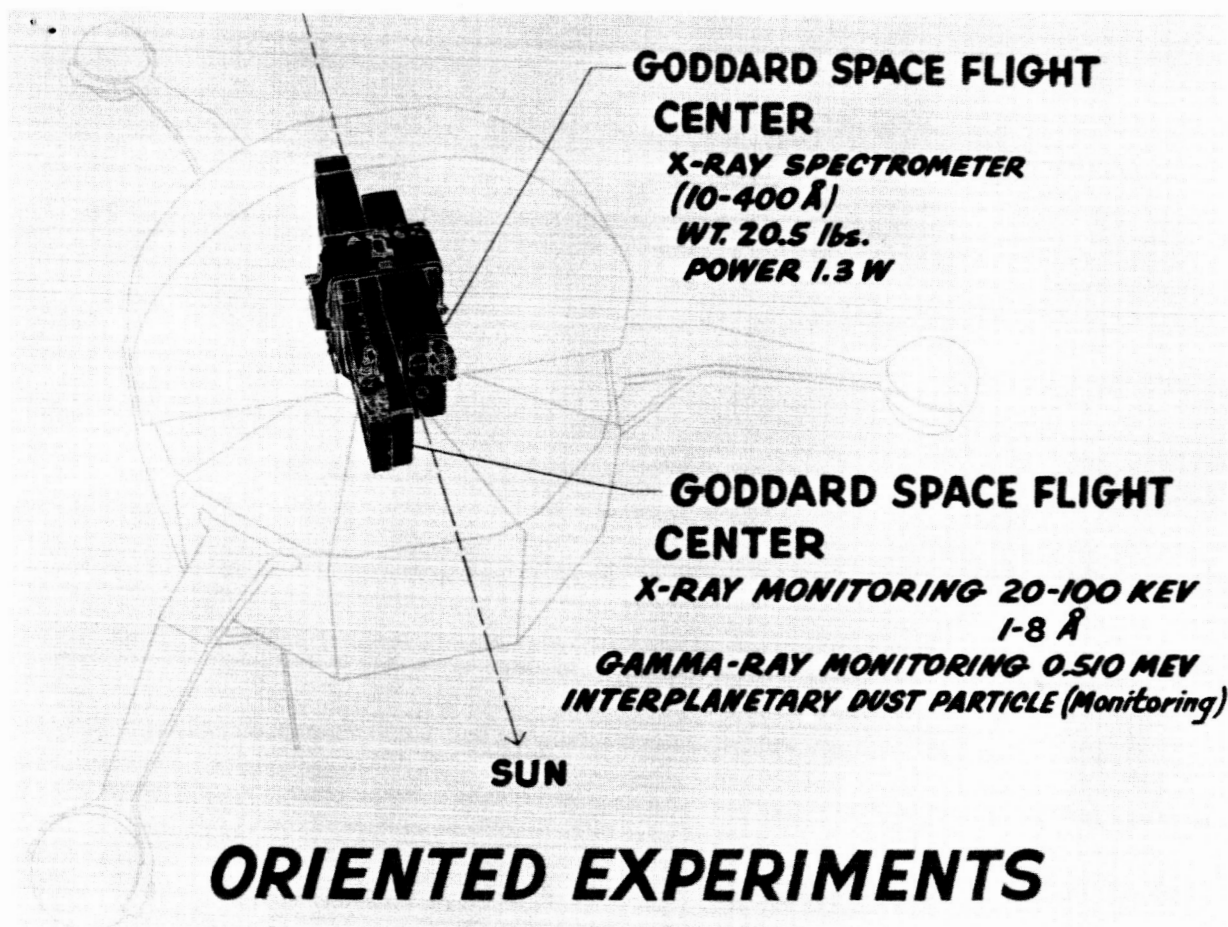


FIGURE 4-7.—Solar-oriented experiments.

The open-loop transfer function becomes

$$\frac{\phi_0(s)}{\phi_{in}(s)} = \frac{K}{I_s} \frac{1}{s \left(s + \frac{K_d}{I_s} \right)} \cdot c(s) \quad (4.5)$$

The fine and coarse azimuth servos were to have a pointing accuracy of 1 minute and 2 degrees, respectively. To make this possible, an angular deviation of 1 minute and 2 degrees had to produce a torque at the output sufficient to overcome the coulomb and viscous friction.

Thus, we have

$$K_F = \left(\frac{0.05 \text{ lb-ft} + 0.015 \text{ lb-ft}}{2 \text{ min}} \right) \left(\frac{60 \text{ min}}{\text{deg}} \right) \left(\frac{180 \text{ deg}}{\pi \text{ rad}} \right) = 111.73 \text{ lb-ft/rad}$$

and

$$K_c = \left(\frac{0.05 \text{ lb-ft} + 0.015 \text{ lb-ft}}{(2 \text{ deg})(0.966)} \right) \left(\frac{180 \text{ deg}}{\pi \text{ rad}} \right) = 1.93 \text{ lb-ft/rad}$$

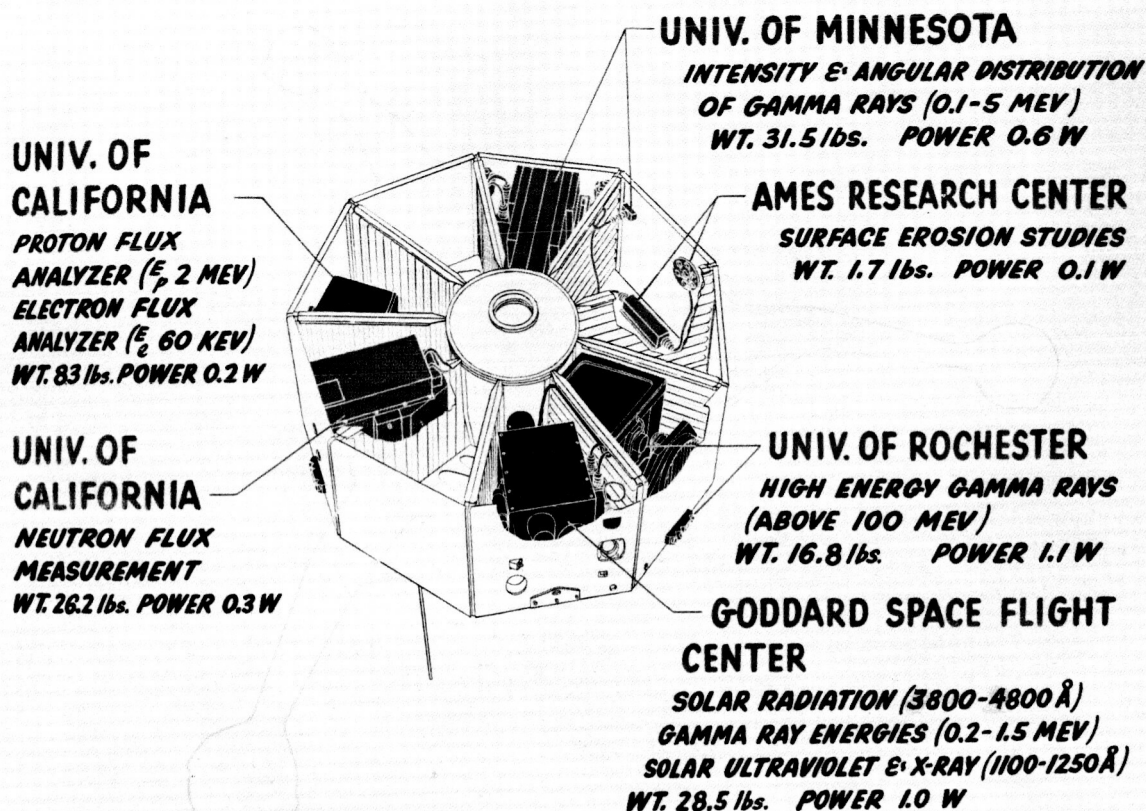
for the minimum fine and coarse azimuth gains, respectively. The factor 0.966 in the denominator of the latter equation was introduced to account for the reduced eye gain at ± 15 degrees away from the sun.

Azimuth Servo

The simplified equations of motion of the wheel and upper structure about the azimuth shaft are

$$I p^2 \phi = -T \quad (4.6)$$

$$I_\omega p^2 \phi_\omega = T \quad (4.7)$$



WHEEL EXPERIMENTS

FIGURE 4-8.—Wheel experiments.

where

$$p = \frac{d}{dt}$$

I = the moment of inertia of the upper structure

I_ω = the moment of inertia of the wheel

ϕ = the azimuth angle of the upper structure

ϕ_ω = the azimuth angle of the wheel

T = the torque applied between the two sections

Inclusion of the torque motor on the azimuth shaft results in the equations

$$I_p \ddot{\phi}(t) = -K_d p[\dot{\phi}(t) - \dot{\phi}_\omega(t)] - T_F - c \quad (4.8)$$

$$I_\omega p^2 \phi_\omega(t) = K_d p[\dot{\phi}(t) - \dot{\phi}_\omega(t)] + T_F + c \quad (4.9)$$

where K_d is the torque motor damping constant; T_F is the torque due to the friction of the bearings, torque motor brushes, the slip rings; and c is the control torque. The control torque, c , is given by the expression

$$c = K \left(\frac{\tau_1 p + 1}{\tau_2 p + 1} \right) \phi(t) \quad (4.10)$$

where K is the torque gain and $\tau_1 p + 1 / \tau_2 p + 1$ is the equalization function. Substitution of Equation (4.10) into Equation (4.8) and (4.9) yields

$$I_p \ddot{\phi}(t) = -K \frac{\tau_1 p + 1}{\tau_2 p + 1} \phi(t) - K_d p[\dot{\phi}(t) - \dot{\phi}_\omega(t)] - T_F \quad (4.11)$$

$$I_\omega p^2 \phi_\omega(t) = K \frac{\tau_1 p + 1}{\tau_2 p + 1} \phi(t) + K_d p[\dot{\phi}(t) - \dot{\phi}_\omega(t)] + T_F \quad (4.12)$$

$$\left[\frac{I}{K} s^2 (\tau_2 s + 1) + \frac{K_d}{K} \left(1 + \frac{I}{I_\omega} \right) s (\tau_2 s + 1) + (\tau_1 s + 1) \right] \phi(s) = - \left(\frac{\tau_2 s + 1}{K} \right) \frac{T_F}{s} + \left[\frac{I}{K} s (\tau_2 s + 1) + \frac{K_d}{K} \left(1 + \frac{I}{I_\omega} \right) (\tau_2 s + 1) + \tau_1 \right] \phi(0) + (\tau_2 s + 1) \left[\frac{I}{K} s + \frac{K_d}{K} \left(\frac{I}{I_\omega} + 1 \right) \right] \frac{\dot{\phi}(0)}{s} + \frac{I}{K} \tau_2 \ddot{\phi}(0) \quad (4.13)$$

Equation (4.13) shows that the upper section is stabilized with respect to inertial space. The errors in the servo system were due to the frictional torque and the initial velocity of the upper section at each spacecraft dawn.

By use of the final value theorem, it is seen that the servo error due to the friction torque, T_F , is given by the expression

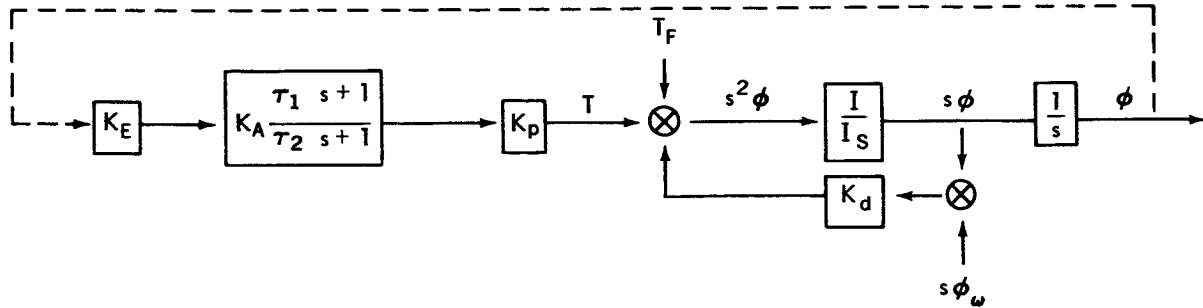
$$\epsilon_{T_F} = \frac{T_F}{K}$$

and the error due to initial velocity (or motor damping) is given by

$$\epsilon_{\dot{\phi}(0)} = \left(1 + \frac{I}{I_\omega} \right) \frac{K_d}{K} \phi(0) \quad (4.14)$$

The value of K for the azimuth servo (fine mode) is 250 lb-ft/rad; the friction torque of the bearings, slip rings, and motor brushes is approximately 0.05 lb-ft; and the motor damping constant, K_d , is 0.038 lb-ft/sec. The moments of inertia I and I_ω are 2 and 24 slugs, respectively.

Therefore, the errors due to the frictional torques and motor damping are 0.60 and 1.55 minutes of arc. These errors are, however, of a particular sign. During control system testing, these errors were biased out so that the actual pointing errors due to friction and motor damping were caused only by deviations from the biased value. The following is a block diagram of the azimuth servo.



In the block diagram the following notation is used:

K_E = eye gain ($\mu a/\text{rad}$)

K_A = DC amplifier gain (volts/ μa)

K_P = power amplifier and torque motor gain (lb-ft/volt)

$K = K_E K_A K_P = 250$ lb-ft-rad

T_F = friction torque = 0.05 lb-ft

K_d = motor damping = 0.038 lb-ft-sec

I = moment of inertia of upper section about the spin axis

$I_\omega = 24$ lb-ft-sec²

τ_1 = lead time constant (sec)

τ_2 = lag time constant (sec)

The open-loop transfer function for the block diagram can be written as

$$G(s) = \frac{K(\tau_1 s + 1)}{K_d I s (\tau_2 s + 1) \left(\frac{s}{K_d} + 1 \right)} \quad (4.15)$$

The lead-lag equalization network readily stabilizes this system; however, it was found in development of the control system that structural resonances significantly altered the control open-loop transfer characteristics. The dominant resonance at 29.5 cps had amplification of approximately 40 db, which was sufficiently high to cause instability when the

requisite gain was used. To cure the problem, a high-frequency roll-off filter was introduced

in the control error channel. The resulting controller transfer function, $G_c(s)$, was

$$G_c(s) = \frac{K(0.225s + 1)}{(0.0115s + 1)[1.35(10^{-7})s^3 + 6(10^{-5})s^2 + 2.18(10^{-2})s + 1]} \quad (4.16)$$

The Bode plot of this transfer function and the measured frequency response of the flight hardware are shown in Figure 4-9.

The Bode plots of the total loop transfer function, including the controller characteristics and the structural resonances, are shown in Figure 4-10. To obtain this plot, a linear power amplifier was used instead of the pulsed power amplifier.

The gain margin of the linear system is shown to be 7 db. The gain margin of the actual system correlated closely with the linearized model. The fast roll-off of the con-

troller filter succeeded in attenuating sufficiently the resonant peaks.

The azimuth coarse mode used the same roll-off network that was used for fine. The lead-lag equalization and the system gain were different, however. The lead-lag network can be represented as

$$G(s) = \frac{3.2s + 1}{0.2s + 1}$$

The Bode plot of the actual flight hardware equalization, including the roll-off network, is shown in Figure 4-11.

The torque gain for the coarse mode was nominally 50 lb-ft/rad.

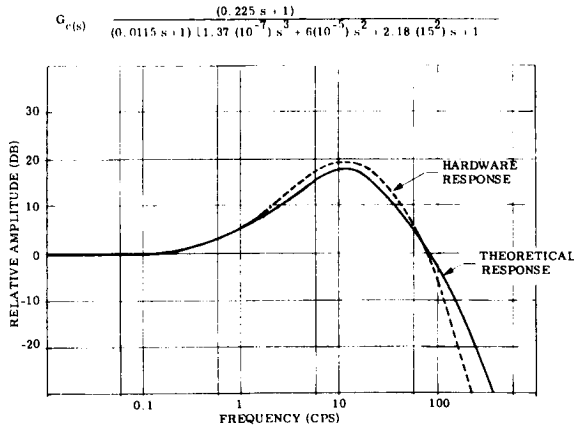


FIGURE 4-9.—Azimuth fine mode equalization.

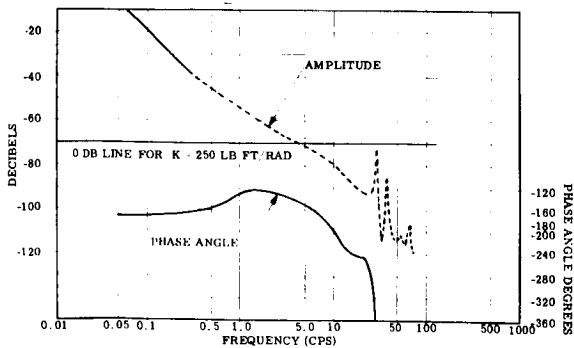


FIGURE 4-10.—Linearized azimuth servo bode plot.

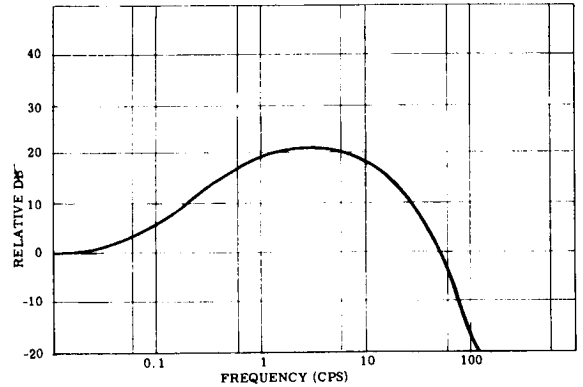


FIGURE 4-11.—Azimuth coarse mode equalization.

Elevation Servo

Simplification of the equations of motion as given in Chapter 2 (see Equation (2.59)) results in the equations

$$I p^2 \alpha = K \frac{\tau_1 p + 1}{\tau_2 p + 1} (e - \alpha) + K_d p (e - \alpha) + T_F \quad (4.17)$$

$$I_{x_3} p^2 (e - \alpha) = -K \frac{\tau_1 p + 1}{\tau_2 p + 1} (e - \alpha) - K_d p (e - \alpha) - T_F \quad (4.18)$$

The solution of the above equations is given by

$$[I_{x_3}s^2(\tau_2s+1)+K_d s(\tau_2s+1)+K(\tau_1s+1)][e(s)-\alpha(s)]=-\frac{T_F}{s}(\tau_2s+1)+[I_{x_3}s(\tau_2s+1)+K_d(\tau_2s+1)+K\tau_1][e(0)-\alpha(0)]+[I_{x_3}(\tau_2s+1)+K_d\tau_2][\dot{e}(0)-\dot{\alpha}(0)]+I_{x_3}\tau_2[\ddot{e}(0)-\ddot{\alpha}(0)] \quad (4.19)$$

At each satellite dawn, $\dot{e}(0)=\dot{\alpha}(0)=\ddot{e}(0)=\ddot{\alpha}(0)=0$, and the servo error is given by

$$\epsilon_{e-\alpha}=-\frac{T_F}{K}$$

The servo error for a T_F of 0.05 lb-ft and a gain, K , of 150 lb-ft/radian is approximately 1 minute of arc and, unlike the azimuth servo, cannot be biased out.

The elevation servo block diagram is identical to that of the azimuth servo except that the motor damping input from the wheel does not exist.

The parameters used on the elevation servo are as follows:

$$K=150 \text{ lb-ft/radian}$$

$$K_d=0.015 \text{ lb-ft-sec}$$

$$I_{x_3}=1.3 \text{ lb-ft-sec}^2$$

$$\tau_1=\text{lead time constant}=0.22 \text{ sec}$$

$$\tau_2=\text{lag time constant}=0.016 \text{ sec}$$

A Bode plot of the elevation loop using these parameters and a 20-db structural resonance is shown in Figure 4-12. The phase margin with nominal gain is 58 degrees.

Mechanization of the Electrical Control System

The servo control for orienting the instruments to the sun operated about two orthogonal

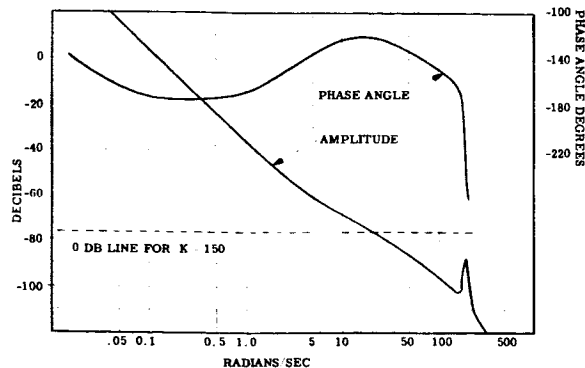


FIGURE 4-12.—Linearized elevation servo bode plot.

axes, the azimuth axis and the elevation axis. The two servo systems were independent of one another, but were nearly identical. The azimuth servo system was slightly more complicated because of the large acquisition angles required. It operated about the azimuth of spin axis of the spacecraft and was used to hold the oriented section fixed in space by driving against the rotating wheel. The elevation servo rotated the instruments a few degrees in a plane containing the solar vector and the spin axis.

A block diagram of the azimuth servo system is shown in Figure 4-13. Error signals pro-

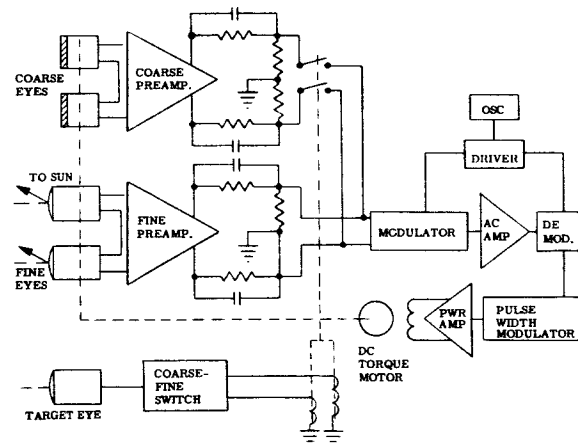


FIGURE 4-13.—Azimuth servo block diagram.

duced by the coarse and fine optical eyes, i.e., error detectors, were sent to the coarse and fine preamplifiers and then through appropriate lead-lag stabilization networks. If the error angle was large, the coarse eyes with their 360-degree field of view were connected into the circuit along with the fine eyes. When the error angle was less than 5 degrees, the coarse eyes were then switched out of the circuit and only the fine eyes exercised control.

The error signal was chopped by a solid-state chopper at 2 kc, amplified at that frequency and

demodulated by a solid-state demodulator. The amplifier then fed the pulse-width modulator, which converted the dc error signal to a pulse train in which the pulse width was proportional to the amplitude of the error signal. The repetition rate of the pulse-width modulator was about 55 pulses per second.

The pulse-width signal drove a power amplifier and transistor bridge which, in turn, drove the motor. The transistors in the bridge were either saturated or cut off by the pulse-width signal so that very little power was dissipated in the bridge.

The motive element was a dc torque motor using a permanent magnet field. This type of motor delivered large torques directly to the output shaft. The elements mentioned briefly above will now be discussed in more detail.

Electrical Control Hardware Description

The following discussion presents the theory and method of operation of each of the functional elements used in mechanizing the eleva-

tion and azimuth position controls. Support elements such as the readout amplifiers are also discussed. The packaging techniques used on the electronic circuits of the control system are illustrated in Figures 4-14 through 4-17.

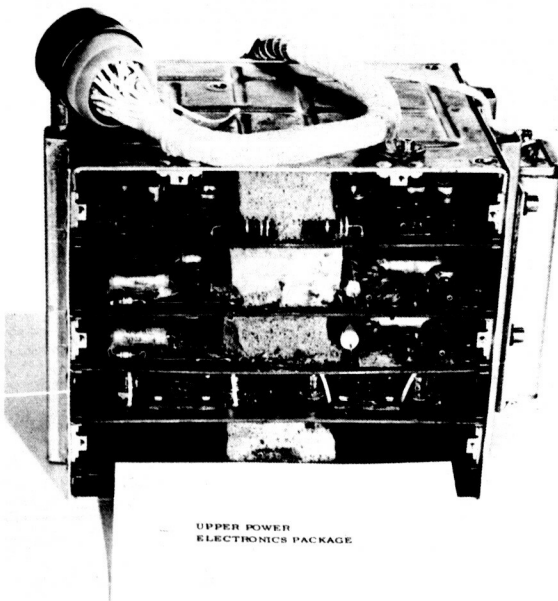


FIGURE 4-14.—Servo drive electronics.

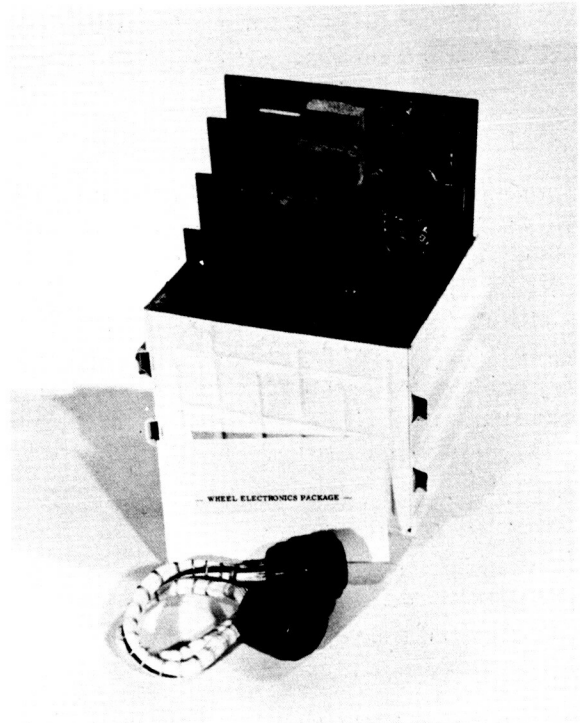


FIGURE 4-15.—Wheel electronics package.

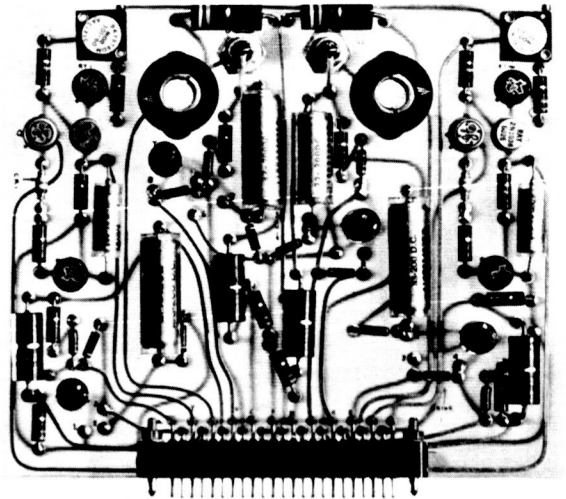


FIGURE 4-16.—Servo power amplifier.

Error Detectors

Three types of eyes were used in the OSO servo system: coarse eyes, disabling eyes, and fine eyes. Coarse eyes were used for coarse pointing and acquisition; disabling eyes were used to switch the azimuth coarse servo off; and fine eyes were used for accurate pointing.

Coarse Eyes—The coarse eyes each had a characteristic curve approximating a cosine curve. In OSO I, the four coarse eyes were arranged as shown in Figure 4-18, using shadowing masks to modify the cosine curve into that shown in Figure 4-19. This arrangement provided an error signal with a stable null toward the solar vector and an unstable null in the antisolar direction. In addition, the arrangement was specifically designed to aid in solving the acquisition problem. At spacecraft dawn, the oriented section (sail) was spinning at the

same angular velocity as the wheel section. Because of this, the sun had to be acquired each spacecraft morning by stopping the sail. Since the motor torque was not sufficient to stop the sail in a half revolution, a signal was needed that would provide a stopping drive for a large portion of a rotation. A rate gyro would, of course, have been an ideal type of device for generating such a signal, but the power consumed and the reliability problem with such a device would not allow its use. By shaping the coarse eye output curve as shown in Figure 4-19, the error rate damping (lead network) generated signals which provided sail despin torques over an appreciable percentage of a revolution.

The azimuth coarse servo was able to orient the upper section to the sun to an accuracy of a few degrees. When this was done the upper section was stopped and coarsely oriented. The fine eyes, which had a 10-degree field of view, were then able to see the sun and to orient the instruments more precisely.

Disabling Eye—When the servo had settled down in the fine-eye region, the coarse eyes were no longer necessary and, if left in the system, they would have contributed errors in the pointing. The errors were caused by signals due to the earth's albedo and to the difference in the coarse and fine eye null positions. A disabling eye actuated the switch disconnecting the coarse eyes from the circuit when no longer needed. The disabling eye was identical to a fine eye in construction except that the reticle was a rectangular aperture. The angular characteristics of a disabling eye are shown in Figure 4-20. A time delay of 2 to 3 seconds

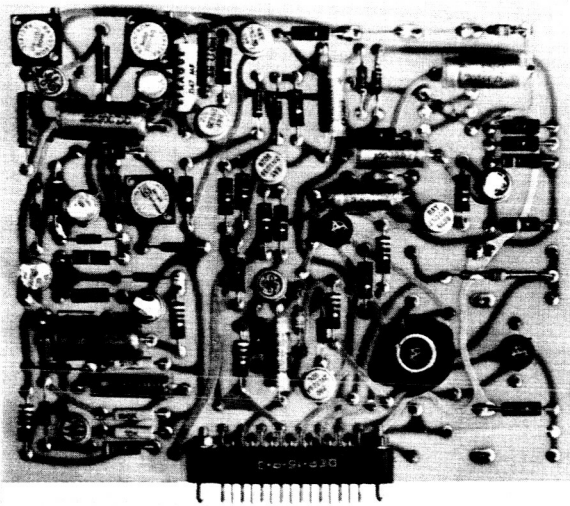


FIGURE 4-17.—Pulse width modulation circuit.

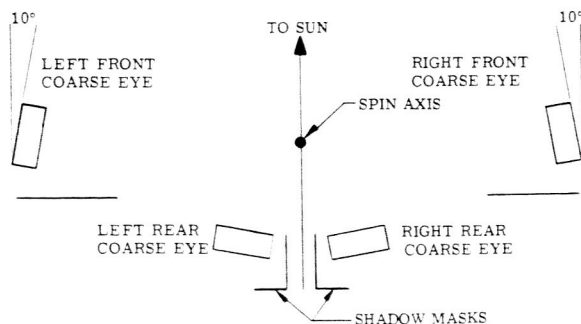


FIGURE 4-18.—Arrangement of the OSO I coarse eyes.

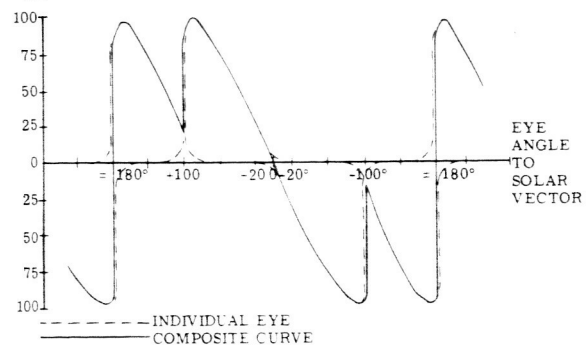


FIGURE 4-19.—Angular characteristics of the coarse eye.

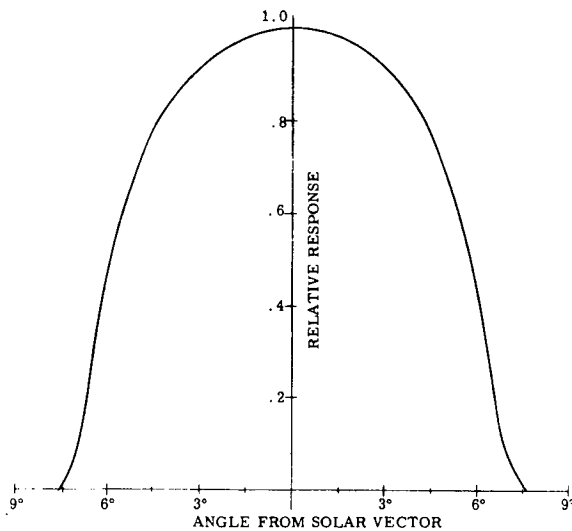


FIGURE 4-20.—Angular characteristics of disabling eye.

was provided with the switch so that switching would not continually take place during the initial acquisition.

Fine Eyes—The fine eyes for both elevation and azimuth were mounted directly on the pointed instruments. Fine-eye detectors provided the error signals for the final servo control of the instruments pointed at the sun. Two eyes differentially connected were required to exercise control about a single axis.

Figure 4-21(a) shows the angular characteristics of the fine eye. The combined field of view of an eye pair was a cone with a half-angle of nearly 10 degrees. If the conical field were cut by a plane passing through the axis of symmetry, the resulting half-cones would correspond to the field of view of the individual eyes. In practice, the half-cones were slightly lapped, or toed-in, in order to linearize the response of the pair through the zero position (see Figure 4-21(b)).

The major components of this detector were: an objective lens, reticle, filter, back aperture, and silicon-cell detector, as shown in Figure 4-22. The objective lens was a plano-convex lens with a back focal distance of 0.320 inch. Its purpose was to project the image of the solar disc on the surface of the reticle.

The reticle was vacuum deposited on the front surface of a deep-red glass filter disc and

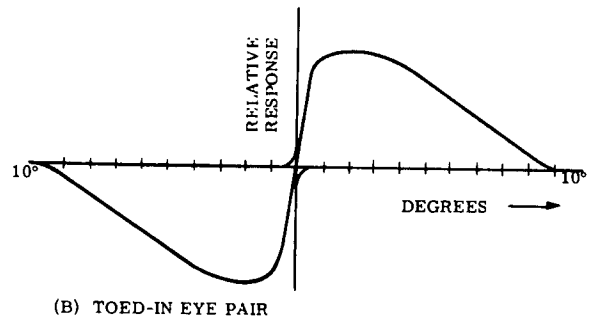
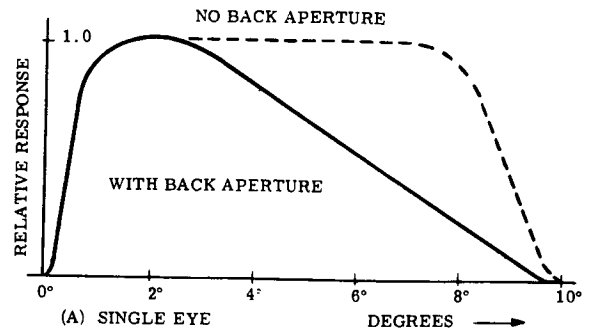


FIGURE 4-21.—Fine eye response.

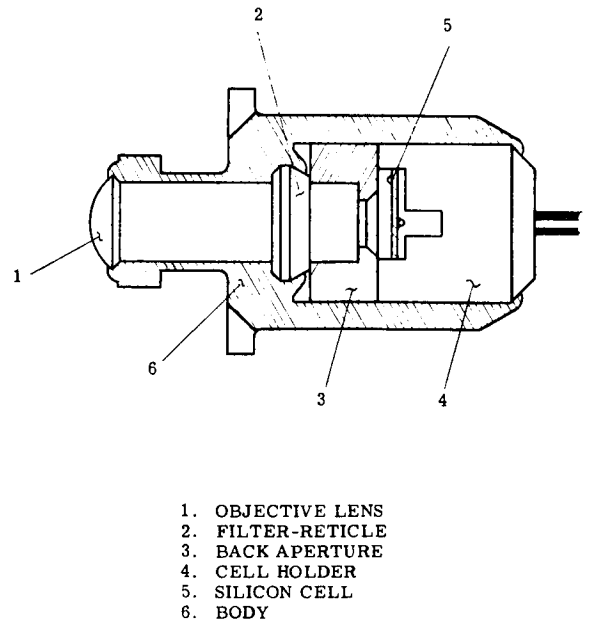


FIGURE 4-22.—Fine eye assembly.

was in the form of a knife edge, the region to one side of the edge being opaque (see Figure 4-23).

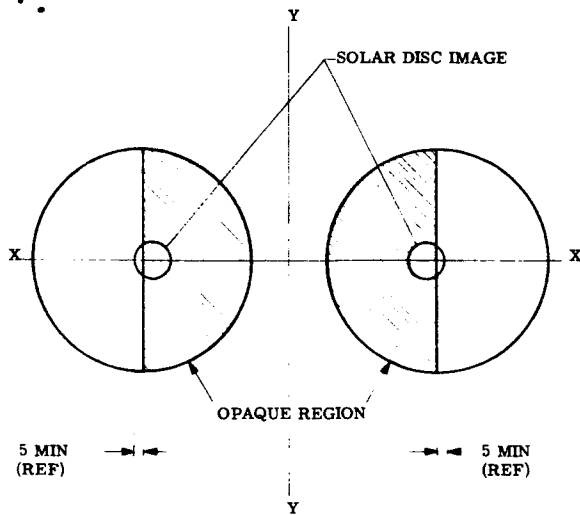


FIGURE 4-23.—Fine eye pair image position "on target" condition.

The deep-red filter was used so that the pass band accepted by the detector would fall in a natural window in the atmosphere. The cutoff point of this filter on the low side was approximately 0.60 micron, below which there was severe atmospheric attenuation. The filter therefore permitted ground testing of servo systems to establish loop gains which changed less than 20 percent for operation outside of the atmosphere.

The back aperture controlled the output from errors between 1 degree and 10 degrees in such a way that overshooting was minimized for large initial position errors.

The lens and reticle were staked into the same piece of metal so that no possible shifting could occur as a result of vibration. The cylindrical tube mount of the objective lens was designed so that it could be deformed by a special adjusting tool to achieve a boresight adjustment of the detector. In these eyes, 0.0001-inch movement of the lens corresponded to a minute of angle, thus the importance of a fixed alignment between objective lens and reticle is apparent.

Photo Cells—The photo cells used in the coarse and fine eyes were small silicon solar cells. The silicon solar cell has excellent characteristics for use in such an application. The time constant is small (microsecond); the

output current is linear with light level and the photosensitivity is very stable.

To have the current output linear with light level, and to minimize changes with temperature, it was necessary to use the cell as a short-circuit current generator. For this size cell and expected cell illumination levels, the short circuit current characteristics would be obtained if the load is less than about 100 ohms. The preamplifier following the eyes was a grounded base amplifier which had an input impedance of about 80 ohms.

The type of photo cell used in the disabling eye was a Clairex CL-7 cadmium sulfide cell, a photo-resistive device.

Drive Motors

Both servos used dc torque motors as driving elements. These large-diameter pancake motors delivered large torques directly to the output shaft without the use of gears. The azimuth motor provided a peak torque of 0.83 lb-ft, and the elevation motor, 0.47 lb-ft. Since the motors and drive system were directly exposed to vacuum, the elimination of gears was very important. A serious problem with this type of motor was in the use of brushes exposed to vacuum. However, by treating the brushes with a special lubricant it was possible to run these motors for several times the required number of revolutions in a vacuum of 2×10^{-8} mm Hg with negligible brush wear.

At the normal spin rate used for the spacecraft wheel section, the back emf of the torque motors produced no effect other than a reduction of the net drive voltage by a small amount. The back emf produced no appreciable servo damping.

Readout Amplifiers

The readout amplifiers were part of the monitoring system and provided information from which the pointing error angle of the servo was determined. These amplifiers were required to determine the azimuth and elevation pointing angles to within ± 20 seconds of arc.

The azimuth and elevation pointing angles were measured by employing two fine-eye solar detectors. A fine-eye detector consisted of two silicon solar cells mounted as shown in Figure

4-22 and connected such that a linear-output, short-circuit current was provided over the range of ± 30 minutes of arc (see Figure 4-21). The most important electrical consideration in regard to obtaining accurate readings from a fine eye was that it operated into a very low-impedance load, i.e., a load approaching a short circuit.

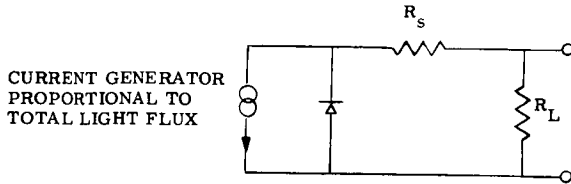


FIGURE 4-24.—Equivalent circuit of a solar cell.

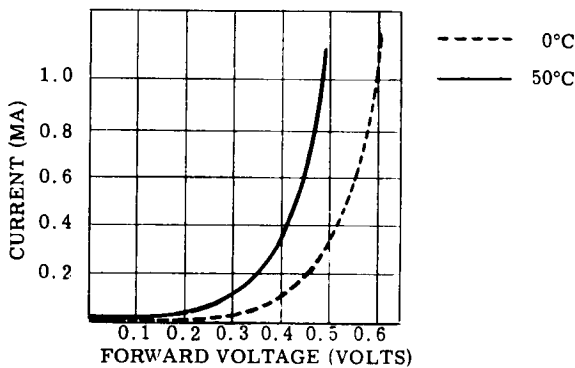


FIGURE 4-25.—Forward characteristics of a silicon diode.

Figure 4-24 shows an equivalent circuit of a single solar cell. Typical forward characteristics of the shunt diode are shown in Figure 4-25. It can be seen that a portion of the current from the current generator will be diverted by the diode depending upon the voltage developed across the diode. The accuracy requirements of the monitor were such that the diverted current had to be less than 1 percent of the generator current. This meant that for a peak current of 1.7 milliamperes, the current through the diode had to be less than 17 microamperes. By using a forward characteristic curve plotted for maximum operating temperature, the voltage required to cause 17 microamperes to flow through the diode and source resistance was determined to be 150 millivolts. Therefore,

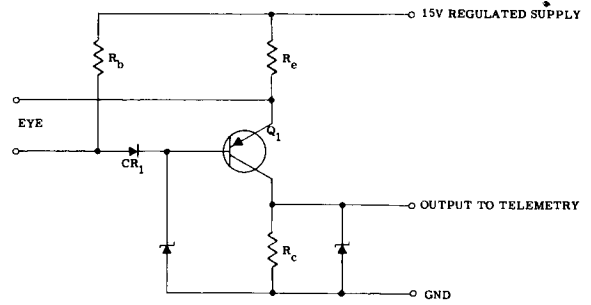


FIGURE 4-26.—Schematic of readout eye amplifier.

the amplifier impedance had to be less than 0.150 volts/ 1.7×10^{-3} amp, or 88 ohms.

A common-base configuration was chosen for the input amplifier in order to obtain the necessary low-input impedance. Figure 4-26 shows the schematic diagram. The input impedance of this circuit was 50 ohms. The circuit parameters were chosen such that the amplifier output was 2.5 volts for a zero eye current and the output ranged from zero to five volts, depending on the magnitude and polarity of the eye current. The diode, CR1, was used to nullify the transistor base-to-emitter voltage drop to keep the voltage between the input terminals less than 15 millivolts over the temperature range, 0°C to 50°C .

A zener diode was used to supply the reference voltage of the common-base circuit. This diode was selected to have a value of *temperature coefficient of voltage change* which would temperature-compensate the amplifier circuit. Using a 6.8-volt zener with a ± 5 -percent tolerance resulted in amplifier output drift of a little less than ± 0.5 millivolts per degree centigrade.

One problem with this amplifier was its output sensitivity to variations in supply voltage. This sensitivity was due to the dynamic impedance of the zener diode. Negative feedback provided by the emitter resistor solved this problem.

The input impedance of the amplifier can be derived using the equivalent circuit, Figure 4-27, where it is assumed that the biasing resistor, R_b , is large compared to the other impedance to ground from that input node, and that R_c is small compared to the r_c of the transistor.

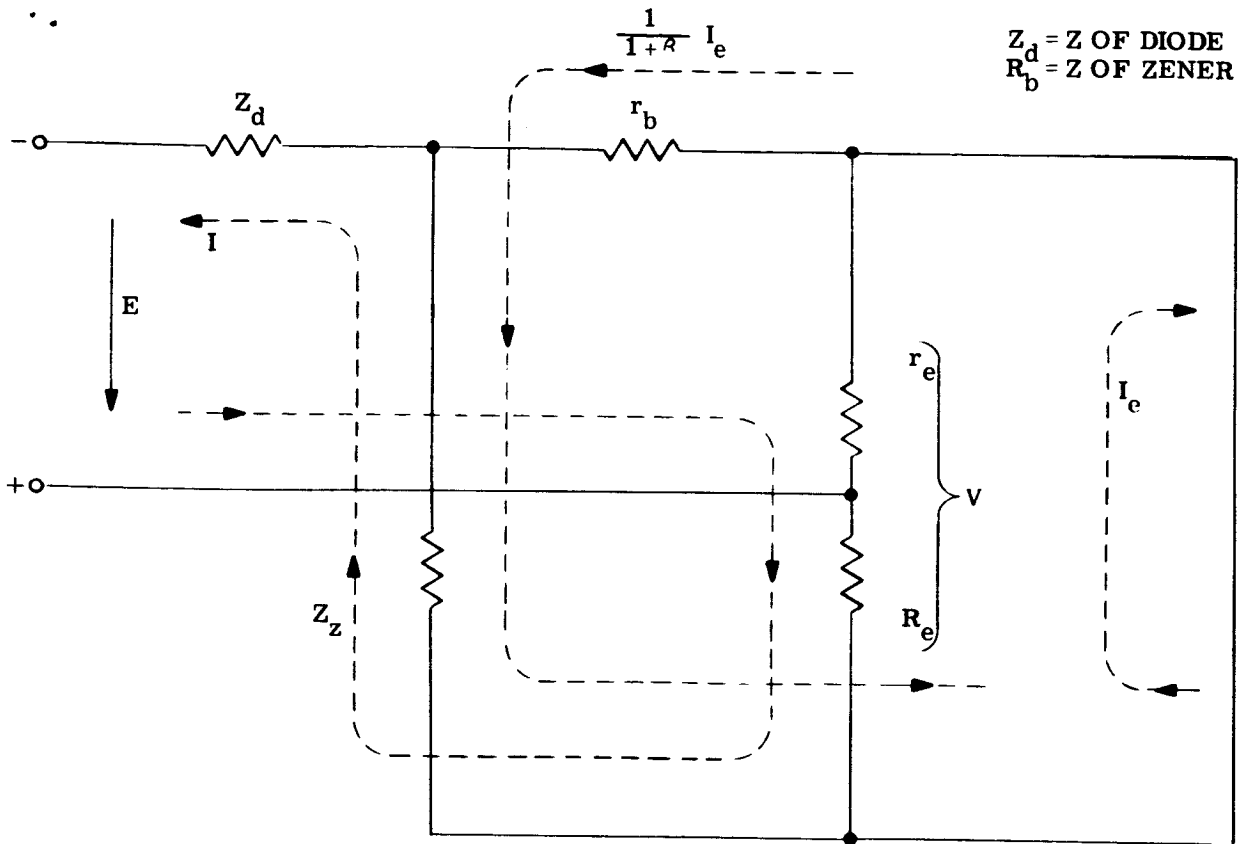


FIGURE 4-27.—Equivalent circuit of a readout amplifier.

The input impedance seen by the eye is

$$Z = \frac{E}{I} \quad (4.20)$$

To find I , the loop equations are solved. First, we have the relation

$$E = I(R_e + Z_z + Z_d) - I_e \left(\frac{1}{1+\beta} Z_z + R_e \right) \quad (4.21)$$

where

R_e is the emitter resistor
 Z_z is the zener diode impedance
 Z_d is the impedance of CR1
 β is current gain of the transistor

If the voltage dropped between the transistor junction and ground is V , then:

$$V = IR_e + I_e(R_e + r_e) \quad (4.22)$$

$$-V = -IZ_z + I_e \left(\frac{1}{1+\beta} \right) (r_b + Z_z) \quad (4.23)$$

Adding Equations (4.22) and (4.23) gives:

$$0 = -I(R_e + Z_z) + I_e \left[\left(R_e + r_e + \frac{1}{1+\beta} \right) (r_b + Z_z) \right] \quad (4.24)$$

The matrix of Equation (4.21) plus Equation (4.24)

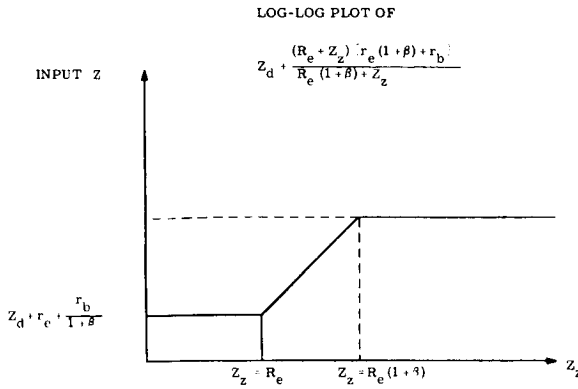
$$\begin{bmatrix} E \\ 0 \end{bmatrix} = \begin{bmatrix} R_e + Z_z + Z_d & -\left(\frac{1}{1+\beta} Z_z + R_e \right) \\ -(R_e + Z_z) & R_e + r_e + \frac{r_b + Z_z}{1+\beta} \end{bmatrix} \begin{bmatrix} I \\ I_e \end{bmatrix} \quad (4.25)$$

and

$$Z = \frac{E}{I} = \frac{\Delta}{R_e + r_e + \frac{r_b + Z_z}{1+\beta}} \quad (4.26)$$

or

$$Z = Z_d + \frac{(R_e + Z_z)[r_e(1+\beta) + r_b]}{(1+\beta)(R_e + r_e) + r_b + Z_z} \quad (4.27)$$


 FIGURE 4-28.—Response of input Z to Z_s .

With the eye currents used, the minimum value of R_e is $4K$. Therefore, r_e and r_b can be neglected in the denominator, and Equation (4.27) becomes

$$Z = Z_d + \frac{(R_e + Z_z)[r_e(1 + \beta) + r_b]}{R_e(1 + \beta) + Z_z} \quad (4.28)$$

As can be seen from Figure 4-28, the input impedance is not increased until Z_z approaches R_e . If $Z_z \ll R_e$ the important factors are Z_d and r_e .

The amplifier was temperature compensated by selecting a zener diode of such voltage that its temperature coefficient was nearly that of the base-emitter diode but of the opposite sign. The theory of this is as follows.

Temperature compensation of the amplifier requires that the collector current remain constant with temperature (assuming that the collector resistor is temperature stable as was the case with the metal-film resistors used). The collector current is αi_e and

$$i_e = (V_s - V_z - V_T) \frac{1}{R_e} \quad (4.29)$$

where

V_z = zener voltage

V_s = supply voltage

V_T = base-emitter diode drop

R_e = emitter resistor

Therefore, we have

$$\alpha i_e = (V_s - V_z - V_T) \frac{\alpha}{R_e} \quad (4.30)$$

It is required that the expression

$$(V_s - V_z - V_T) \frac{\alpha}{R_e}$$

be constant with temperature. The major change with temperature in this expression is the $-2.5 \text{ mV}/^\circ\text{C}$ change in V_T with temperature and the change in V_z with temperature. The temperature coefficient is slightly positive for α .

Since, above about $5v$, a zener diode has a positive temperature coefficient which increases with zener voltage, the amplifier is compensated if the zener voltage is at a value such that its temperature coefficient is near $+2.5 \text{ mV}/^\circ\text{C}$ (a coefficient slightly less than $+2.5 \text{ mV}/^\circ\text{C}$ is needed to offset the positive coefficient of α). A $6.8v$ zener has this coefficient. Using $6.8v$ zeners with a ± 5 percent tolerance resulted in amplifier output drift of a little less than $\pm 0.5 \text{ mV}/^\circ\text{C}$. An order of magnitude improvement can be realized in this value if the zener voltage is picked within this $\pm 5\%$ range to match the particular transistor diode coefficient and the particular transistor, α . On some transistors, such as the Fairchild 2N1613, the base-emitter coefficient from transistor to transistor is uniform enough so that the diode need only be picked for the α measured at room temperature. Thus, matching becomes possible without making temperature runs on the individual transistors.

The output sensitivity of the amplifier due to variations in supply voltage is primarily due to the zener voltage changing by a factor $\frac{Z_z}{(R_b + Z_d + Z_z)}$ times the change in supply voltage, where $Z_z + Z_d$ (the dynamic impedance of the zener and normal diodes) is very small compared to R_b . Therefore, for changes in supply voltage, Equation (4.29) becomes

$$\Delta i_e = \frac{\Delta V_s}{R_e} \quad (4.31)$$

since V_T is completely independent of current. Then

$$\Delta i_c = \alpha \Delta i_e = \frac{\alpha \Delta V_s}{R_e}$$

and

$$\Delta i_c = \frac{\Delta V_c}{R_c}$$

Then the change in collector voltage is

$$\Delta V_c = \alpha \left(\frac{R_c}{R_e} \right) \Delta V_e. \quad (4.32)$$

Thus it can be seen that R_e should be kept large compared to R_c to minimize sensitivity variation with power supply voltage change.

Preamplifier

The servo preamplifiers provided the proper load impedance for the error detectors and conditioned the error signals by converting the low-impedance, input-circuit current to a relatively high-impedance, circuit-stabilized voltage.

A simplified schematic of the preamplifier is shown in Figure 4-36, and the detailed schematics for the servo preamplifiers and readout

amplifiers are shown in Figures 4-30 and 4-31. Referring to Figure 4-36, the bases of Q1 and Q2 are clamped to a fixed potential by the zener diodes CR1 and CR2, so that the voltage at the emitters is held constant by the base-emitter voltage. The quiescent current flowing through each transistor at null (when no light current is flowing from the eyes) is slightly greater than the maximum current delivered by the photo-cells.

If the current generated by the eyes is as shown in the schematic, additional current equal to i_{eye} must flow through R2 and Q1 to prevent the voltage at the emitter of Q1 from changing. The quiescent current flowing in R5 and Q2 must decrease by precisely i_{eye} if no change in voltage is to occur at the emitter of Q2. There is a differential output between the collectors of Q1 and Q2. The two separate outputs are acted on by passive lead networks formed by C1, R7, R9, and R11; and C2, R8, R10, and R11, to

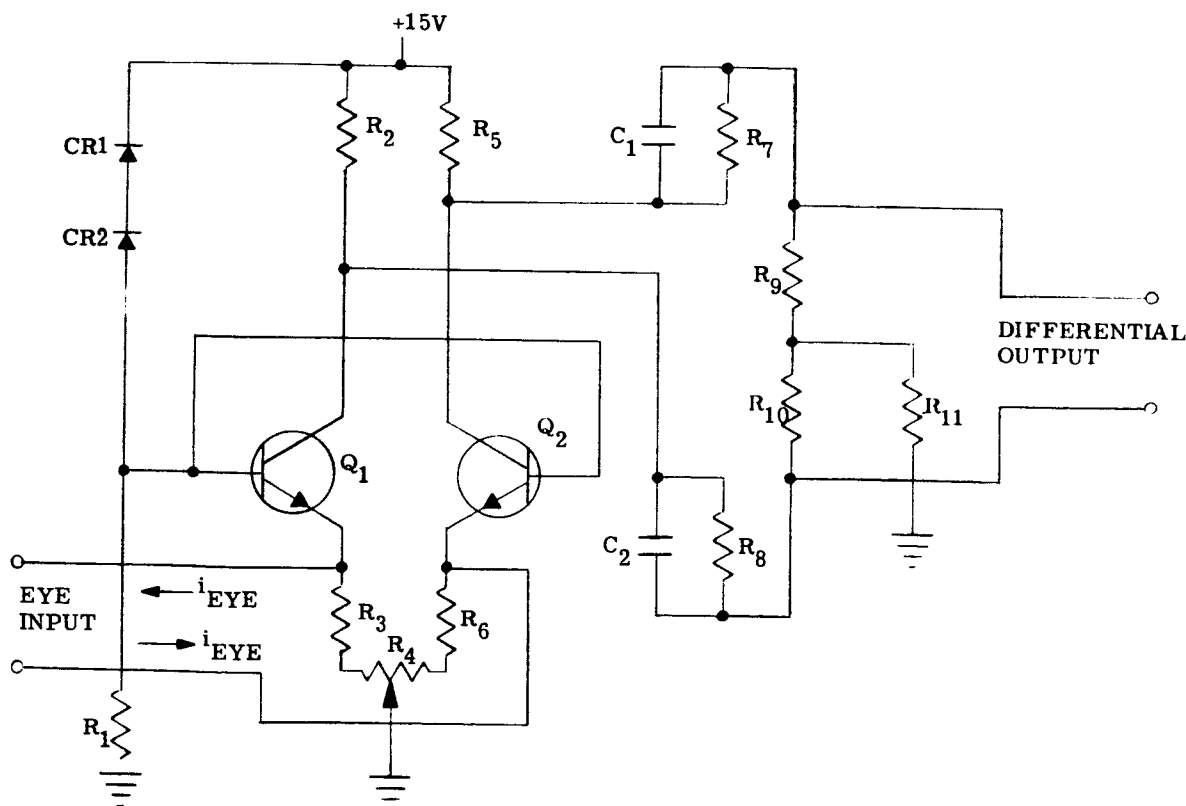


FIGURE 4-29.—Servo preamplifier.

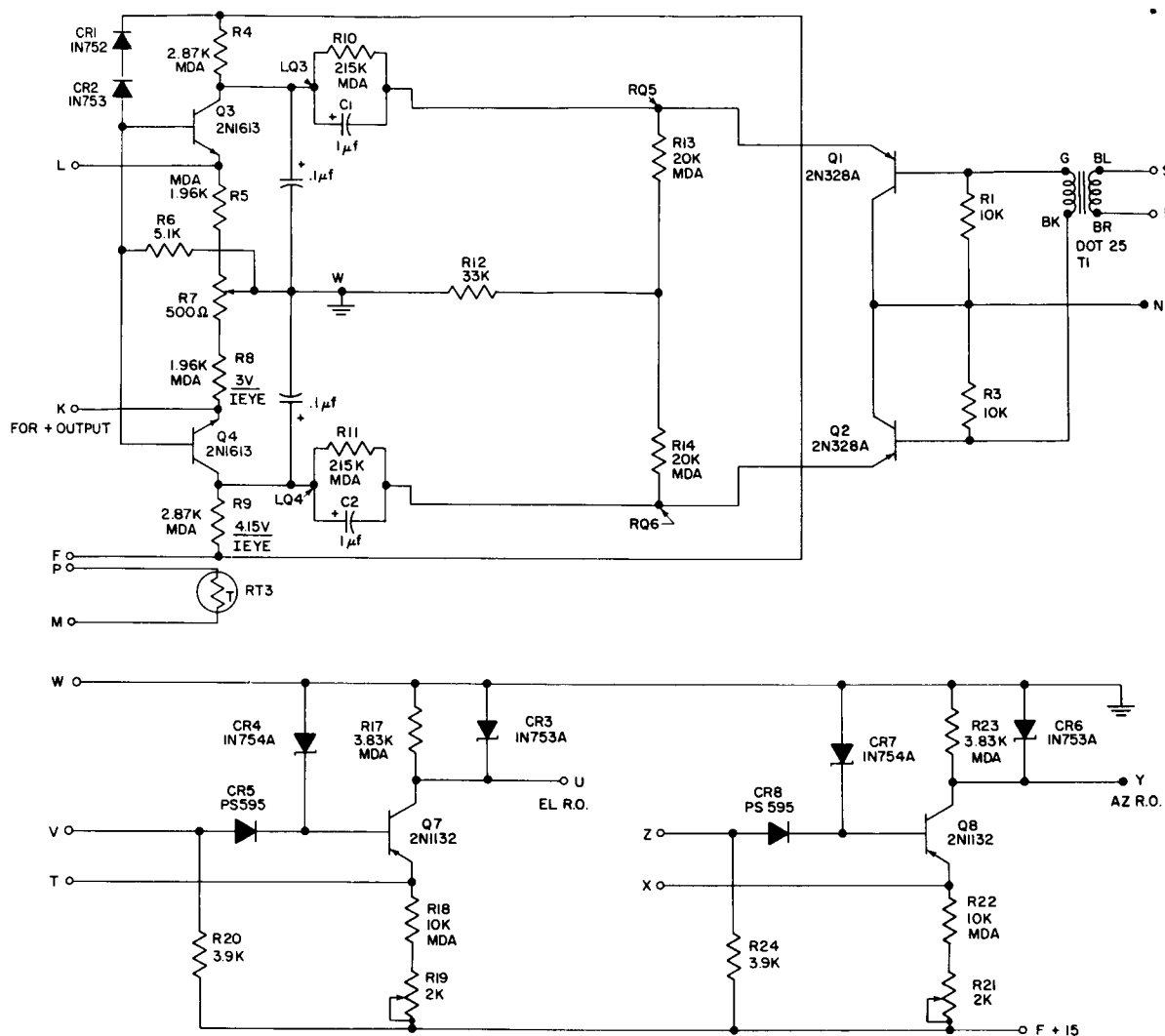


FIGURE 4-30.—Elevation servo preamplifier and readout amplifiers.

provide the necessary stability for the servo system. The differential stability of the lead networks is then sent to the chopper and ac amplifier.

The azimuth coarse and fine preamplifiers and the elevation fine preamplifier were identical except for different value components needed by differing coarse and fine eye currents and differing lead network parameters.

The preamplifier was mechanized as a differential amplifier because it was self-temperature stabilized to a first approximation, it had inherent rejection to supply voltage changes, and good common mode rejection. The bases

were tied together in order to minimize the effect of differential temperature coefficients. Thus, the temperature drift of this stage was the difference in the drifts in the base-emitter voltage, the collector and emitter resistors, and, to a small degree, differential drifts in α . The collectors of the preamp were biased to 10v to allow about a $\pm 5v$ peak swing.

Modulators

In order to provide the necessary gain for the servo loop, a fairly high-gain voltage amplifier was required. The standard method of synchronous chopping (modulating) of the input

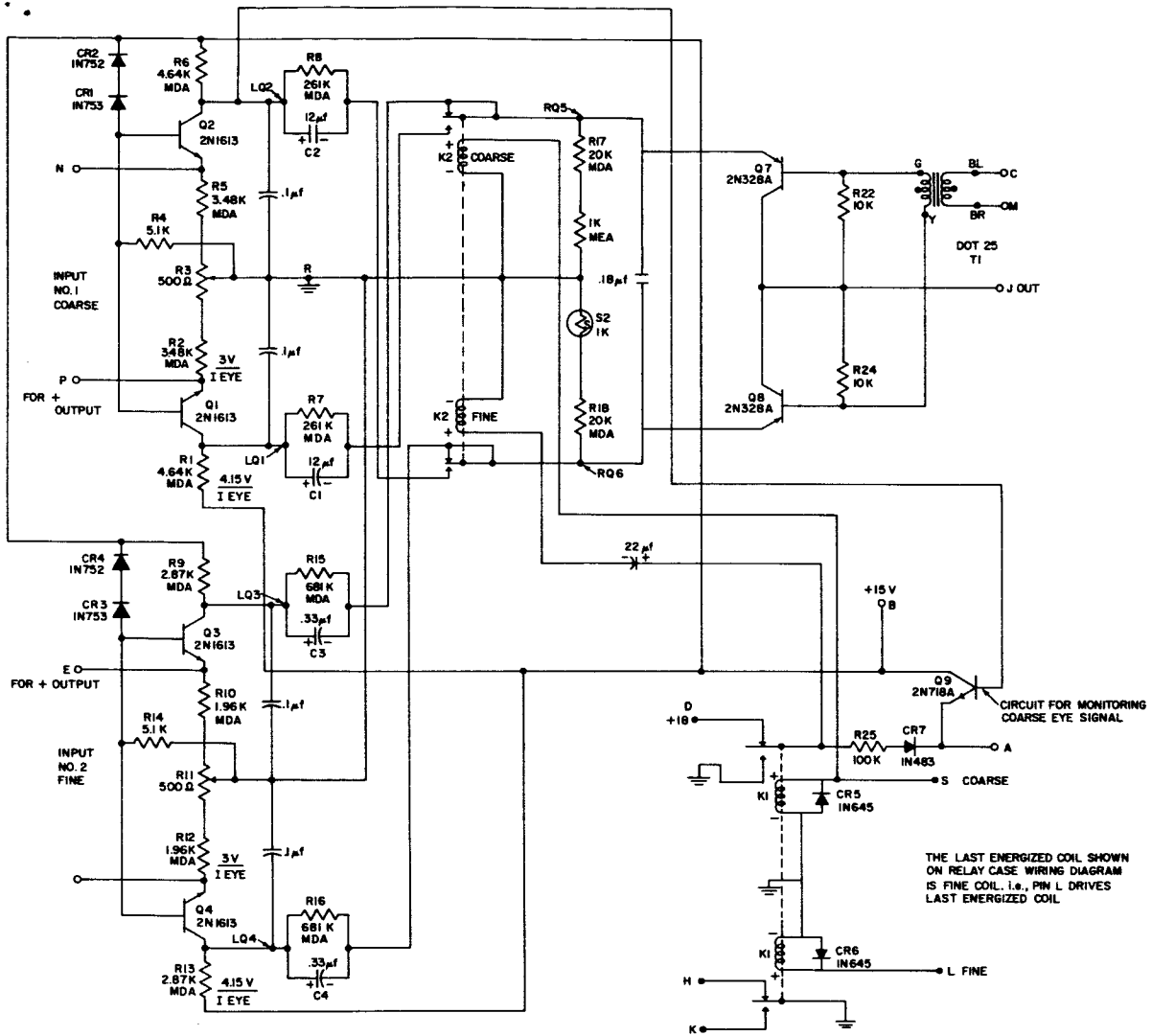


FIGURE 4-31.—Azimuth servo preamplifier and modulator.

and output of an ac amplifier to provide the necessary low drift characteristics was used.

The modulator alternately sampled between the differential outputs of the preamp. The output of the modulator was thus a square wave with peak-to-peak amplitude equal to the differential outputs of the preamp.

At a frequency such that the reactance of the lead capacitor was small compared to 20K, the signal output voltage to the modulator was nearly equal to the collector voltage swing, or 4.2v peak. Therefore, the modulator transistors had to be able to hold off this voltage.

Thus, the modulator drive voltage had to be greater than 4.2v peak, and the base-emitter diode had to have a breakdown greater than the peak-to-peak signal, or 8.4 v. The reasoning is as follows.

If the maximum signal were applied to the modulator, the voltage at Q1 emitter (Figure 4-32) would be 14.2v and at Q2 emitter, 5.8v. When Q1 is conducting, the collector voltage of Q2 would be about 14.2v. To keep Q2 from drawing current through the collector-to-base diode, its base voltage must be greater than 14.2v. Since the emitter of Q2 is at 5.8v, the

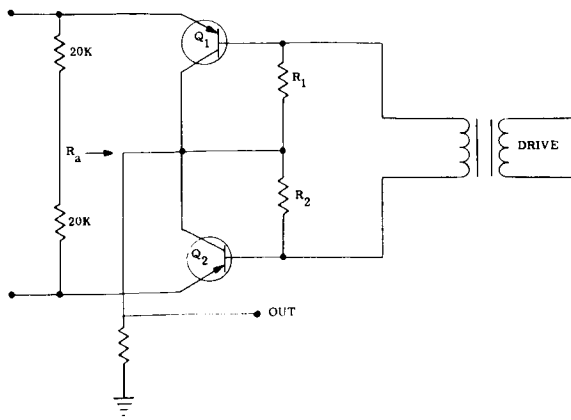


FIGURE 4-32.—Typical modulator circuit.

base-emitter breakdown must be in excess of the difference in these voltages.

The drive current must be such that the offset voltage (collector-to-emitter voltage when conducting) at the signal current being drawn is small compared to the required system resolution, and it must be greater than the current being drawn divided by β . The collector-base, β , is very low. In the case of the 2N328's it was near 1.

The drive current is controlled by the value of R_1 and R_2 . When Q_1 is cut off, the base-collector voltage of Q_2 is about 0.6v. Therefore, R_2 is nearly shorted out and the drive voltage less 0.6v appears across R_1 . Thus, the base current is nearly equal to the drive voltage divided by R_1 .

The modulator drive signal had to have a rise time, and the modulator transistor had to have a switching time, which was short compared to the chopping period, so that changes in these parameters would not affect the gain of the amplifier. This was easy to accomplish with typical conditions and components, except for the transformer. The transformer had to be driven from a lower impedance than that for which it was rated in order to obtain rise times faster than 50 μ sec. The low-impedance drive was also necessary to minimize droop in the modulator drive signal.

In addition to the transformer impedance and turns ratio, the interwinding capacitance was important since it was the greatest cause of switching transients in the modulator. The

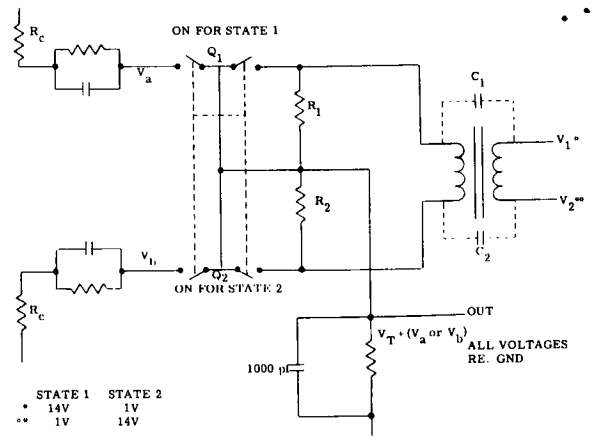


FIGURE 4-33.—Equivalent circuit of modulator transformer and associated circuit.

transients are caused by the transformer interwinding capacitance and the input impedance of the ac amplifier in parallel with the differential output impedance acting as a differentiating circuit. There are three effects that give rise to a signal through this differentiating circuit. They are as follows: (see Figure 4-33)

1. A nonbalanced to ground drive. An unbalanced drive causes the transient voltage V_t to vary from half-cycle to half-cycle. In other words, if the circuit consisted of only C_1 and C_2 , V_T would assume a value between V_1 and V_2 depending on the values of C_1 and C_2 . However, if during one-half cycle $V_1=13v$ and $V_2=2v$, then the other half $V_1=3v$ and $V_2=13v$, V_T would change. This would result in a spike every half-cycle as this voltage discharged through the lead network resistor.

2. If C_1 and C_2 were different in value there would be an effect similar to that in 1., above.

3. At the instant of switching, the peak-to-peak drive voltage appears across R_1 or R_2 in parallel with R_c , thus making V_T highly positive or negative depending on which transistor is on when V_1 is positive.

The pulse generated at switching had the same polarity for each half cycle. In the case of the modulator constructed, the pulses were always positive. However, the voltage was only 0.1 to 0.2 volts because the time constant of C_1 and R_1 paralleled by R_c (since a sharp pulse shunts the lead network and appears on

R_c) was about $0.14 \mu\text{sec}$. Since the rise times of the drive signals were considerably longer, the voltage across R_1 was the true derivative of the drive voltage, which only had a value during the rise or fall of the drive signal and was much smaller than the peak-to-peak voltage.

The first two effects gave positive and negative spikes on alternate half-cycles because the spike was the derivative of a square wave. These spikes were of greatest concern because of the pulse widths which were 10 to 20 μsecs . This pulse width was determined by the time constant of the 1000 μmf shunt capacitor of the ac amplifier discharging through the preamp collector resistor. Without this capacitor the height would have been much greater but the width would have been much less. The spikes were nearly eliminated by putting 0.1 μf capacitors from the collector to ground of each transistor in the preamp. Since these capacitors were across the collector resistor, they had little effect on the frequency response.

The third problem, caused by the transformer interwinding capacitance, was an off-balance at the modulator input. This was caused by the rectifier action of the choppers. When chopper Q_1 is conducting, C_1 dumps $C(V)$ coulombs on one lead capacitor, and C_2 removes $C_2(V)$ from the same lead capacitor.

When the drive switches polarity, C_1 removes $C_1(V)$ coulombs from the opposite lead capacitor, and $C_2(V)$ is injected into it.

With both preamp inputs at 10v, the voltage across C_1 for state 1 in Figure 4-33 is 4v. When it switches to the opposite state, it will charge to -9 volts if the drive is symmetrical. Thus, the total voltage change across each capacitor regardless of state is 13v. The current into one lead capacitor is $fV(C_1-C_2)$ and into the other lead capacitor is $fV(C_2-C_1)$.

Since most of this current flows through the two 20K resistors, the unbalance voltage is the difference between these currents multiplied by 40K or

$$40K \times 2KC \times 13V [(C_1-C_2) - (C_2-C_1)]$$

or

$$2080 \times 10^6 (C_1 - C_2)$$

The measured values of C_1 and C_2 in one transformer were 19 μmf and 26 μmf . The voltage unbalance caused by these values of interwinding capacitance is:

$$2080 \times 10^6 \times 7 \times 10^{-12} = 14 \text{ mw}$$

which is close to the measured unbalance of about 17 mv due only to chopper drive.

The effects on the amplifier of the first two problems were not serious since the spikes were well filtered at the output. However, the spikes caused a reduction in gain proportional to the spike area and the area of the signal output square wave. This resulted in minor cross-over distortion since the ratio became less with increasing signal level. The spikes also had various minor effects because the amplified spike could cause the output stages of the ac amplifier to limit.

The major problem was the change in C_1 and C_2 with temperature causing a change in unbalance voltage. This was the major cause of drift.

Modulator Driver

The modulator driver (see Figure 3-34) provided the switching drive to the modulator and demodulator. Its driving requirements were not severe except for rise and fall time. The circuit had to supply approximately 1 ma of drive current.

The rise and fall time was important because of its effect on output noise. In a carefully designed system, the output noise from spikes is primarily at the switching rate, since the majority of energy is at this frequency and the output filter has the least attenuation at this frequency. To keep the noise energy at the switching frequency to a minimum, the switching time must be kept at a minimum, thus requiring a fast rise and fall time.

To accomplish fast rise and fall times with normal audio transformers, the transformer driving impedance must be much less than rated. This also reduced droop. Double emitter followers were used to provide this low impedance.

The driver circuit used a unijunction trigger circuit to provide trigger pulses to a flip-flop.

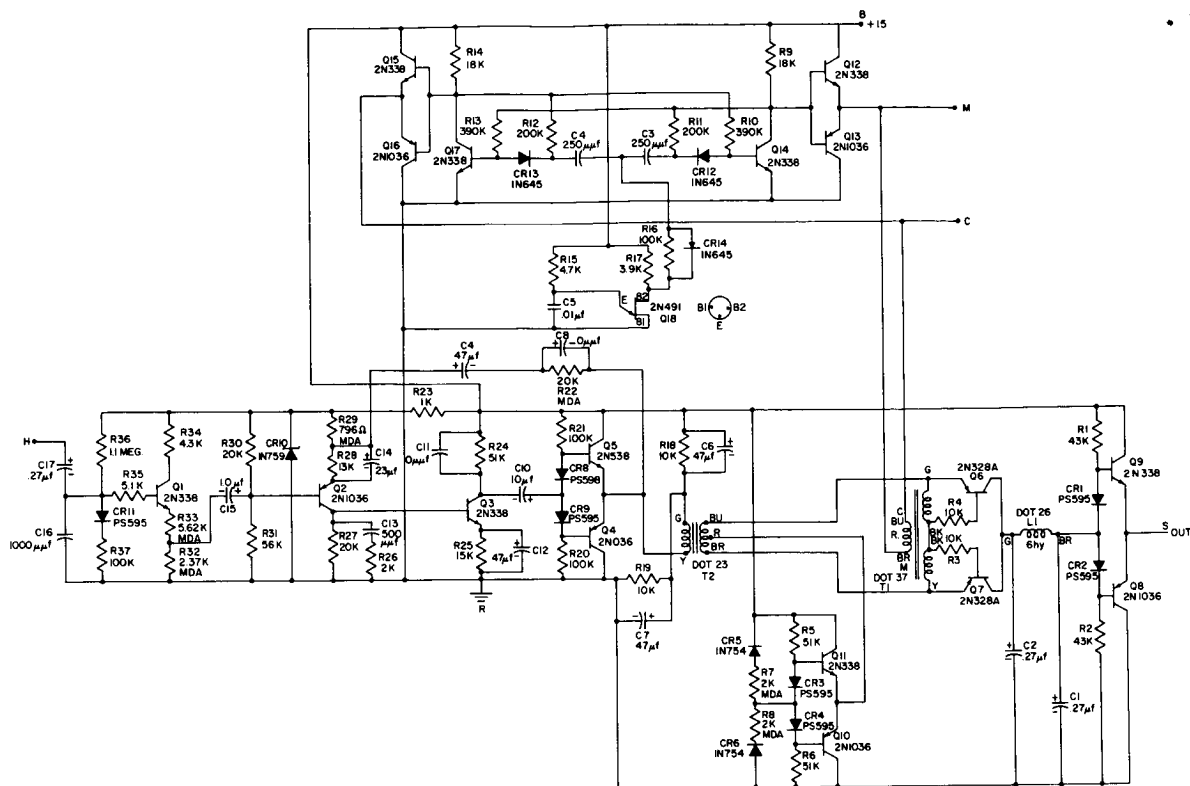


FIGURE 4-34.—AC amplifier, demodulator, and driver.

It was found that on occasion the fall of the trigger pulse turned on the transistor that the trigger rise had just turned off. To eliminate this, a diode shunted by a resistor to discharge the coupling capacitor was used in the trigger line. It was later found that the high-speed diodes used for CR12 and CR13 also cured this problem.

The flip-flop was common except for the steering circuit. The diode of the off-side of the circuit was reverse-biased so the turn-off pulse would not reach the base. However, the diode in the on-side was not reverse-biased and the turn-off pulse could readily pass to the base. This scheme provided positive steering of the pulses.

The flip-flop had good rise and fall times. To keep the flip-flop from being loaded and to provide a low-impedance drive for the transformers, complementary emitter followers were used on each output. This circuit provided the low impedance of an emitter follower on both half-cycles without the requirement for the very

high currents needed to accomplish this with an emitter resistor at drives approaching the supply voltage.

The transformers were connected from side to side instead of one side to ground to eliminate dc in the primary, thus reducing droop.

AC Amplifier

The ac amplifier amplified the square wave from the modulator (see Figure 4-34). It had to have sufficient bandwidth to preserve the rise and fall times, minimize droop, and maintain stable ac gain. The output impedance had to be low and not put dc in the primary of transformer, T2, so that a good square wave could be retained at the transformer secondary.

Lowest noise level from the modulator was obtained when the impedance that the modulator worked into was high. Also, since this impedance was high in comparison to the shunt impedance in the rate network (20K), the variation in the amplifier impedance had little effect on the lead network. Therefore, an

emitter-follower input stage was decided upon.

It was desired that the input stage clip at a level somewhat greater than that required for output saturation. To accomplish this, a resistor was put in series with the base and collector lead. When the signal went positive, the collector voltage decreased to the emitter voltage and clamped the emitter voltage at a value determined by the resistance divider made up of emitter and collector resistors, or about 5v. The resistor in the base lead limited the current when the clamping voltage was reached. The value of collector and base resistor had to be kept fairly low or the high-frequency response would have been jeopardized due to capacitance. Negative clipping was accomplished by transistor cutoff. Because it was originally planned to clip at $\pm 0.5\text{v}$, the emitter quiescent voltage was set at 0.5v. To stabilize this voltage and thus the clipping voltage and quiescent current against temperature drift, a diode was put in the biasing circuit.

To adjust gain, the emitter resistance was made up of 2 resistors as a voltage divider.

The two amplifier stages were conventional grounded emitter circuits, with feedback to the emitter of the first stage. The amplifier had a lag network in the first collector to control amplitude and phase at high frequencies to improve amplifier stability.

Decoupling was provided by a 1K resistor and a zener to ground. A zener diode was used instead of the conventional capacitor so that the gain would not be modulated by low-frequency ripple of supply voltage.

The output stage used complementary emitter followers for a low-impedance output. This stage differed from that used in the modulator driver outputs by the addition of the diodes in the base circuit. The purpose of the diodes was to provide proper bias for the emitter followers. The diode voltage drops tracked the base emitter voltage needed to keep a small quiescent current flowing in the stage and thus eliminate crossover distortion. They also provided low-impedance coupling to the bases of the two transistors.

To eliminate dc from the transformer primary, the end of the transformer which was not connected to the emitter followers was con-

nected to a voltage divider circuit which used capacitors to keep this end of the transformer at signal ground.

Feedback was taken from the transformer primary to reduce the transformer driving impedance and thus help to compensate for transformer deficiencies.

Demodulator

The demodulator consisted of a 7.5v reference, synchronous demodulator, π -section filter, and output complementary pair emitter follower for low-impedance drive (see Figure 4-34). Since it operated synchronously with the modulator, its output was a dc voltage proportional to the amplified solar detector eye signal.

The voltage reference was used to provide the quiescent output voltage for no ac signal voltage. The actual reference came from matched zener diodes. These zeners were put in series between +15v and ground with a resistor between them. The resistor was center-tapped, thus giving 7.5v. The 7.5v was fed to a complementary pair of emitter followers like those used for the ac amplifier output. This provided a low-impedance 7.5v so that the reference was not loaded at full output.

The demodulator consisted of the driver transformer, current-limiting resistors, and the demodulating transistors. The transformer provided a peak voltage in excess of the peak voltage expected from end to end from the ac output transformer to assure that the demodulator transistor block was cut off for one-half cycle.

The resistor limited the drive current to a value several times that drawn from the demodulator during the conduction half-cycle.

The transistors were alternately forward and reverse-biased. When forward-biased, the collector was clamped to the emitter and the drive current was limited only by the transformer resistance, current-limiting resistor, and forward diode drop of the base-emitter diode.

When reverse-biased, no current could flow in the base-emitter or base-collector diode if the voltage across the transformer was equal to instantaneous end-to-end voltage across the amplifier transformer. More explicitly, at maximum output, the voltage at the input to the

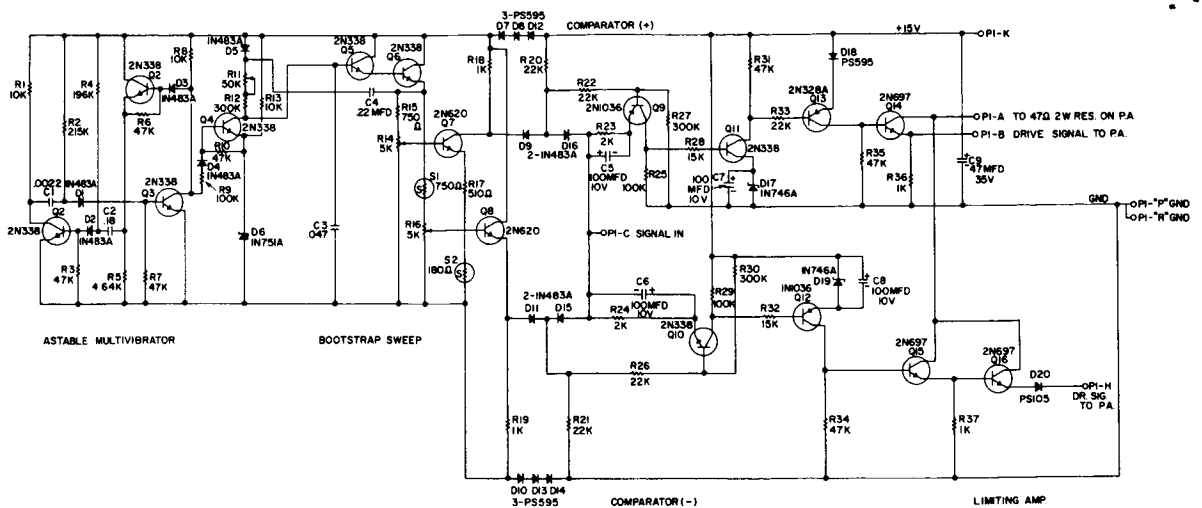


FIGURE 4-35.—Pulse-width modulator.

filter was 13v. The emitter voltage of the OFF transistor at this output was 11 volts less, or 2 volts. Since the base-collector diode was to be reverse-biased, the transformer had to put the base at a voltage in excess of 13v. Thus the transformer had to supply more than 13 minus 2, or 11v. Obviously, it was necessary that the base-emitter diode be able to withstand this reverse voltage. The only transistor found with this high rating was the Raytheon 2N328A, which had a V_{EBO} greater than 70 volts, ten times the maximum for the next highest transistor found.

The filter was a simple π -section filter used to filter the transient from switching, as well as to generate a rapid roll-off in both azimuth and elevation channels. The roll-off was necessary because of structural resonance. Although the phase shift at this resonance was such that the system could not oscillate, the resonance was excited by the pulse drive, and therefore gave a signal at the amplifier output. It was desired to keep this signal low in amplitude so that no motor drive power was consumed at this frequency.

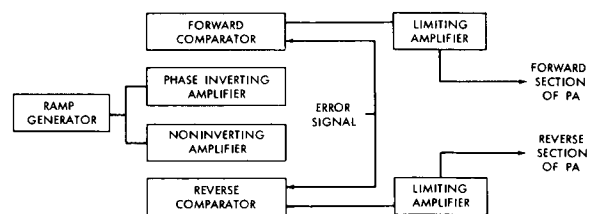
The output amplifier was of the same type used for the 7.4v reference, except for lower resistance base-biasing resistors. These were decreased to enable the output stage to supply more current in the resistors. At maximum drive, these resistors had only 2v across them.

Thus the output current divided by the transistor β at the coldest operating temperature had to be smaller than 2v divided by base-biasing resistance.

Pulse-Width Modulator

The pulse-width modulator supplied pulses with a duration proportional to the error signal from the demodulator. These pulses drove the power amplifier.

The schematic of this circuit is shown in Figure 4-35. The block diagram below illustrates the operation.



Two ramp signals were obtained from the ramp generator by means of an inverting amplifier and a noninverting amplifier. These two voltages were then compared with the input error voltage by means of a forward comparator and a reverse comparator. Figure 4-36 shows the relationship between the two ramp signals and the error signals.

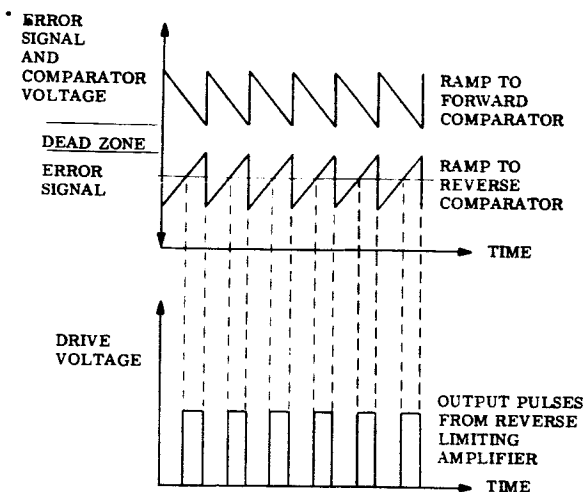


FIGURE 4-36.—Pulse-width modulator output characteristics.

When the error signal was of such a value that it was in the dead zone of the PWM, neither comparator supplied an output, and the servo was in its null position. If the error signal changed about the null value by an amount which placed it outside of the dead zone, one of the comparators supplied a signal to its limiting amplifier. The output of the comparator was the portion of the ramp lying between the error voltage and the dead zone. The limiting amplifier was driven into saturation for almost the entire duration of the signal from the comparator. The limiting amplifier output was therefore a pulse whose duration was equal to the duration of the signal from the comparator. Figure 4-36 shows how this pulse is derived. It can be seen that the width of the pulse is proportional to the error voltage and that it will vary from a zero width to a constant dc voltage.

Figure 4-37 shows how the motor torque varied with the error signal to the pulse-width modulator. The torque increased from zero torque linearly to a maximum torque at which point the drive from the pulse-width modulator was maximum or a dc voltage. This linearity of torque versus error signal was obtained by using a linear ramp and by making the pulse repetition period much greater than the motor armature time constant. In OSO I this repetition rate was 55 pulses per second. The choice of this pulse rate was a compromise

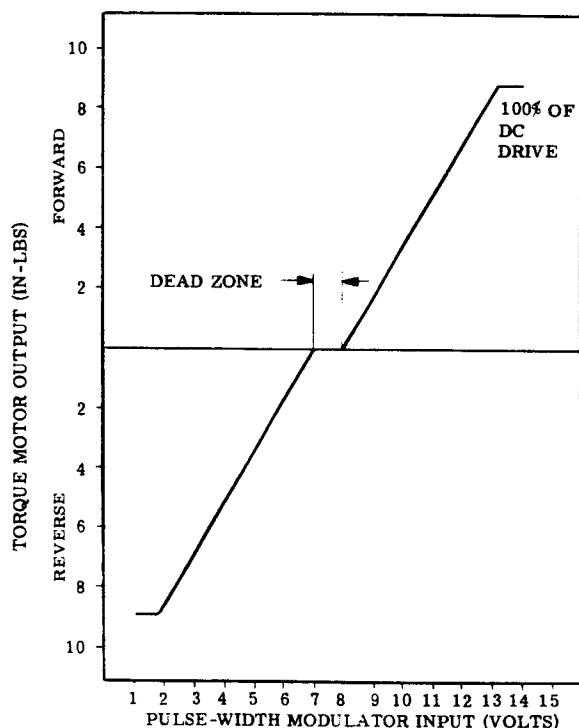


FIGURE 4-37.—Torque motor output versus error signal.

between a rate much higher than the cutoff frequency of the dc preamplifier and a pulse period much greater than the motor time constant.

Power Amplifier

One of the major requirements of the OSO I servo electronics was that it had to use a minimum of power. This requirement for low power consumption was met in part by designing the power section of the electronics to have a high power efficiency.

The most efficient mode of operation for a transistor is to use it as a switch. That is, the transistor is either cut off or it is driven into saturation. In the cutoff state, the transistor uses no power, while in the saturated state the power used is small because of the low saturation resistance.

The power amplifier used to drive the torque motors consisted of two transistors and two silicon-controlled rectifiers connected in a bridge circuit as shown in Figure 4-38.

The power amplifier operated as follows.

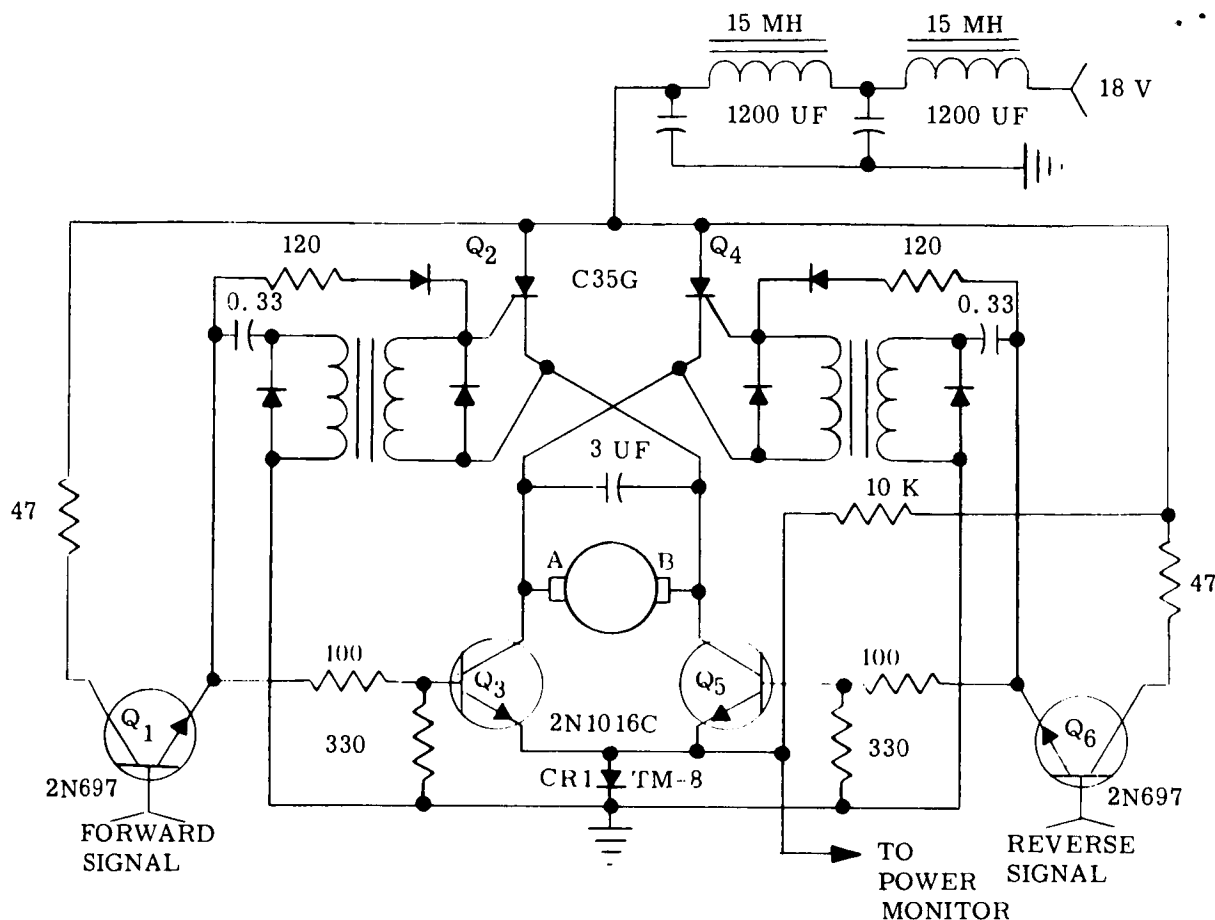


FIGURE 4-38.—Power amplifier schematic.

A positive pulse of about 10 volts amplitude from the pulse-width modulator was applied to the base of the emitter follower Q1. This pulse fired the silicon-controlled rectifier Q2 and drove the transistor Q3 into saturation. The torque motor was then connected across the line through Q2, Q3, and CR1 with brush B positive and brush A negative. The average torque developed by the motor was controlled by the duration of pulse applied to Q1. When the signal was removed from the base of Q1, Q3 ceased to conduct and reset the silicon-controlled rectifier. The voltage transient due to the motor inductance was limited to a safe peak value by the $3\mu\text{f}$ capacitor across the motor. Reversal of the motor was accomplished by applying a positive pulse to the base of Q6, which drove Q5 into saturation and fired Q4. This connected the motor across

the line through Q4, Q6, and CR1 with brush A positive and brush B negative.

Some typical instantaneous power distribution figures for this circuit with the motor stalled are as follows:

Motor armature resistance of 10 ohms

Supply voltage of 18 volts

Voltage drop due to Q2 and Q3, and CR1=3 volts

Modulator power=0.05 watt

Driving power from Q1=1.8 watts

Power dissipated in Q2, Q3 and CR1=4.5 watts

Power delivered to the motor=22.5 watts

Total power delivered by the supply =28.8 watts

Power efficiency of the power amplifier

$$= \frac{28.8 - 6.3}{28.8} \times 100\% = 78\%$$

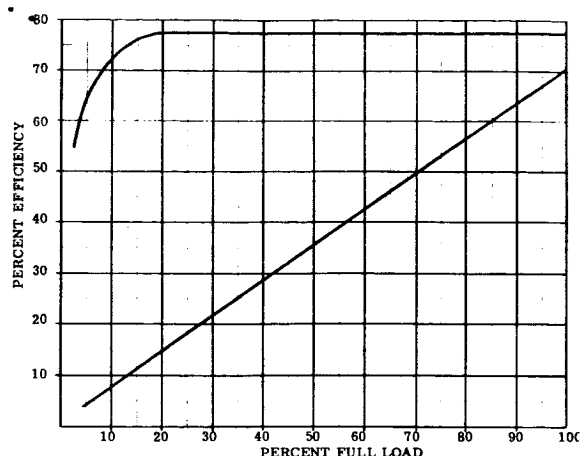


FIGURE 4-39.—Control electronics power section efficiency.

This efficiency was independent of the power delivered to the load except at small load power when the power required by the modulator and voltage regulator became appreciable compared to the power delivered by the power amplifier.

Figure 4-39 shows how the efficiency of the power section of the control electronics varied with load. This figure also shows how the efficiency of a conventional linear power section varied with load. It can be seen from Figure 4-39 that a pulse-width modulation system requires much less power than a conventional linear amplifier system except at near maximum loads. The normal load on the power amplifier was between 5 and 10 percent of maximum. This was the load after acquisition when the servo was maintained at null. During acquisition the load on the power amplifier increased to maximum value, but this lasted only for about one minute at the start of the 60-minute sunlit portion of the orbit. The circuit schematic (Figure 4-40) shows the power amplifiers, drivers, and monitor circuits.

A double L-section energy storage network or filter was used in series with the 18-volt power supply to provide peak currents for the power amplifier. The purpose of the filter was to reduce the pulse current load on the power supply, thereby preventing modulation of the power supply voltage at the power amplifier pulse rate. The filter also increased the efficiency somewhat

because with it in the circuit, most of the current was supplied by the solar-cell array. Without the filter the peak currents would have had to be supplied partially by the battery with a resulting decrease in overall efficiency.

15-Volt Regulator

Many circuits in the OSO I control system electronics required a stable supply voltage. The main spacecraft power was obtained from batteries, and the voltage on the power line could vary due to the state of charge of the batteries and due to varying loads. An electronic regulator was developed to operate from the battery voltage and supply a +15-volt output with a regulation of better than ± 1 percent for the expected line and load variations.

Figure 4-41 shows the schematic diagram for this regulator. The input voltage from the batteries was connected to pins M and P. The regulated 15 volts appeared on pins K and P. Transistors Q6, Q5, and Q3 were the high gain dc amplifier. Changes in the output voltage were coupled to the input of the dc amplifier by D1, D2, and R12. These changes were amplified and the signal appearing at the output was coupled to the base of Q4, the series regulating transistor. If the output voltage of the regulator tended to increase, then Q4 operated to minimize this increase. Figure 4-42 shows typical transfer characteristics for the regulator.

Spin Jet Control

The purpose of the spin jet control was to maintain the spacecraft wheel section rotating at a nearly constant rate so that the nutation damper would work effectively. The desired period of rotation for the OSO I was 0.495 seconds ± 0.025 seconds. The nutation frequency was directly proportional to the wheel spin rate. The nutation damper was a sharply tuned energy absorber designed to operate in a narrow range of wheel spin rates. A change in nutation frequency of about 5 percent reduced the effectiveness of the damper by a factor of 2 which was about all that could be tolerated. Torques to maintain the spin rate were provided by gas jets mounted on the arms. Gas was supplied to these jets by a solenoid valve that



FIGURE 4-40.—Servo power amplifier and driver circuits and power monitor circuits.

actuated when the spin decreased below the preset value.

Figure 4-43 is a block diagram of the spin jet control for OSO I. A photocell located on the rotating wheel supplied a current pulse every time its field of view swept across the sun. This current pulse was shaped to obtain a constant amplitude pulse with small rise and fall times. The differentiated output of the pulse shaper was used as a trigger input to a bistable multivibrator. The bistable multivibrator switched from one stable state to the other once every revolution of the wheel; therefore, the duration of the output pulses of this circuit was equal to the period of the wheel rotation.

A bootstrap-type integrator was used to obtain the time integral of the pulses from the bistable multivibrator. The integrator out-

putation was a ramp which had a peak value proportional to the period of rotation. The voltage ramp was compared with a reference voltage by a circuit designed to furnish an output only if the ramp exceeded the reference voltage. Therefore, as the wheel slowed down, the peak of the ramp increased. As soon as the ramps had increased above a critical value, the comparator supplied a trigger to the 4-second monostable. The 4-second pulse from the monostable multivibrator passed through one stage of power amplification and opened a solenoid valve. This valve supplied gas to the jets from gas storage bottles for 4 seconds, which was sufficient to speed up the wheel by the desired amount.

Several problems associated with this technique are not obvious. As the spacecraft comes

CONTROL SYSTEMS

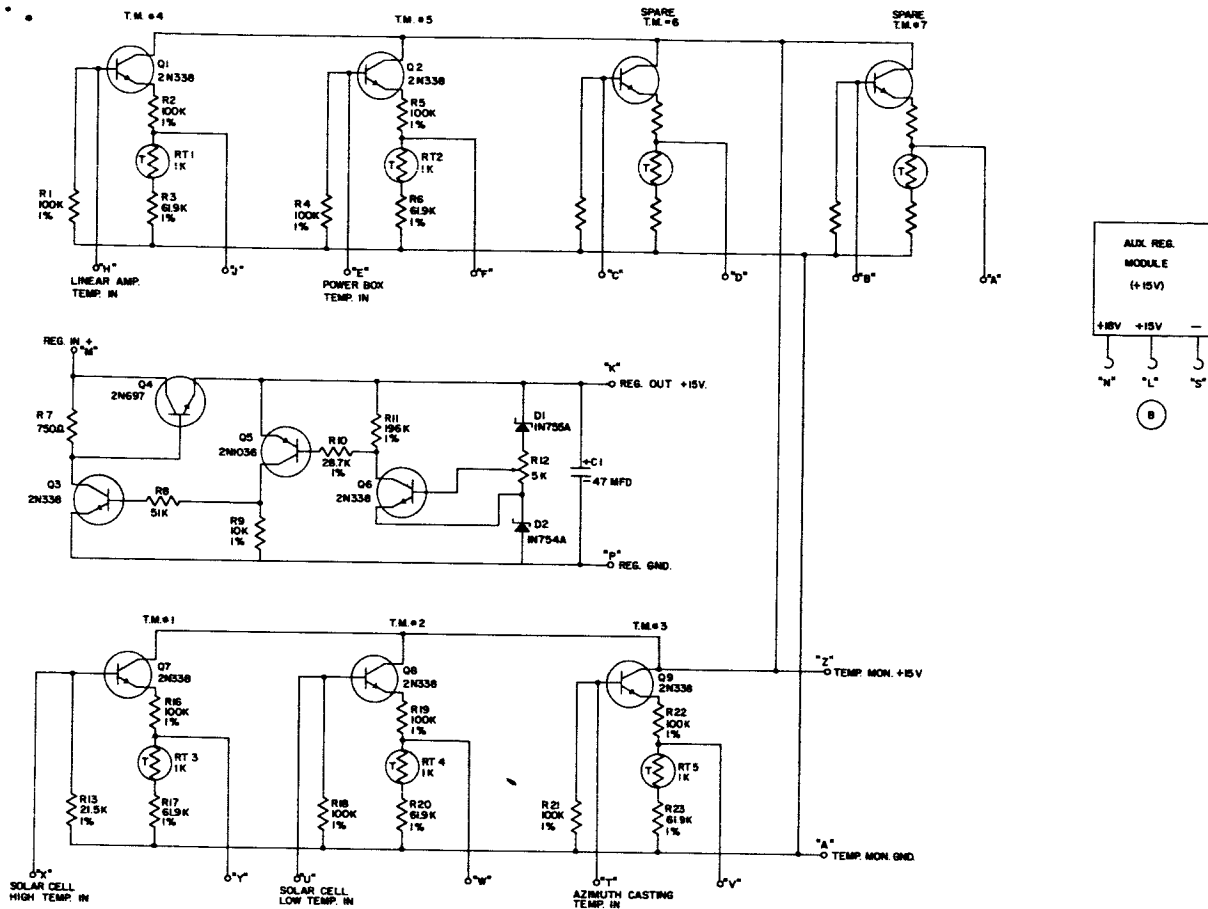


FIGURE 4-41.—Temperature monitor and regulator circuits.

into the sunlight, it passes through the penumbral shadow where only a part of the sun is visible. If the turn-on circuitry applies power

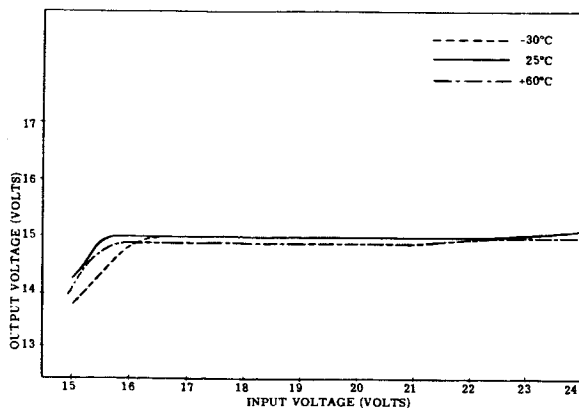


FIGURE 4-42.—Typical 15-volt regulation characteristics showing effects of temperature.

to the spin-up system before the spin eyes are producing pulses of sufficient amplitude, gas would be expended when not required. Thus, very high-spin eye sensitivity was required. Furthermore, to take another precautionary measure, the regenerative trigger discussed below has its power delayed for 5 seconds. It takes about 9 seconds for the spacecraft to go through the penumbral shadow.

Figure 4-44 is the schematic diagram of the electronics for the spin jet control system.

The circuit consisting of Q1 and Q2, etc., is for pulse shaping. In the absence of a current from the photocell, both transistors are cut off. The collector of Q1 has a potential of 15 volts, and the collector of Q2 has a potential of 0 volts. A signal causes both Q1 and Q2 to conduct and regeneration is provided by R1 and C1. The transistors Q1 and Q2 are therefore caused to

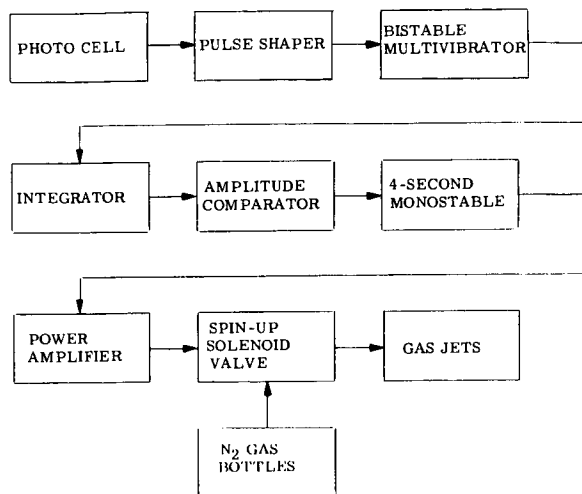


FIGURE 4-43.—Spin jet control system block diagram.

switch rapidly to full conduction by an input current.

The voltage at the collector of Q1 rapidly drops to a low value and remains there as long as there is sufficient input current. When the input current drops below a critical value, both transistors switch off. The output signal from Q1 is a negative square pulse having relatively constant amplitude and fast fall and rise times.

This pulse is differentiated by C3 and the negative spike triggers the bistable multi-vibrator comprising Q3 and Q4. The bistable circuit is a conventional flip-flop.

The time integral of the flip-flop output is obtained in a synthetic manner by use of a bootstrap ramp generator. This circuit comprises Q5, Q6, and Q7. When Q4 is cut off, base current is supplied to Q5 which shunts the current through R16 around C8 to ground. When the flip-flop is switched to the other stable state, the base drive to Q5 is removed and the current through R16 charges C8. Capacitor C7 is the bootstrap capacitor. It is charged to 15 volts and acts as an additional 15-volt power supply. The voltage on C8 is coupled to the integrator output by a double emitter follower where it appears in series with the voltage across C7. As C8 charges, the voltage supply to R16 also increases giving a linear rising ramp at the integrator output. This ramp voltage rises until the flip-flop is switched back to its first stable state at which time the ramp is abruptly terminated. The output of the integrator then drops to its minimum value. It is evident from the above discussion that the peak voltage of the ramp is proportional to the spin period.

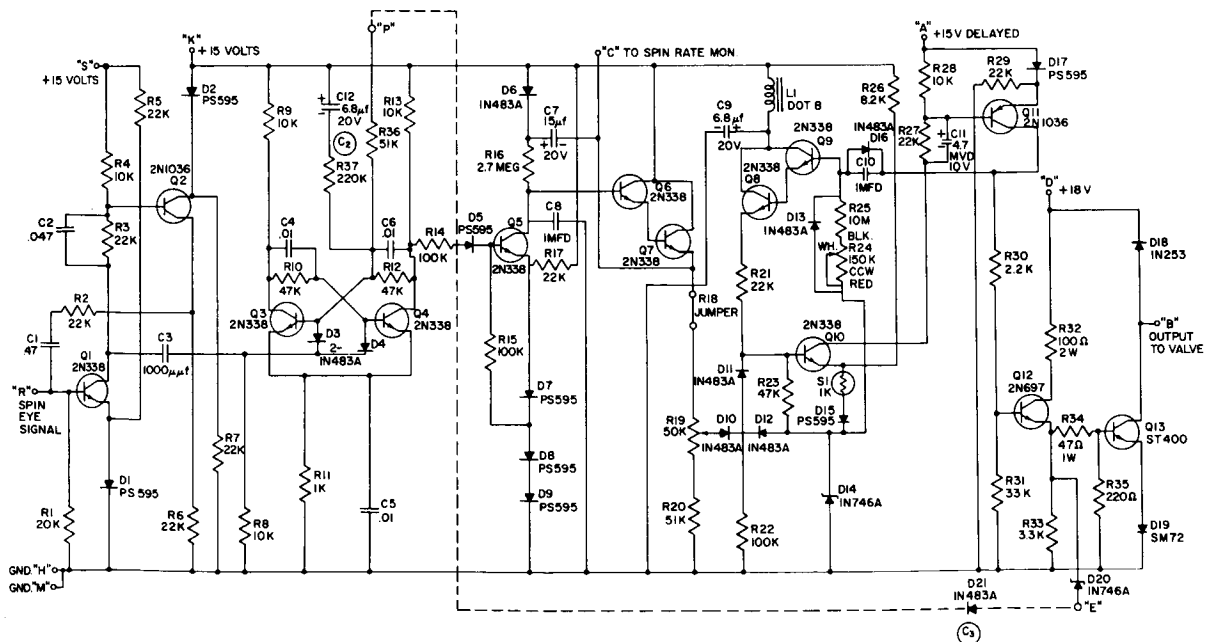


FIGURE 4-44.—Spin jet control system schematic.

The voltage comparator consists of the zener diode D14 and diodes D10 and D12. D14 supplies the reference voltage which appears across R22. When the voltage at the arm of the potentiometer R19 exceeds the reference voltage, diode D12 is back biased and the potential across R22 rises. This increase of voltage on R22 is the trigger signal for the 4-second monostable multivibrator.

The 4-second monostable circuit comprises transistors Q8, Q9, Q10 and Q11. This circuit operates the same as the pulse shaper except that it operates as a monostable, rather than as a Schmitt type circuit. The time the circuit spends in the transient or unstable state is determined by the time constant of C10 and the resistance immediately to the left of C10. Temperature stabilization of the spin controller operating point is provided by S1, a silicon resistor, which has a positive temperature coefficient.

The application of 15 volts to Q11 was delayed because, when the voltage is first applied to the spin-circuit, C7 charges through D6 and R18, R19 and R20. This causes a large transient at the input to the comparator which is sufficient to trigger the 4-second monostable. The delay of voltage to Q11 allows this transient to die out.

The power amplifier used to drive the solenoid valve is made up from transistors Q12 and Q13.

The anticipated temperatures in the wheel were within the range of 0° C to 25° C. Within

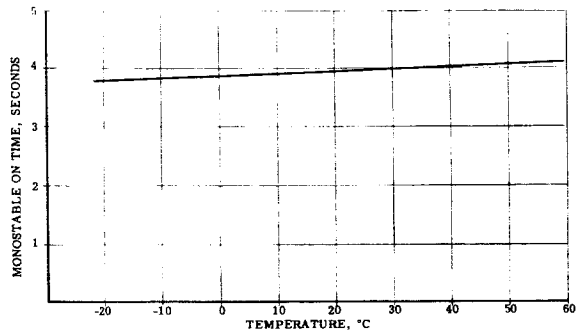


FIGURE 4-46.—Temperature dependence of 4-second monostable.

this temperature range the spin controller was to maintain the desired spin rate within ± 5 percent. Figure 4-45 shows the effect of temperature on the operating point of the spin circuit. It can be seen that the set point variation is less than ± 5 percent over the anticipated temperature.

Figure 4-46 shows the temperature dependence of the 4-second monostable which controls the valve open time.

Input signal sensitivity of the spin controller was such that proper operation would be obtained if the photocell resistance dropped below the illuminated value listed below:

Temperature (°C)	Photocell Upper resistance	Illuminated photocell resistance
-30-----	500K	168K
25-----	500K	209K
60-----	500K	250K

These sensitivity figures represent the maximum practical sensitivity capability of the spin circuit. The reason for setting this circuit to maximum sensitivity was to insure that the spin controller had a sufficiently large input signal for operation when the power was turned on by a less sensitive turn-on circuit at dawn.

The standby power consumption of the spin controller was 90 milliwatts; i.e., 5 ma from the regulated line.

During the despinning operation, the spin control functioned as follows: The spacecraft and third-stage engine assembly were spun up to approximately 2 rps before the third-stage engine fired. This provided spin stabilization for the assembly under thrust conditions. This spinup was the zero time reference for the space-

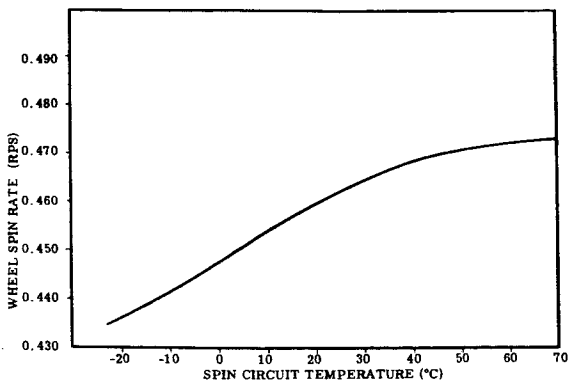


FIGURE 4-45.—Dependence of spin rate on spin circuit temperature showing wheel spin rate at gas jet turn on.

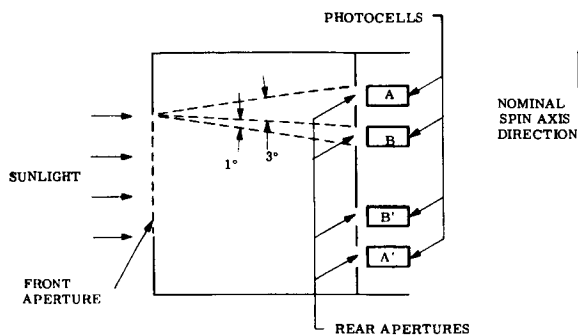


FIGURE 4-47.—Pitch control eyes schematic.

craft sequence timer. At 200 seconds after third-stage spinup, the sequence timer energized the despin relay. (The third-stage engine had been separated from the spacecraft and the spacecraft's arms had been extended prior to 200 seconds.) The despin relay applied power to open a solenoid valve which allowed gas to be supplied to the despin jets. These jets slowed the wheel down until the lower limit of the spin-control circuit was passed, at which time the normal spin-control circuit actuated and allowed

gas to be supplied to the spin-up jets for 4 seconds. As soon as the spin-up circuit opened the spin-up solenoid valve, the despin relay was latched over to its original state which closed the despin solenoid valve. This prevented any further actuation of the despin solenoid valve.

Pitch Control System

The pitch control system kept the spin axis of the spacecraft approximately normal to the solar vector so that the elevation servo system was within its operating range. The controller operated whenever the spin axis was more than 3 degrees from the normal to the solar vector. The controller operated gas jets which precessed the spacecraft until the spin axis was approximately one degree out of normal to the solar vector in the opposite direction. A schematic of the pitch control eye system is shown in Figure 4-47, and a block diagram illustrating the total pitch control system is given in Figure 4-48.

SUPPLY VOLTAGE

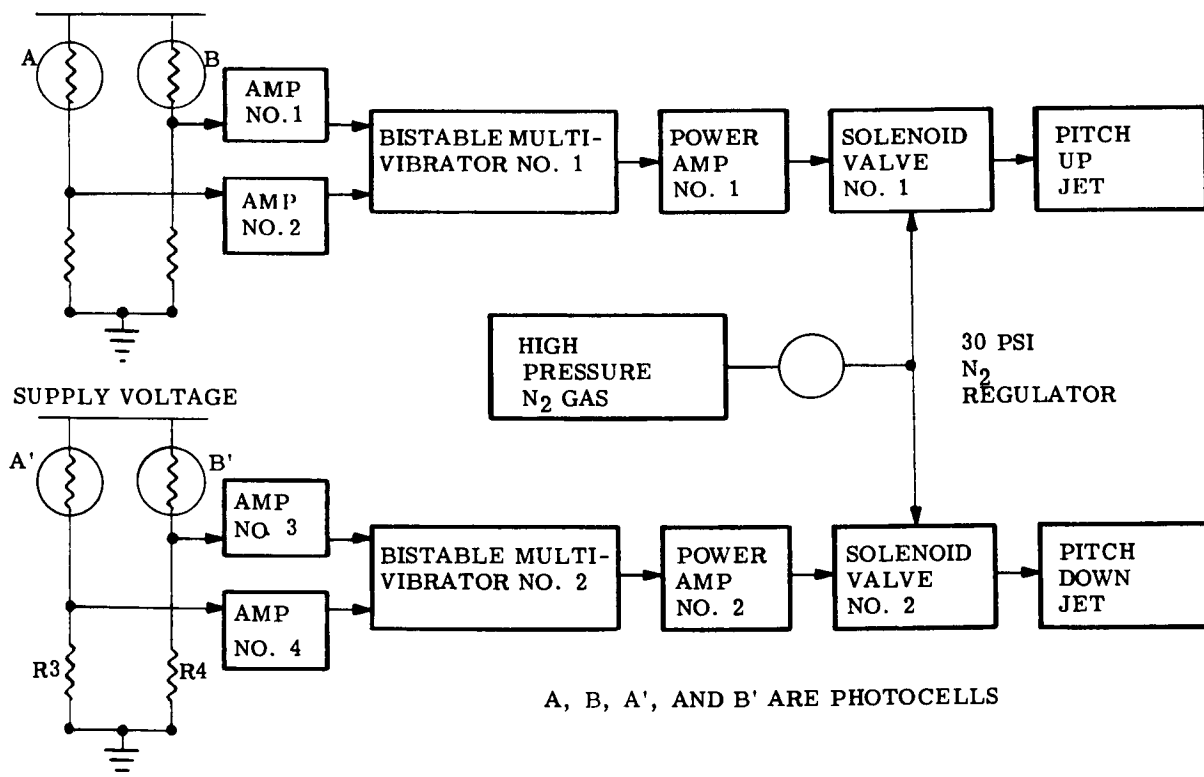


FIGURE 4-48.—Automatic pitch angle control block diagram.

The control photocells, Clairex CL-7 cadmium-sulphide detectors, have an electrical resistance which is a function of illumination. When they are illuminated by sunlight, they have a low resistance (a few thousand ohms); and when they are dark they have a resistance of several megohms. Referring to Figures 4-47 and 4-48, with $89^\circ < \phi < 91^\circ$, the outer two photocells are shadowed by the mask and very small signal currents are supplied to the electronic control. Photocell B is illuminated. Its resistance is a low value which causes a relatively large current to flow into the input of Amplifier No. 1. This input current holds the output voltage of Amplifier No. 1 at zero, thereby removing the reset signal from bistable multivibrator No. 1.

As soon as ϕ reaches 93° , Photocell A is also illuminated and a large signal current is supplied to Amplifier No. 2 which causes its output voltage to change from zero to some large value. The output voltage from Amplifier No. 2 causes the bistable Multivibrator No. 1 to switch to its alternate stable state which supplies an input voltage to Power Amplifier No. 1. Power Amplifier No. 1 in turn operates the solenoid valve which supplies compressed gas regulated to 30 psi to the up-jet nozzle. The pitch-up jet applies a torque to the spacecraft in such a direction as to cause ϕ to decrease. As ϕ decreases, first Photocell A is darkened and the output of Amplifier No. 2 drops to zero. When $\phi = 89^\circ$, a 1° overcorrection, Photocell B is also darkened and the output voltage of Amplifier No. 1 goes from zero to some large voltage. The output from Amplifier No. 1 causes the bistable Multivibrator No. 1 to switch to its original stable state. This removes the driving signal from Power Amplifier No. 1 which closes the solenoid valve and removes the torque from the spacecraft.

For $\phi < 89^\circ$, the operation is the same except Photocells A' and B' and their associated control electronics are employed. Figure 4-49 shows the electrical schematic of the automatic pitch control system.

Day Power Turn-on

During the dark portion of the satellite orbit, all electronic systems using the sun for a signal source were turned off. These include all of the upper section electronics and instruments and the spin control system in the wheel.

As mentioned before, an important requirement of the day power turn-on circuit was that it should not apply power to the spin control system until this circuit had received sufficient sunlight to insure reliable operation. The reason for this requirement was that the spin control system could have interpreted an absence of an input signal as being due to a low wheel spin rate and would therefore expend gas.

The sensitivity of the day power turn-on circuit was set so that it would not apply day power until there was sufficient sunlight to cause each input photocell resistance to decrease to less than 40 kilohms. The sensitivity of the spin control was set so that as soon as there was sufficient sunlight to cause its input photocell resistance to drop below 100 kilohms, reliable operation would be obtained.

Figure 4-50 shows the location about the wheel of 6 parallel connected turn-on photocells together with their resistance in sunlight as a function of their orientation to the sun. The dark or night resistance of the parallel combination of turn-on photocells was 150 kilohms.

Figure 4-51 is the schematic diagram of the printed circuit board which contained the day power turn-on circuit plus the despinn relay, spin rate monitor amplifier, and the delayed 15 volts for the spin control circuit. Operation is as follows:

The six photocells located around the wheel are connected in parallel and supply the input signal to the day power turn-on circuit. The photocells are connected between the base of Q1 and the 18-volt supply bus. In sunlight the combined resistance of the eyes drops to less than 15,000 ohms so that sufficient current is supplied to the base of Q1 to cause it to saturate. The drop in potential at the collector of Q1 is coupled to a magnetic latching relay through transistors Q5, Q6, and Q7.

ORBITING SOLAR OBSERVATORY

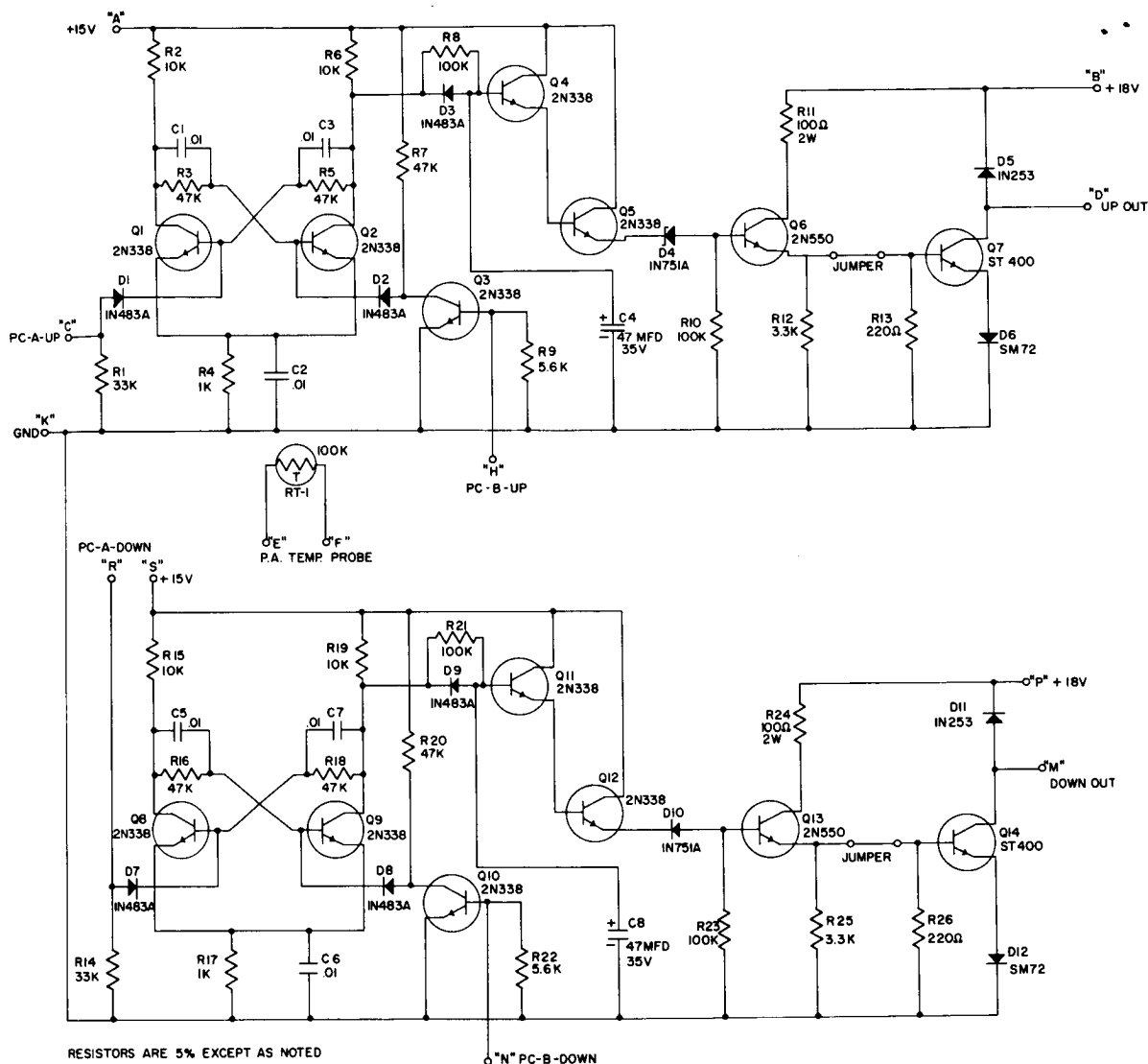


FIGURE 4-49.—Pitch control schematic.

The relay is latched over and turns on the day power. In order to conserve power, a set of contacts on this same relay is used to remove the voltage from the last energized relay coil once the relay is latched.

When the satellite enters the dark portion of the orbit, the photocell resistance increases and Q1 is cut off. The positive potential at the collector of Q1 is coupled through transistors Q2, Q3, and Q4, thus energizing the OFF relay coil. The relay latches over to remove power from the day power bus and at the same time grounds the base of the relay

driver transistor to remove power from the relay coil.

Undervoltage Switch

The undervoltage switch was located between the spacecraft batteries and the spacecraft battery load.

The spacecraft batteries were nickel-cadmium cells which had a rather flat discharge curve until they became almost fully discharged. Near the fully discharged state, the voltage would fall abruptly and the impedance would rise so that very little power could be drawn

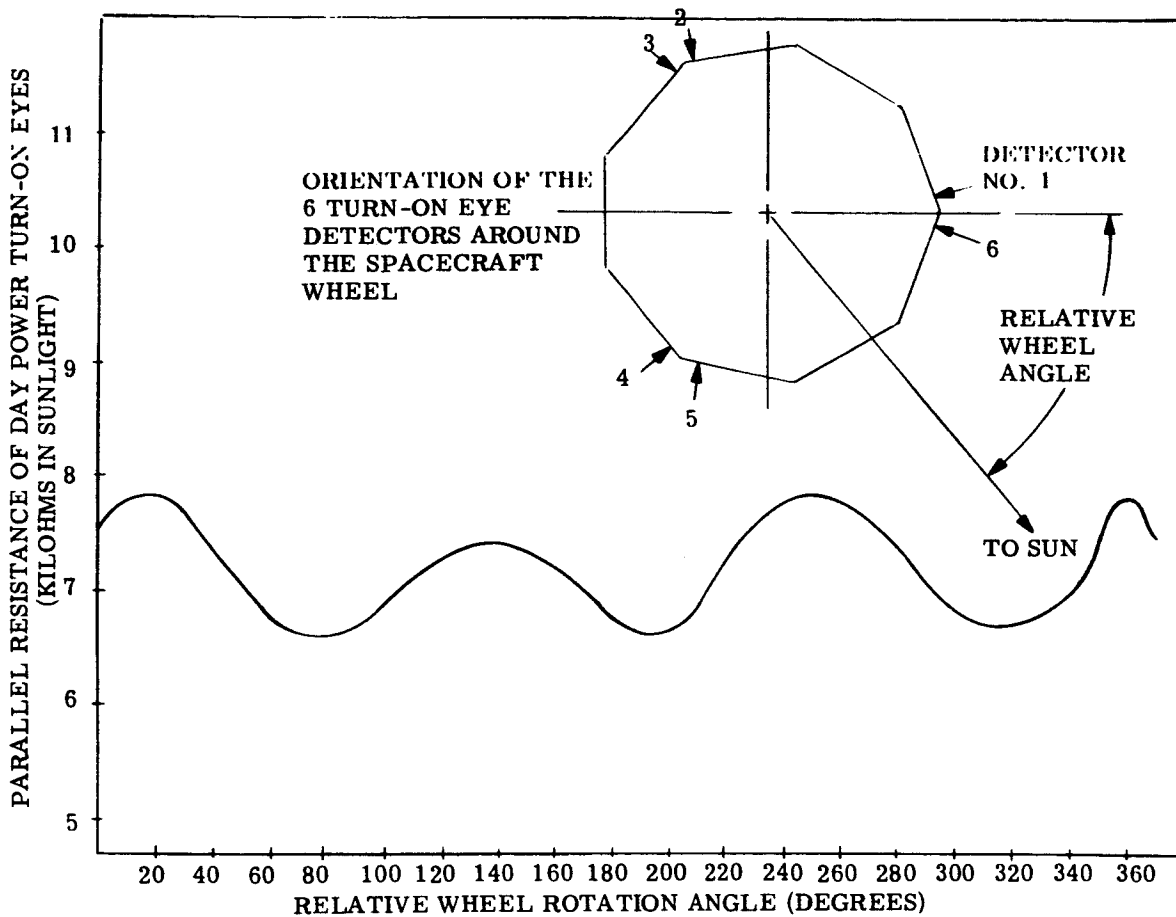


FIGURE 4-50.—Detector output characteristics.

from them. If the batteries had become almost fully discharged they would not have been able to deliver the current necessary to allow the servo to catch and orient the upper section. The azimuth servo would have tried to acquire the sun throughout the daylight portion of the orbit and thus would never have permitted the batteries to recover. This would have led to a disabling of the spacecraft. By ground command, sections of the spacecraft could be turned off to conserve power so that the undervoltage condition could be corrected. However, if the command system was not operative and the batteries discharged to the point where the voltage dropped abruptly, a last measure was available to prevent the loss of the vehicle. This was the undervoltage switch which could shut down the entire

spacecraft and permit the batteries to charge from the rotating solar-cell array. When the batteries were about 10 percent charged, the undervoltage switch turned on the spacecraft again. A block diagram of the undervoltage switch is shown in Figure 4-52.

All the power to the spacecraft arrived through a 6-pole double-throw latching relay. Five of the poles carried spacecraft power and one was used for circuit switching by the undervoltage switch.

Figure 4-53 shows a typical charge-discharge curve for an OSO I battery pack. Reference to this figure shows that the battery voltage starts dropping rapidly once it reaches 17 volts, and at 15.5 volts the drop in voltage is so great that the battery pack can be considered discharged at this point.

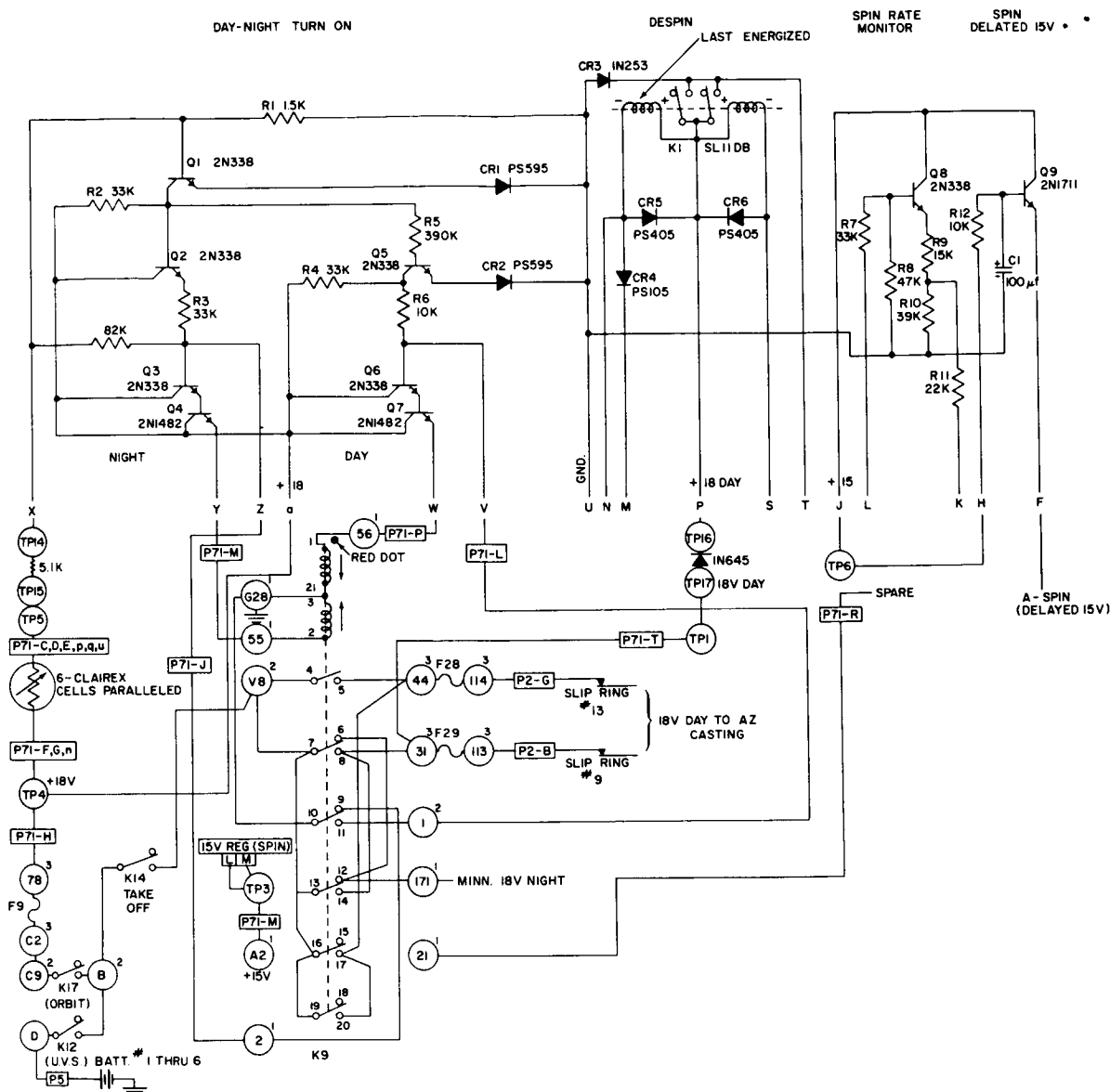


FIGURE 4-51.—Day night turn-on despin schematic.

On *charge* the battery voltage rapidly increased until it was 19.0 volts, or about 10 percent charged. Above 19 volts, the rate of increase of voltage was approximately constant until a peak was reached at about 90 percent of full charge.

The undervoltage switch was designed to remove the load from the batteries if their voltage dropped below 15.5 volts, and it would not reconnect the load until the battery voltage increased to 19.9 volts. The choice of the

upper switching point of the undervoltage switch was dictated by the temperature characteristics of the batteries (see Figures 4-54 and 4-55). The turn-on point of the undervoltage switch had to be chosen low enough so that if the batteries were warm, that is, 30°C , the voltage would increase above the set point. If the batteries were cold, that is -10°C , the set point had to be high enough to insure that the batteries receive a minimum charge before the load was reconnected.

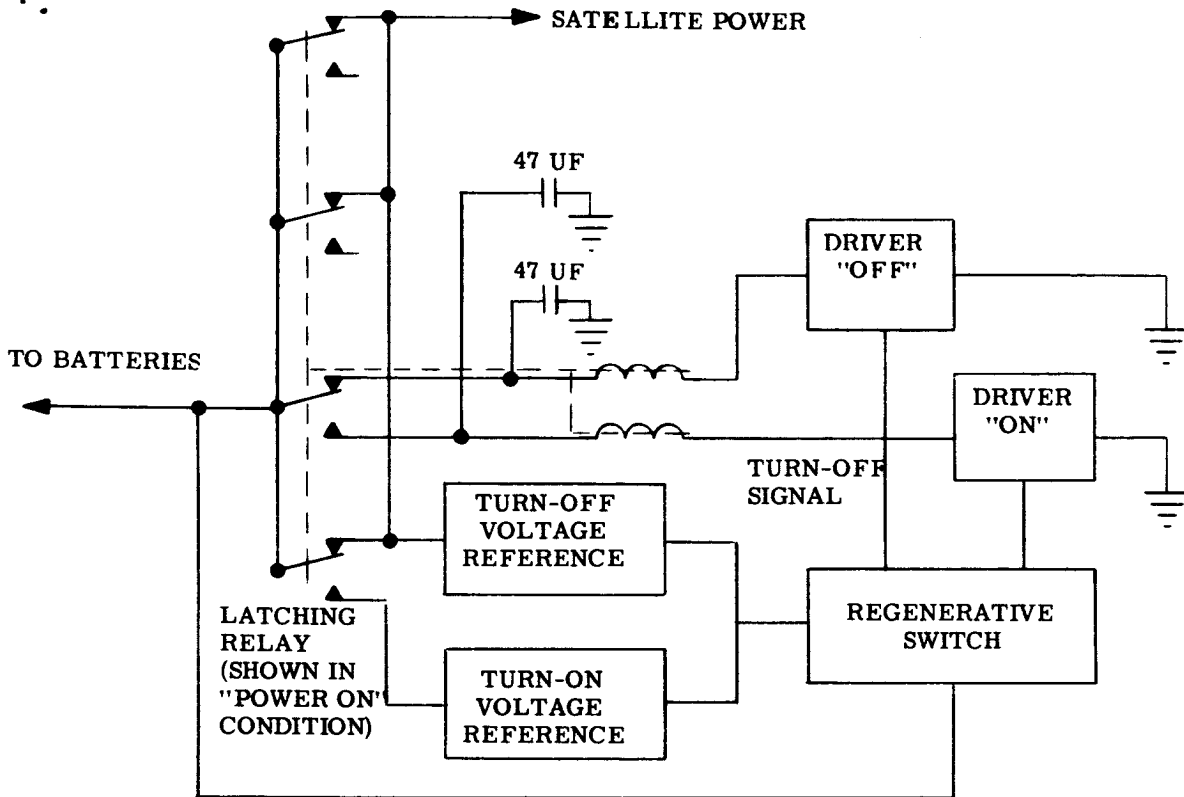


FIGURE 4-52.—Undervoltage switch block diagram.

Figure 4-56 is the schematic diagram of the undervoltage switch together with some temperature monitoring circuits and a voltage monitor circuit.

Transistors Q1 and Q2 are connected so that they operate as a regenerative switching

circuit having a characteristic similar to a *Schmitt circuit*. The collectors of both these transistors are coupled through emitter followers Q3 and Q6 to power transistors Q4 and Q5. When the voltage level at Q1 exceeds a preset value determined by the zener diode D5, both Q1 and Q2 switch to full conduction and the collector of Q2 goes positive. This positive voltage causes the power transistor Q6 to conduct and operate the magnetic latching relay whose contacts connect the load to the battery packs. This has a set of contacts which switch between two different voltage divider circuits at the input to Q1. When the load is connected to the batteries, the voltage divider keeps Q1 in saturation until the battery voltage drops below a lower limit determined by the setting of R7. When this lower limit is reached, Q1 and Q2 switch off and the collector of Q1 goes positive causing Q4 to conduct and latch the relay over to remove the battery load.

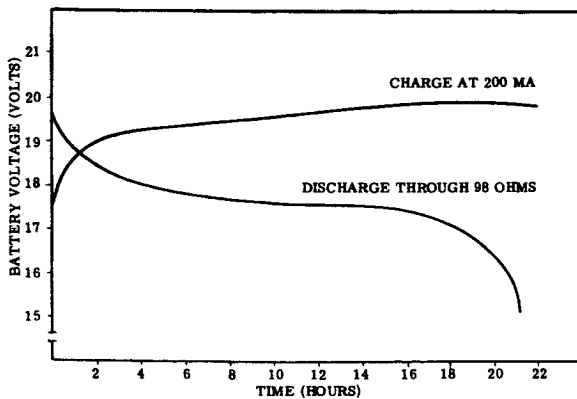


FIGURE 4-53.—Typical room temperature battery charge/discharge curves.

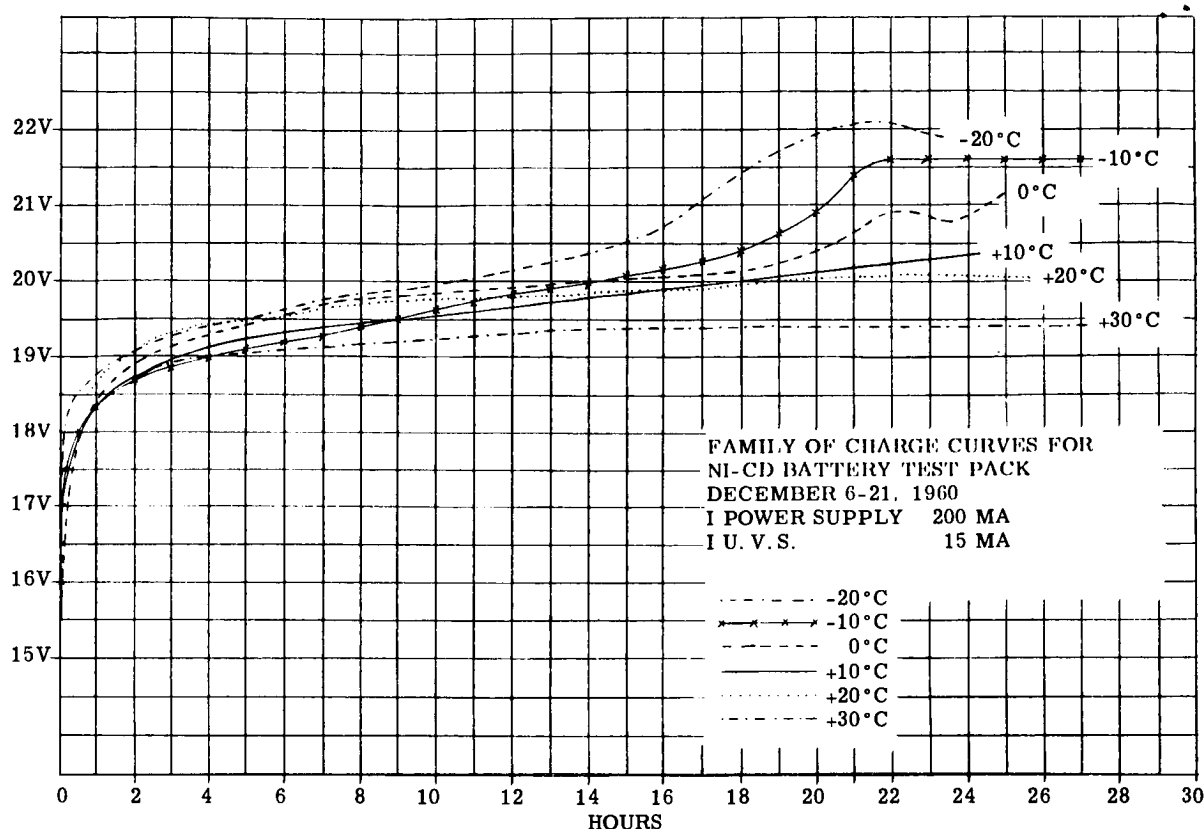


FIGURE 4-54.—Family of charge curves for NI-CD battery test pack.

The load is not reconnected until the battery voltage increases to a value sufficient to enable the second voltage divider to overcome the bias provided by D5 and cause Q1 to conduct. The upper battery voltage sufficient to cause conduction of Q1 is set by adjustment of R3.

From the explanation of the circuit operation, it can be seen that this circuit disconnected the battery load when the voltage dropped below a level determined by R7 and reconnected the load when the voltage increased to a level determined by the setting of R3. A set of contacts on the relay was used to disconnect the relay coil last energized to conserve power.

Figure 4-57 shows the effect of temperature on the upper and lower switching points of the undervoltage switch. These measurements were taken with the upper switching point set for 18.5 volts at room temperature. Power consumption of the circuit board, which includes

the temperature monitors, was 27 ma at 18 volts, or 49 milliwatts.

Monitor Circuits

The following section describes the operation of the spacecraft monitoring circuits.

Temperature Monitor

The temperatures of various points throughout the spacecraft were measured by the temperature monitor circuits. The temperature monitors furnished an output of 0 to 5 volts, covering the temperature range of -40°C to $+80^{\circ}\text{C}$, with the exception of the solar-cell sail temperature monitors. The solar-cell sail temperature was monitored by two circuits; one circuit furnished an output of 0 to 5 volts covering the range of -70°C to $+20^{\circ}\text{C}$, and the other circuit output varied from 0 to 5 volts covering the range of $+10^{\circ}\text{C}$ to $+120^{\circ}\text{C}$.

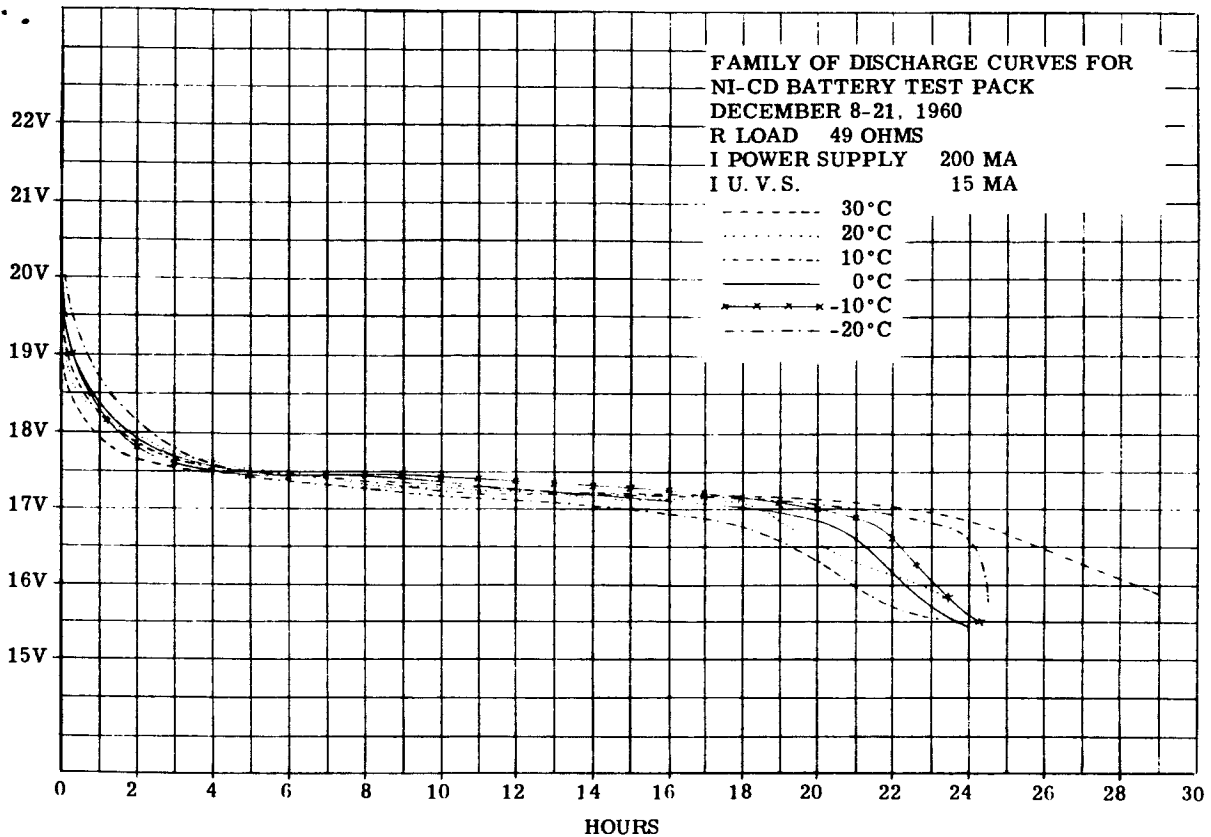


FIGURE 4-55.—Family of discharge curves for NI-CD battery test pack.

Bead thermistors were used as the temperature sensitive elements. These devices have a resistance-temperature characteristic, which is closely approximated by the following equation:

$$R(T) = \frac{R(T_0)}{e^{\beta/T_0}} e^{\beta/T} \quad (4.33)$$

where

$R(T)$ is the resistance at absolute temperature, T

$R(T_0)$ is the resistance at absolute temperature, T_0

β is a constant depending on the type of thermistor

Figure 4-58 shows a plot of the resistance characteristics of a Fenwall Type GA51L2 bead thermistor. This unit has a resistance of 100 kilohms at room temperature.

In order to obtain a linear temperature voltage characteristic, the temperature monitor

circuit had to have a resistance voltage characteristic which was the same as the resistance temperature characteristic of the thermistor. This requirement (shown in Figure 4-59) of the monitor circuit was nearly achieved by the circuit shown in Figure 4-60.

Figure 4-61 is a calibration curve for a typical temperature-monitor circuit. These monitors have proved to be stable in calibration and will repeat the calibration curves within ± 2 percent. Figure 4-62 is a spacecraft schematic showing some temperature monitors.

Battery Voltage Monitor

The battery voltage monitor circuit was located on the undervoltage switchboard. The purpose of this circuit was to supply an output which varied from 0 to 5 volts as the battery voltage varied from about 14 to 20 volts. The region of 14 to 20 volts on the battery charge-discharge curves represented the region which

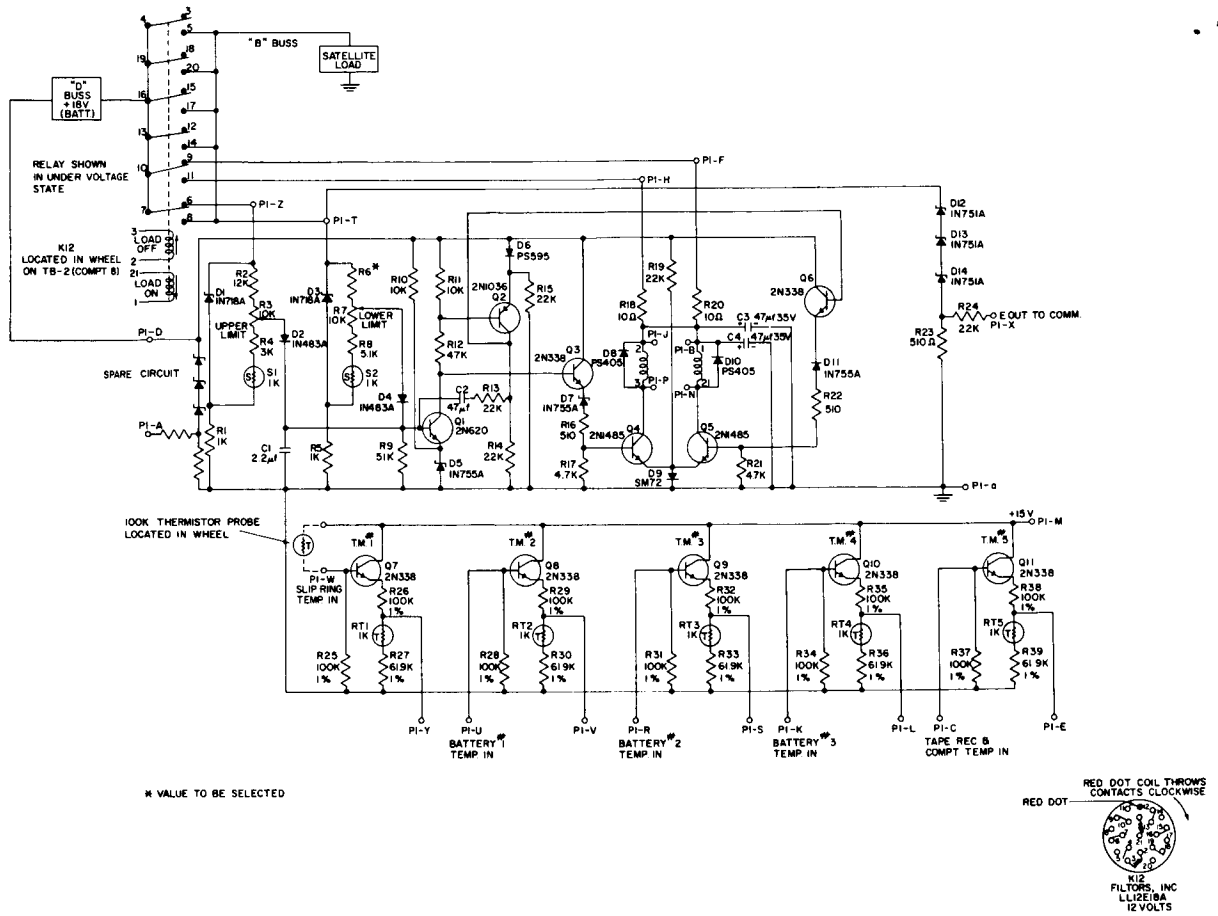


FIGURE 4-56.—Undervoltage switch and temperature monitor schematic.

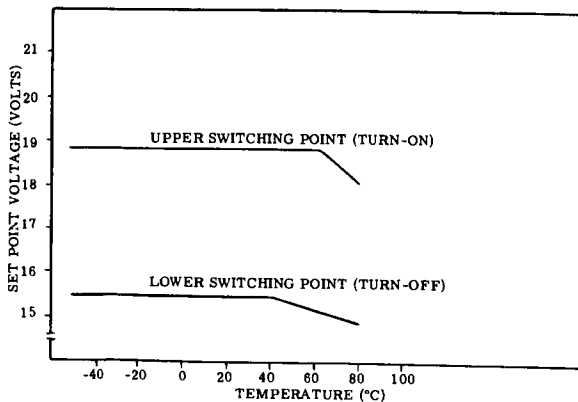


FIGURE 4-57.—Temperature dependence of the switching points of the undervoltage switch.

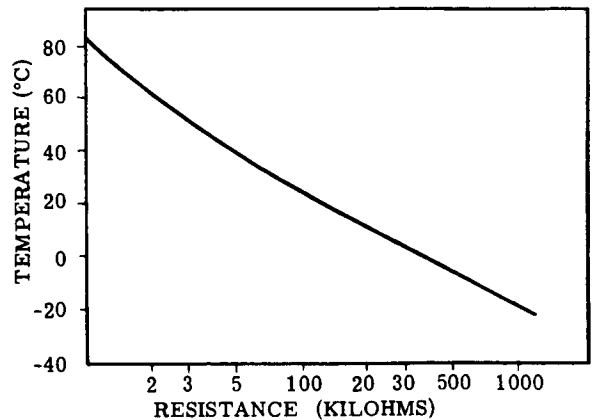


FIGURE 4-58.—Fenwall type GA51LZ bead thermistor resistor temperature characteristics.

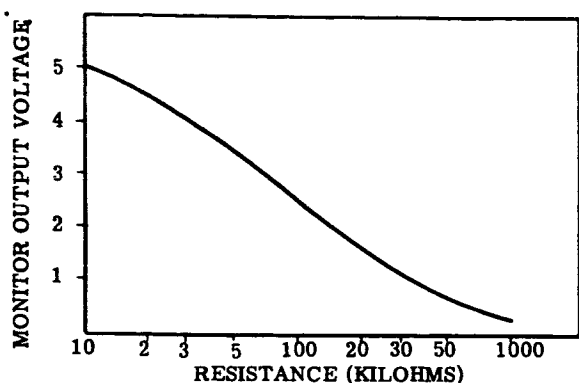


FIGURE 4-59.—Temperature monitor circuit R_1 versus E_o characteristics.

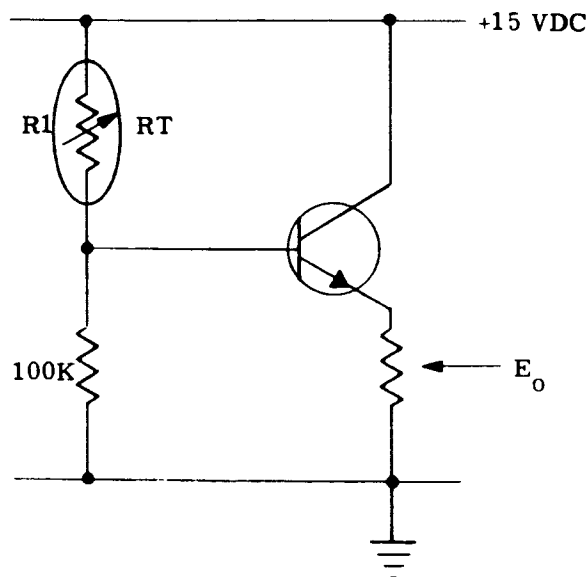


FIGURE 4-60.—Typical temperature monitor circuit.

could give some information on battery state of charge. Calibration curves for a typical circuit are shown in Figure 4-63.

Pitch Angle Monitor

The pitch angle of the spacecraft, defined as the angle between the solar vector and a normal to the spacecraft spin axis, was monitored by means of a solar detector mounted on the sail structure. The monitor had a half-opaque circular aperture through which sunlight was allowed to fall on a silicon cell. As the pitch angle changed, the edge of the opaque half of the entrance aperture shadowed more or less of the cell. This gives a current output

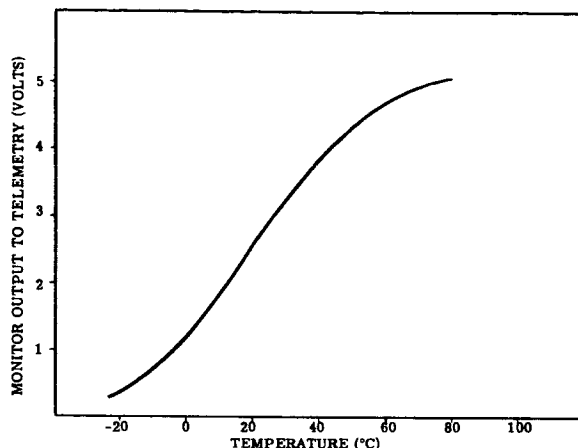


FIGURE 4-61.—Calibration curve for a typical temperature monitor circuit.

proportional to the pitch angle. The detector output current was fed into an amplifier identical to the readout amplifiers previously described (Figure 4-64) except that no emitter resistor was used. Thus, the dark signal gave essentially zero volts output and a fully lit signal produced 5 volts output. The 0 to 5 volt swing represented -5 to $+5$ degrees pitch angle where 2.5 volts is 0 pitch angle.

Spin Rate Monitor

The spin rate of the wheel was monitored by telemetering the timing ramp from the spin control circuit. The peak value of this ramp was proportional to spin period. An emitter-follower impedance-matching amplifier was necessary to isolate the spin control from the telemetry system. Figure 4-65 shows the calibration curve taken on the flight model monitor.

Servo Positioning Accuracy Monitors

In order to determine the elevation and azimuth pointing accuracy of the servo systems, two sets of fine eye detectors (readout eyes) were aligned to point parallel to the optical axis of the pointed experiments and hence boresighted to the control fine eye detectors. The control eyes were mounted on one pointed experiment and the readout eyes on the other. In this manner, information could be recorded to show any relative shift between the pointed

•

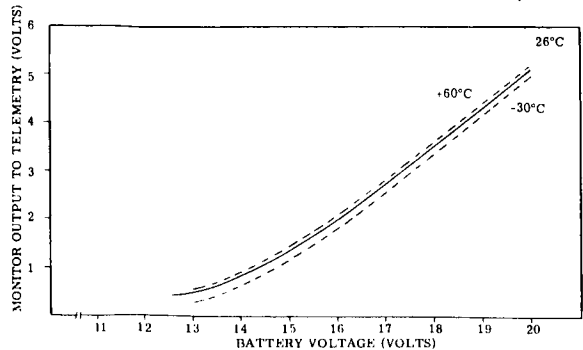


FIGURE 4-63.—Battery voltage monitor output temperature dependence.

The eye outputs were fed into the readout amplifiers, see Page 105. The amplifier outputs were centered around a 2.5-volt null with about 180 mv per minute of arc sensitivity. This allowed a readout of positioning accuracy to better than $\frac{1}{2}$ minute of arc.

The purpose of the OSO I launch sequence timer was to produce a timed sequence of events

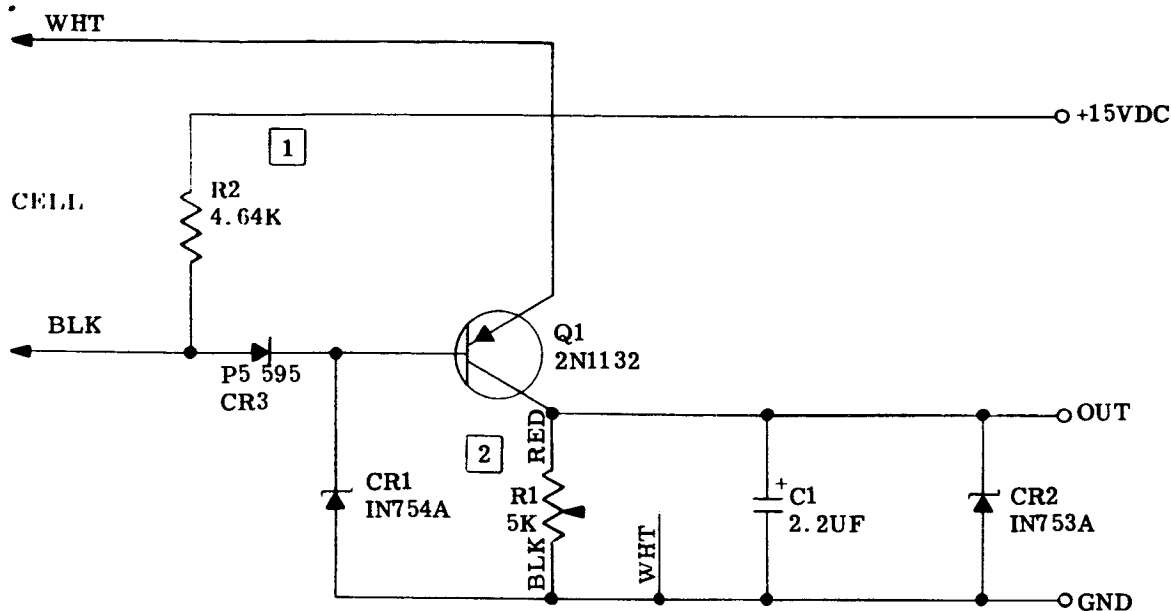


FIGURE 4-64.—Pitch readout amplifier.

after the third-stage spin-up. Figure 4-66 shows the block schematic diagram of this unit. The timer was started by an acceleration switch which was closed by the radial acceleration produced by the third-stage spin-up. The following outputs were programmed: 100 seconds after timer start, a ground closure was supplied to a relay which fired the arm release squibs; at 200 seconds, a ground closure operated the despin relay which started the satellite despin sequence, and a ground closure operated

relays which turned *orbit power* on and the redundant tape recorder off; at 400 seconds, a ground closure operated the relays which fired the azimuth release squibs and the nutation damper release squib; at 800 seconds, a pulse of

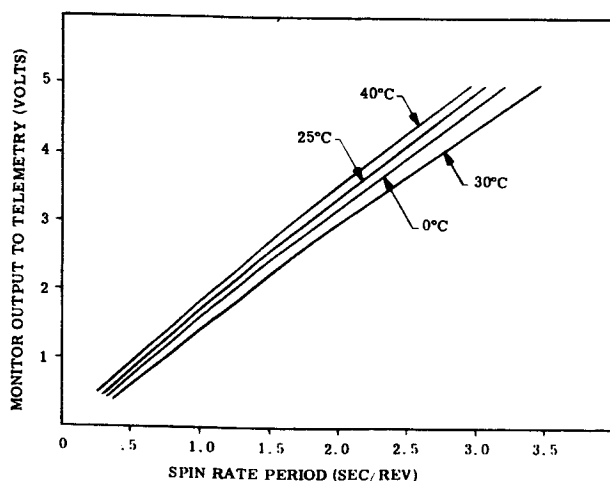


FIGURE 4-65.—Spin rate period (sec/rev).

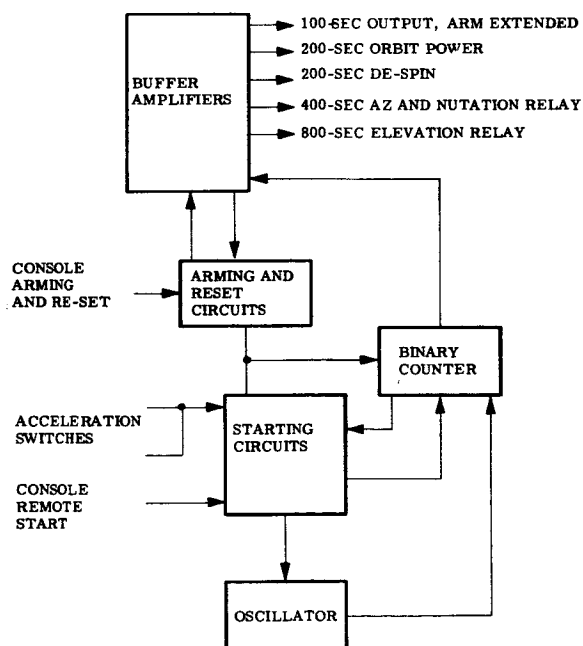


FIGURE 4-66.—Launch sequence timer block diagram.

+18 volts was provided to operate the relay which fired the elevation release squib. In addition, all the previous outputs of the timer were repeated at 800 seconds. Immediately after supplying the 800-second output, the timer turned itself off. Also, to assure reliable sequence, a completely redundant timer was used in parallel. The timer consisted of an 8-stage binary divider, a unijunction oscillator, buffer amplifiers, a 15-volt regulator, and starting and stopping logic circuits. Figure 4-67 shows the schematic diagrams of the timer, Figure 4-68 shows the printed board wiring of the timer, and Figure 4-69 shows the assembled timer package.

The timer was started by an acceleration switch closure caused by the radial acceleration due to the third-stage spin-up. Either of two redundant acceleration switches could start both of the timers. There were two requirements on timer starting: (1) the timers must not start the timing sequence by intermittent closures of the acceleration switches due to shock and vibration during launch, and (2) the timers must have a reset signal present when armed and ready to be started by the acceleration switch. The requirements on starting the timers were met as follows.

When either acceleration switch closed, it operated a double-pole, double-throw relay. One set of contacts was used to apply +15 volt power to the two timer unijunction oscillators. The other set of contacts was used to energize a second relay whose contacts were used to ground the divider reset lines to maintain a positive reset on the counters. The unijunction oscillators supplied pulses with periods of 6.25 seconds. If one acceleration switch remained closed for 12.5 seconds, an output was obtained from the second stage of the counters. This output was used to operate a DPDT latching relay: One set of contacts were used to hold +15 volt power on the unijunction oscillators, and the other set of contacts were used to hold the reset relays energized. If the acceleration switch did not remain closed for 12.5 seconds, power was removed from the oscillators and reset relays when the acceleration switch opened. The counters were also reset to zero.

The timer outputs were derived from multivibrator stages. A buffer amplifier was used to isolate the outputs from the binary divider. An RC coupling network having a time constant of about 2.5 seconds was used at the input to the buffers so that the outputs were negative pulses lasting from 2 to 3 seconds. The outputs of these buffers were either used directly or were used to operate relays which in turn supplied outputs.

At the last timer output (800 seconds after timer start) an 800-second output was supplied and all of the previous outputs were repeated. This was done to increase reliability, i.e., in case one of these outputs had not performed its function, a second redundant output pulse might then have produced the desired result. After the completion of the 800-second outputs, the timers shut themselves off and could not be started again unless a console reset and arming signal was applied followed by the proper acceleration switch closure.

For ground testing, a remote starting input was used. This input latched the starting circuits into the running mode.

Commutator

The OSO I utilized an 8-channel FM telemetry system. One channel of the data system was used for spacecraft systems data monitoring. In order to monitor the many points of interest (52), it was necessary to commutate the information into the data system. This was accomplished by two similar commutators whose outputs were sequenced into the data channel.

The requirements imposed on commutator design were: (1) the power drain had to be low; (2) offset voltages had to be low (1 millivolt or less); and (3) the hardware procurement had to be compatible with the OSO I development schedule. The relay commutator was selected over alternate methods such as solid-state switching because it best fit the overall requirements. The final design had a sampling rate of 0.5 to 1 sec which was compatible with relay operation, and it had a total power drain of 137 milliwatts. The contact switching also guaranteed near zero dc offset voltages.

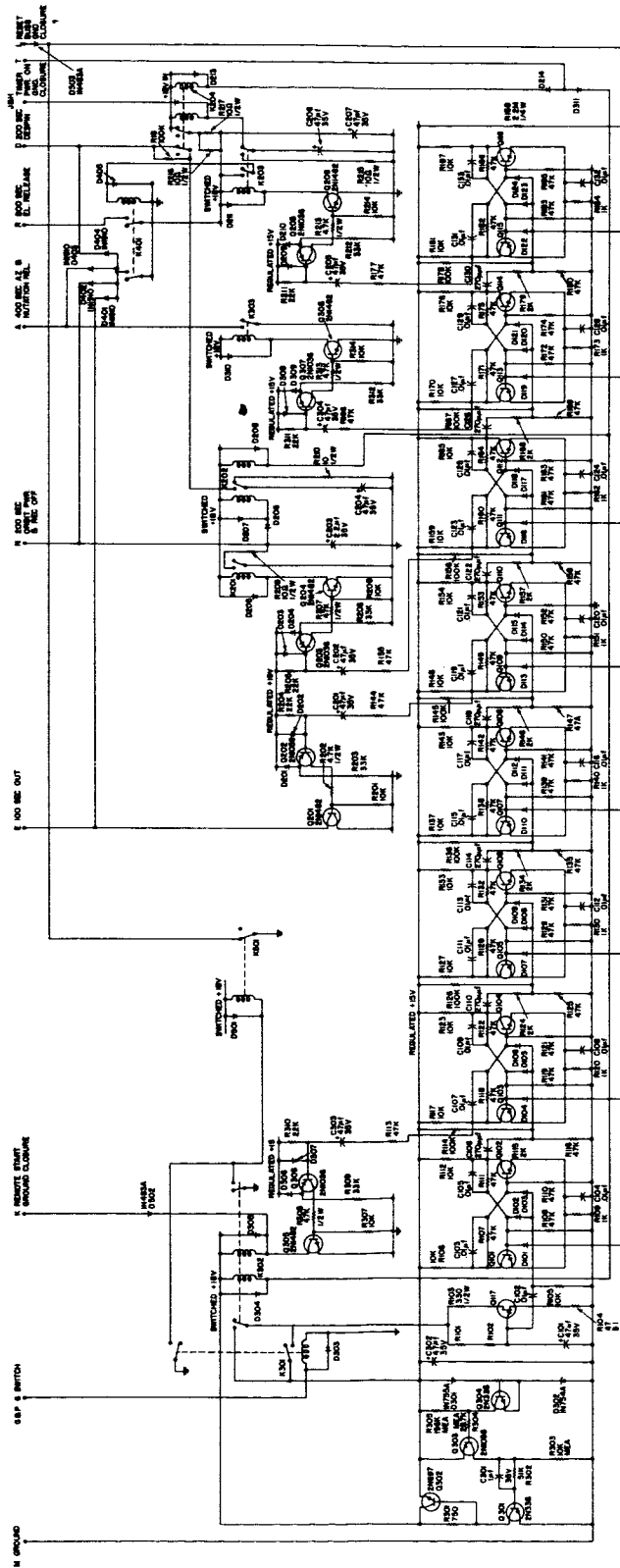


FIGURE 4-67.—Timer schematic.

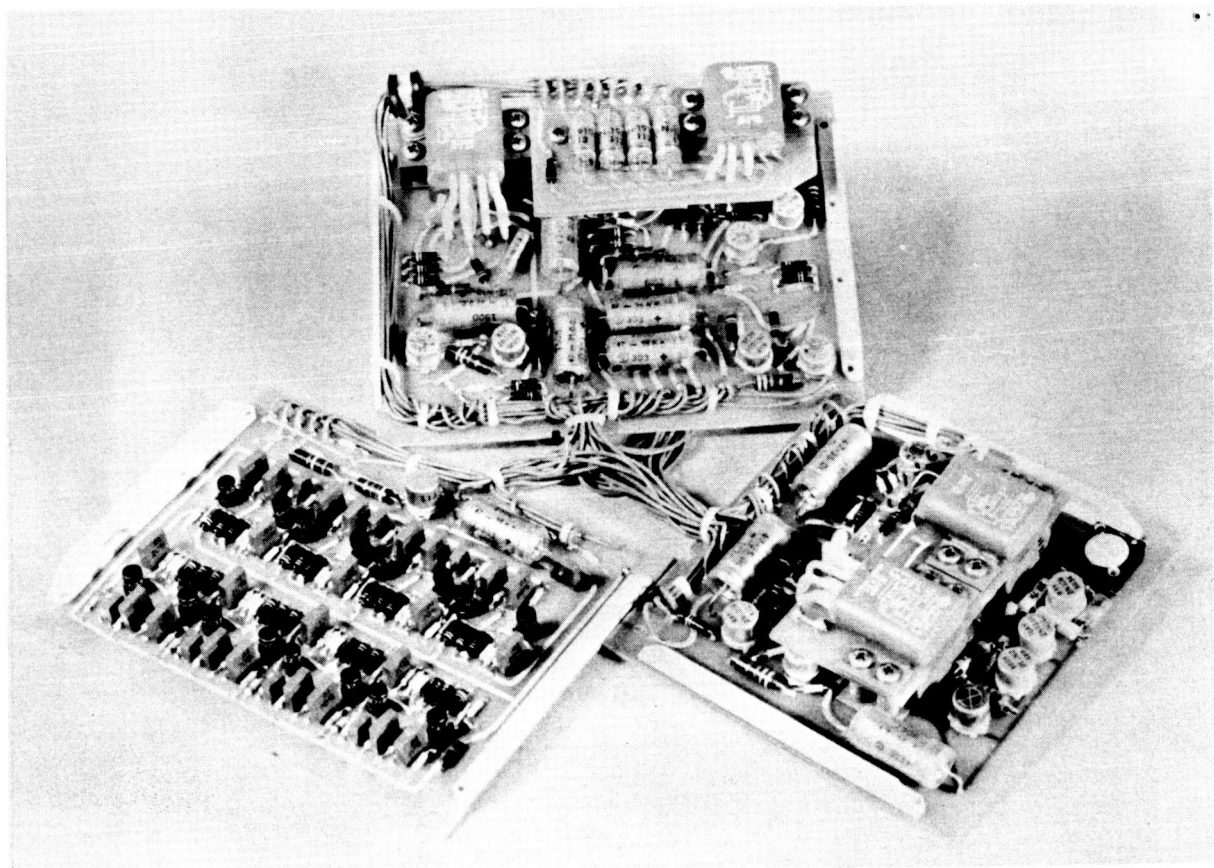


FIGURE 4-68.—Launch sequence timer circuitry.

The OSO I commutator was mechanized as a ring counter in which magnetic latching relays were the active elements. One pole of a relay was in the ring counter circuit and the other was used for the commutated signal.

The commutator, which was driven by a unijunction clock, had a feature such that if certain of the components had failed, the commutator would have reset itself to channel 1. The commutator would thus continue to operate after a failure between channel 1 and the channel with the bad component.

The commutator package also contained a voltage divider from the regulated 15-volt supply that was used to calibrate the telemetry channel and a circuit to measure the 15v and 18v supplies.

The commutator had one DPDT latching relay for each commutated channel. One pole of each relay was connected to the commutator

output (see Figure 4-70). The channels to be commutated were connected to the contacts of this set. The other set of contacts were used to control commutation. When channel 1 was on, 18 volts was applied to the OFF coil of channel 1, and the ON coil of channel 2. When the clock circuit grounded the return line of the coils (all channels connected in parallel), channel 1 would throw to OFF and channel 2 to ON. This action was assured by the charge stored in C9 as follows: C9 charged in less time than the activation time of the relay. Therefore, if channel 1 relay activated before channel 2 relay had turned on, the charge in the capacitor across the channel 2 ON coil would have thrown the relay on.

By this action, channel 2 came on. Also, power was supplied to the high side of the OFF coil of channel 2 and the ON coil of channel 3. These coils activated when the next ground

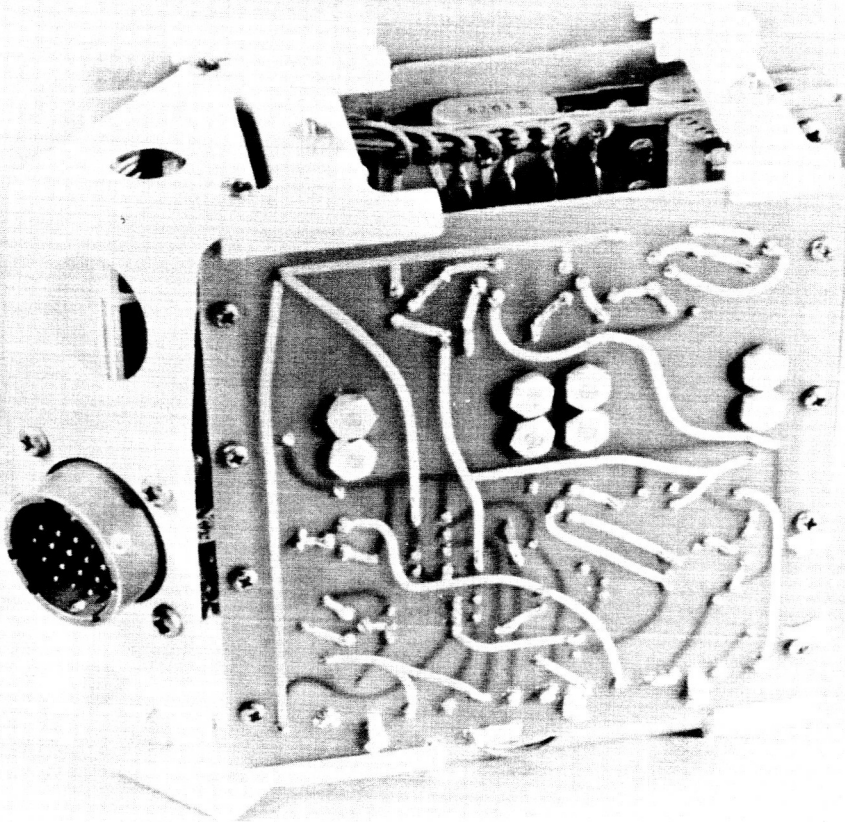


FIGURE 4-69.—Assembled timer package.

closure occurred on the coil return line. This process repeated until the last channel was on, at which time voltage was applied to the OFF coil of the last channel and through a diode to the ON coil of the first channel. When the next grounding of the coil return line occurred the last channel turned off and the first channel turned on again. This completed the cycle.

The clock timing was accomplished with a unijunction transistor in conjunction with timing elements R1, R2 and C2. C2 had a positive temperature coefficient which was partially compensated for by the reduction with temperature of the trigger voltage of the unijunction. The uncompensated drift was about 6 percent over the temperature range of -20°C to $+40^{\circ}\text{C}$. This drift was not serious because the only requirement on timing was that each reading have sufficient duration to establish a good average in the presence of telemetry noise

and yet short enough so that each channel can be sampled often enough.

The unijunction drives a filter circuit used to keep stray, high-frequency pulses from the gate of the controlled switch, Q3. When the unijunction fired, the pulse from its base fired the controlled switch. The switch clamped the relay coil return line to ground as long as the anode current remained above about $0.1\ \mu\text{a}$. When the previously ON relay was turning off, power was interrupted to the relay coil return line for an interval determined by the relay transfer time, about 2 ms. The only current through the controlled switch during this time was from the discharge of stray capacitance. Thus, the controlled switch would then open until the next trigger pulse. The resistor, R7, was a current limiter.

The reset circuit protected against malfunctions (that resulted in removal of $+18\text{v}$ from



FIGURE 4-70.—Communications schematic.

the reset line) by recycling the commutator to channel 1. The circuit protected against the following problems.

1. An ON channel goes off, but the succeeding channel does not go on.
2. A relay that gets *hung up* between contacts.

The above conditions were sensed by the reset line. When voltage to the reset line was removed, the voltage across C6 decayed. When the voltage dropped below the zener voltage of CR14, Q7 was no longer held clamped, and base drive was provided to Q5, which energized the ON coil of channel 1. At the time of the next clock pulse, power would flow through the ON coil of channel 1 on. This broke the circuit to the +18v side of the reset circuit, and thus the controlled switch dropped out and the commutator proceeded normally. CR13 prevented a base-emitter breakdown in Q5 and Q6 when the commutator was working normally; i.e., there was +18v on the cathode of CR13 and ground at the base of Q6.

The voltage monitors were mechanized with zener stacks which were chosen to provide about 2.5v out at normal voltage in. The zener diodes used also had a low temperature

coefficient. Since a 2 percent, or 50 mv, change in the normal telemetry output voltage represents 50 mv, or 0.33 percent, of the 15v regulated line, or 0.28 percent of the +18v line, the supply voltages could be read quite accurately.

Calibration signals were provided by tapping down the 15v regulated line. (The 15-volt supply was essentially insensitive to temperature.) Generation of calibrate signals with zeners with a suitable voltage coefficient would have required a larger power dissipation. Also, since the 15v bus was read to a higher accuracy than the calibration voltages, a correction factor for both the calibrate signal and the 15v bus can be derived as follows:

$V_m=15\text{v}$ monitor output as read by telemetry

V_a = actual output voltage of 15v monitor

V = actual voltage of the 15v line

V_c = actual calibrate voltage

$$V_c = V/3$$

$$V_m = \frac{5}{V_c} V_a$$

$$V_a = V - 12.5$$

Therefore

$$V = \frac{12.5}{1 - V_m/15} \quad (4.34)$$

Thus, the above equation shows that the 15v regulated voltage can be obtained even when a portion of the 15v is used for a reference. By taking the derivative of the equation, the following is obtained:

$$\Delta V = 1.1 \Delta V_m$$

where V_m is 2.5v. Thus, the slope on the 15v monitor calibration curve used on OSO I should have its slope decreased by 1.1 since the calibration curve was obtained with an absolute reference instead of the 5v calibrate as a reference.

The filter circuit was used to keep current pulses drawn while charging the 22 μ f capacitors from putting large voltage transients on the +18v line. Since the controlled switch turned on in one microsecond, the current in the inductor was small. Therefore, C5 had to charge the 22 μ f capacitors and still maintain an appreciable voltage. A value of 550 μ f was chosen for C5 so that the drop would be less than 10 percent. This assured that the 22 μ f capacitors would be at full charge at the time of contact transfer.

Squib Firing Circuits

During pre-launch and launch activities at AFMTC, many very high-power output radar transmitters and high-power RF transmitters were in operation. Any parallel wire cabling was thus subject to RF power pickup of high current generating potential. In order to assure that no such current generation was possible, critical leads had to be properly shielded and twisted. Furthermore, the shielding had to be properly terminated.

The OSO I utilized squib-actuated pin puller devices to retain and then release several mechanisms; namely, the extendable arms, the azimuth and elevation pointed instrument gimbals, and the nutation damper bob.

To avoid the possibility of accidental firing of the squibs, extensive shielding and grounding techniques were used as shown in the schematic

of Figure 4-71. The bridge wires were held shorted by the OFF contacts of the firing relay. The squib leads were shielded up to the squib connector, and the shield was grounded both at the squib connector and at the relay end of the cable.

GAS CONTROL SYSTEMS

This section describes the pneumatic systems which provided torques by discharging gas through nozzles.

Torques from the gas control system had to be applied to the spacecraft for three purposes:

1. *Despin*: The spin-stabilized third stage of the launch vehicle detached, leaving the the spacecraft spinning at about 1.5 revolutions per second. It had to be slowed to a nominal spin rate of about 0.5 rps.

2. *Spin-up*: The spacecraft was slowed by aerodynamic and electromagnetic drag. It then had to be spun up to a nominal spin rate of about 0.5 rps.

3. *Pitch Control*: For coarse elevation pointing, torques were required to precess the spacecraft about the elevation or *pitch* axis.

Description of Gas Systems

Despin and Spin-Up System

This was a dual purpose system, i.e., to reduce the spacecraft spin rate from its high initial value at injection to the nominal operating value of about 0.5 rps, and to maintain the spin rate near this nominal value.

Figure 4-72 is a simplified schematic diagram of the despin and spin-up system. The locations of the components in the spacecraft are shown in Figure 4-73. Nitrogen gas was stored at an initial pressure of 3000 psi in three fiber-glass bottles. The gas was piped through a filter to a pressure regulator, and then to two solenoid valves. When one of these valves was open, gas discharged through three despin nozzles. When the other valve was open, gas discharged through three spin-up nozzles. The pressure at the nozzle inlet was 60 psia. Each nozzle produced a thrust of 0.10 pounds force on a 3.97-foot lever arm for a total torque of 1.19 pound-feet.

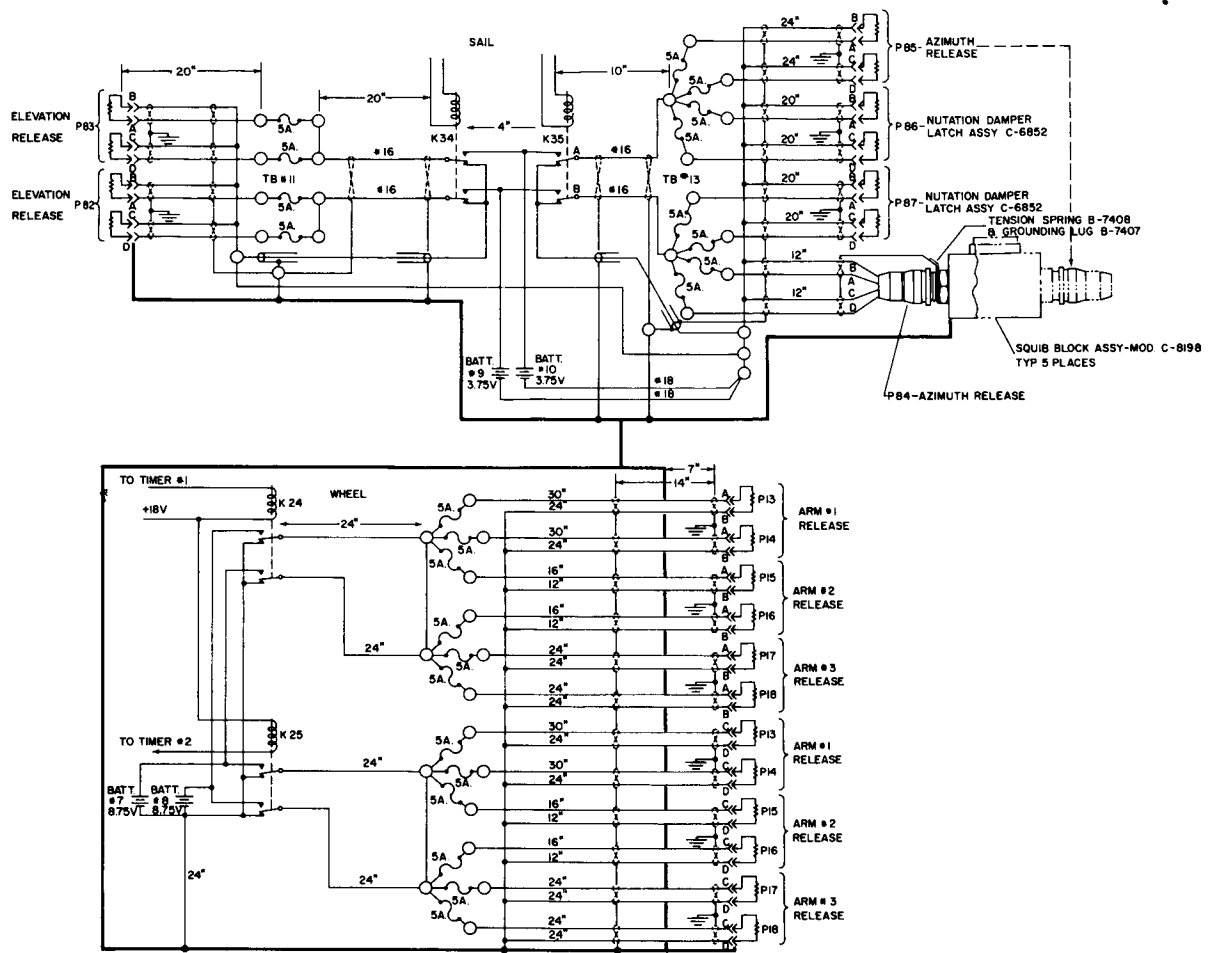


FIGURE 4-71.—Squib firing circuitry.

The electronic circuit for despin is described on page 143. This circuit measured the spin rate of the spacecraft, and whenever the spin rate dropped to a preset level a little below nominal, the circuit opened the spin-up nozzle for four seconds. Since the daytime spin moment of inertia was about 23.5 slug-ft², one burst of the spin-up system increased the daytime spin rate by 0.032 rps. Thus, the spin rate was held close to the nominal value.

The despin valve was used only once, just after separation of the spacecraft from the spin-stabilized launch vehicle. The valve was opened by the launch sequence timer. In about 140 seconds the spin rate dropped below its nominal value. Then, when the electronic circuit switched the spin-up valve on, it switched

the despin valve off permanently by a latching relay.

Pitch Control System

The purpose of this system was to maintain pitch attitude so that the spacecraft spin axis remained within 3 degrees of perpendicular to the solar vector.

Figure 4-74 is a schematic diagram of the pitch control system. The locations of the components in the spacecraft are shown in Figure 4-73. The nitrogen gas, stored at an initial pressure of 3000 psi, was piped through a filter to a pressure regulator which dropped the pressure to 30 psia, and then to two solenoid valves. When one valve was open, gas discharged through a nozzle which aimed up from

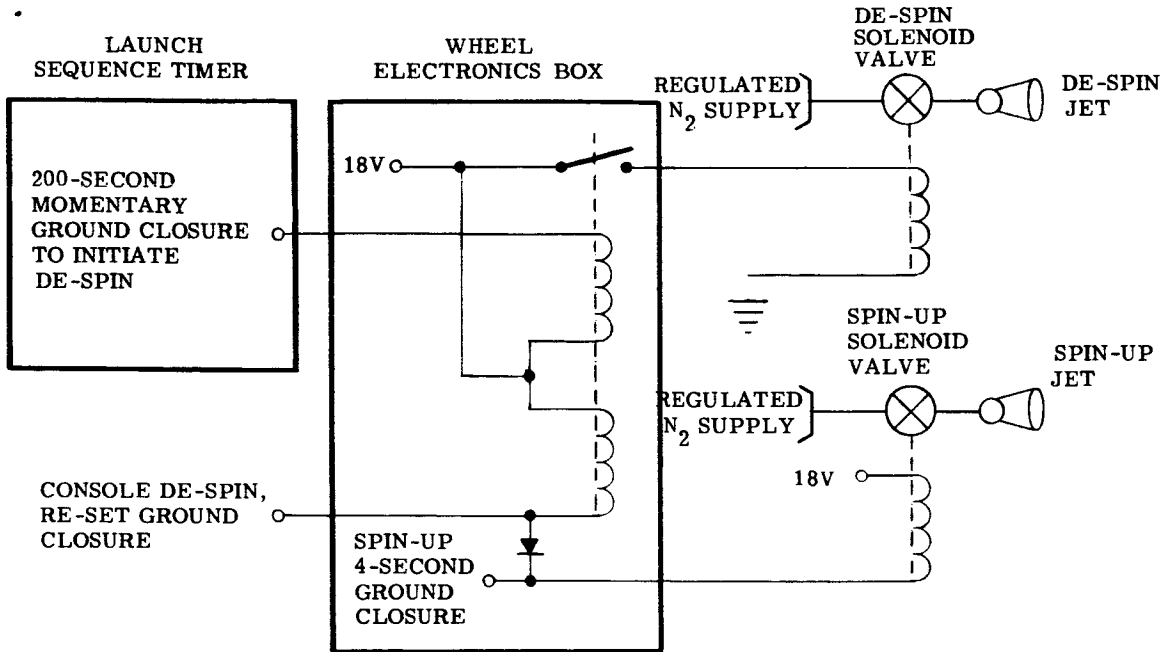


FIGURE 4-72.—Despin—spin-up schematic.

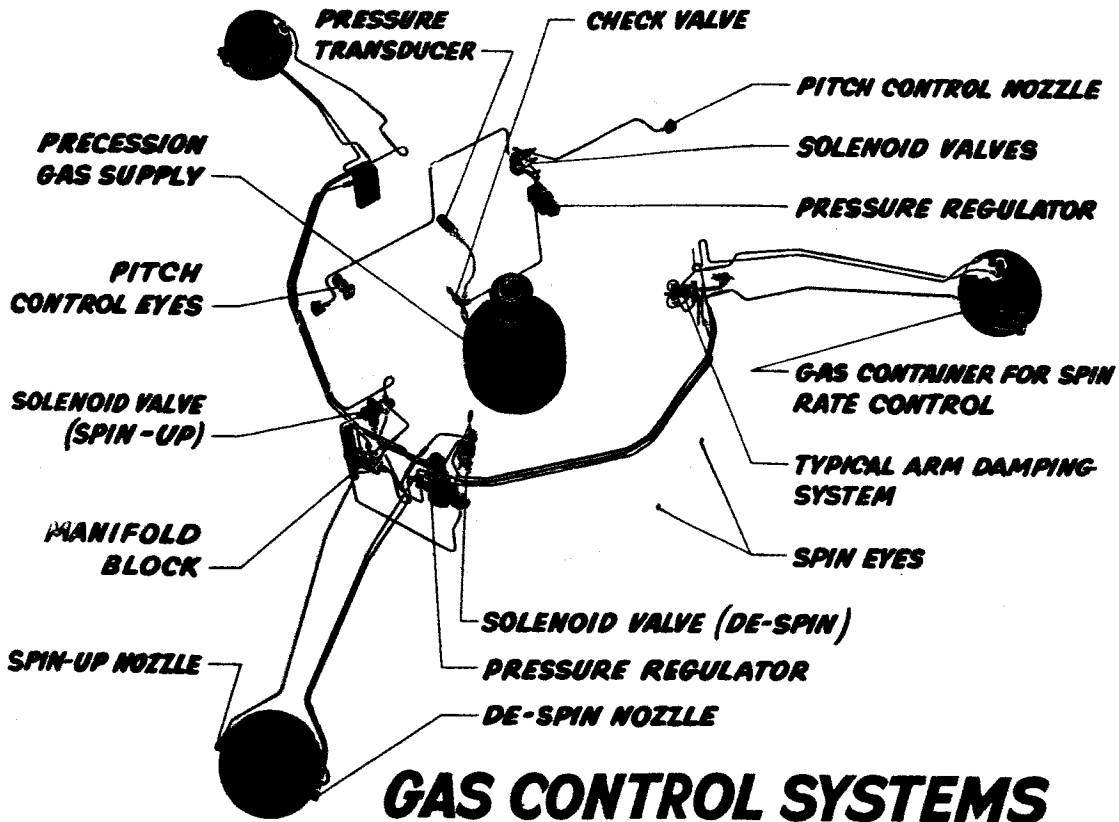


FIGURE 4-73.—Gas control systems.

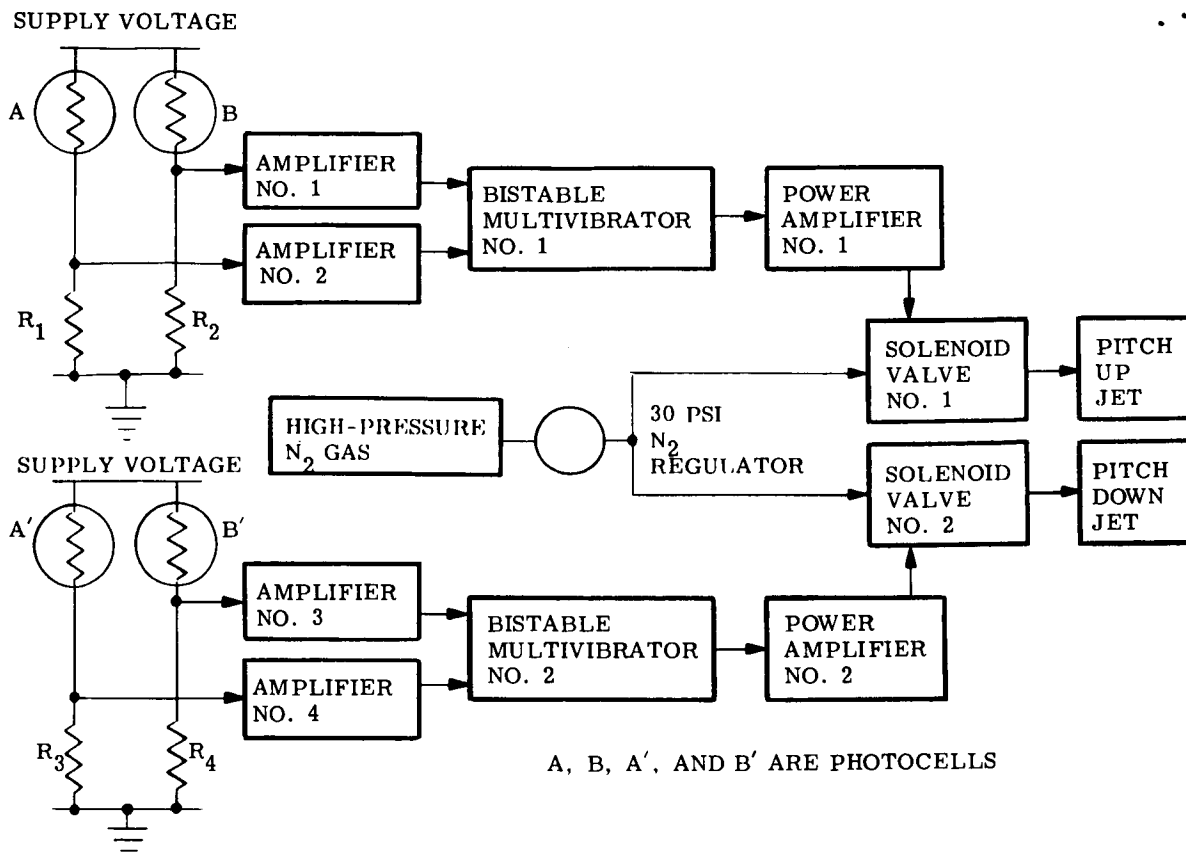


FIGURE 4-74.—Automatic pitch angle control block diagram.

the right side of the sail, producing a torque directed toward the sun. This caused the spinning wheel to precess in the *pitch down* direction. When the other valve was open, gas discharged through a nozzle aimed up from the left side of the sail producing a torque directed away from the sun. This caused the spacecraft to *pitch up*.

The electronic circuit which controlled the valves is described on page 126. Its effect was to open the appropriate valve when the pitch attitude deviated from the solar vector by more than three degrees, and to keep the valve open until the pitch attitude deviated in the opposite direction by about one degree. (This overshoot was provided for one of the experiments.)

The nozzle thrust was 0.05 lb, and the lever arm was 1.93 ft, so the torque was 0.0965 lb-ft. The angular momentum of the wheel with a moment of inertia of 23.5 slug-ft² spinning at

0.5 rps was 73.8 lb-ft-sec. This resulted in a pitch rate of $\frac{0.0965}{73.8}$ rad sec⁻¹, or 0.075 deg sec⁻¹, thus each pitch attitude correction took about 53 seconds.

Basic Design Decisions

This section presents a discussion of decision from which most of the details of the system design were derived.

Mass Ejection

The requirement here was to change the angular momentum of the spacecraft. This could have been accomplished in only two ways: By reacting against some external object (such as the earth's magnetic field), or by storing part of the spacecraft angular momentum in some internal object (such as a spinning disk or a high-velocity gas stream), and then throwing

this object away. Very early in the program the decision was made to use this *mass ejection* method. A gas ejection system is most practical because of the ease of getting high velocity at a large radius, and thus a high angular momentum per pound of ejected mass.

Propellant Selection

The gas which was ejected from the nozzles had to have as little effect as possible on the sensitive optical surfaces in experimental instruments. This consideration, and the desire for system simplicity, dictated the selection of a passive propellant.

If the propellant had been stored as a liquid rather than as a gas, less tank weight and volume would have been required. However, this would have introduced serious balance and phase separation problems. Therefore, it was decided to store the propellant as a gas. To simplify development, a gas known not to condense when passed through pressure regulators, valves, and nozzles was chosen. The propellant therefore had to have a critical temperature well below the expected storage temperature or a critical pressure well above the storage pressure. These considerations limited the choices to nitric oxide, oxygen, carbon monoxide, nitrogen, hydrogen, helium, argon, krypton, and neon.

An important criterion for the selection of a specific propellant is the impulse which can be obtained from a given weight of gas-filled storage bottle. The storage capacity of a spherical fiberglass bottle of the type used is about $1.4 (10^5)$ psi (in³) pound weight of bottle. Now, at 460° R, 1 psi (in³) of an ideal gas weighs $1.095 (10^{-7})$ *MW* pounds, where *MW* is the molecular weight of the gas; so to store 1 pound of gas we need $65.3/MW$ pounds of bottle. Specific impulse, I_s , is defined as the impulse obtained from 1 lb weight of propellant. Then

$$I_T = \frac{I_s}{1 + \frac{65.3}{MW}} = \frac{I_s MW}{MW + 65.3} \quad (4.36)$$

where I_T is the impulse per pound of (gas + bottle).

For the nine gases listed above, we may calculate the following values for I_T :

Gas	I_s	I_T Pound (force) sec/pound (weight)
H ₂ -----	274	8.1
He-----	167	9.6
Ne-----	74	17.5
A-----	52	19.8
Kr-----	36	20.2
CO-----	74	22.3
N ₂ -----	74	22.3
NO-----	72	22.6
O ₂ -----	70	23.0

Of the four gases with the highest I_T , nitrogen was chosen because of inertness.

Quantity of Propellant

Despin and Spin-Up System—When the spacecraft was placed in orbit, it had a moment of inertia of 19.7 slug-ft² and was spinning at about 3 rps. Thus, the initial angular momentum was 247 slug-ft² sec⁻¹. After despin, the moment of inertia was 24.4 slug-ft² and the spin rate was about 0.5 rps, so the angular momentum was 77 slug-ft² sec⁻¹. This change of 170 slug-ft² sec⁻¹ in angular momentum was caused by the despin gas jets, which acted at a radius of 3.97 ft. This requires a jet impulse of 43 lb sec. With nozzle efficiency of 90 percent giving an effective specific impulse of 67 lb sec/lb weight, despin consumed 0.64 lb of gas.

To estimate the amount of gas required for spin-up it would be necessary to calculate the angular impulse delivered to the spacecraft over its six-month life by eddy current damping and by aerodynamic drag. Neither of these impulses can be calculated, so it was arbitrarily decided to provide enough gas to give an angular impulse of 1000 slug-ft² sec⁻¹, or 3.76 lb of gas. This would have been enough gas even if the drag were great enough to reduce the spin rate by 7 percent per day. (The actual spin rate decay, measured on the orbiting spacecraft, was 0.2 percent per day.)

The despin and spin-up system carried 4.4 lb of nitrogen.

Pitch System—The pitch gas had to do two jobs. First, it had to pitch the spacecraft

enough to keep up with the change in direction of the solar vector due to the earth's orbital motion. Second, it had to buck out any external torques tending to pitch the spacecraft. In estimating the gas requirements, we assumed that these two jobs were independent.

The solar vector rotates 1 degree per day. If the spacecraft spin axis makes an angle θ with the ecliptic plane, the spacecraft must pitch $1^\circ \cos \theta$ per day to keep up with the solar vector. Assuming the worst case, the spacecraft would have had to pitch 180° during its six-month life. Since the angular momentum of the spacecraft was $77 \text{ slug-ft}^2 \text{ sec}^{-1}$, 180° pitch required an angular impulse of $77\pi = 242 \text{ lb-ft-sec}$. The pitch jets acted at a radius of 1.93 ft, and provided 67 lb-sec impulse per pound of gas, so 180° pitch required 1.87 pounds of gas.

A precise analysis of the external torques applied to the spacecraft would have been extremely costly; for this reason only a crude worst-case analysis was made. This analysis, which included the effects of gravity gradient, aerodynamic drag, radiation pressure, and current loops, showed that the angular impulse about the roll axis adds up to less than 270 lb-ft-sec during the six-month life of the spacecraft. To buck out this angular impulse, 2.09 pounds of gas were required.

The total pitch gas requirement then was 4 pounds. Since there was a little extra space available, the pitch system was designed to carry 4.4 pounds of nitrogen.

The actual gas usage was much less than anticipated since the actual pitch roll was about 4 degrees per week (about $\frac{1}{3}$ of that expected).

Thrust Level

Despin and Spin-Up—There were two obvious schemes, either of which could have been used to control spin rate. One was for a spin rate sensor to turn the jets on when the spin rate reached some low limit, and to turn the jets off when the spin rate reached some high limit. The other was for the spin rate sensor to turn the jets on for some fixed time interval whenever the spin rate reached the low limit. This second method was simpler and more

reliable electronically, and was the method used.

The valve-on time interval had to be long compared to the response times of various parts of the pneumatic system, yet short enough to be regulated by a simple RC network. Any interval between 2 and 6 seconds was considered usable.

Considering the various tolerances involved, if the spin rate had to be held within a 15-percent range, the spin rate increment had to be no more than 8 percent and no less than 4 percent of nominal rate. That is, the angular impulses had to be between 3 and 6 ft-lb-sec.

These considerations required that each spin-up nozzle provide a thrust between 0.04 and 0.25 lb. Using the same thrust on despin, between 60 and 350 seconds would have been required to be completed.

It was decided to use 0.10-lb thrust on both spin-up and despin nozzles.

Pitch—The pitch jet thrust level was set by the allowable nutation. It was necessary to reduce the nutation amplitude to near 1 minute of arc, or $2.91 (10^{-4})$ radians, to be certain that the nutation would not interfere with pointing accuracy or cause excessive control power usage.

As shown in chapter 2, the amplitude A of nutation caused by control torque L is

$$A = \frac{LI_t}{I^2 \omega^2} \quad (4.37)$$

where

I is the spin moment of inertia
 I_t is the transverse moment of inertia, and
 ω is the spin rate of the spacecraft.

For this particular spacecraft,

$$A = 3.01 (10^{-3}) L, \text{ or } L = 0.0967 \text{ ft-lb}$$

for 1-minute nutation amplitude. Since the pitch jets act at a radius of 1.93 ft, the jet thrust of 1-minute nutation amplitude is 0.05 lb. This was the thrust level used.

Storage Pressure

There was a variety of pneumatic components rated for 3000 psi service, but components rated

for higher pressure were rare. Therefore, to make the storage bottles as small as possible without introducing a component availability problem, the system was designed for 3000 psi storage pressure.

Nozzle Design

Approach to the Problem

The problem was to develop, quickly and cheaply, a nozzle which would produce 0.05 lb thrust with acceptable efficiency.

Although a great deal of work had been done on large rocket engine nozzles, the design rules developed for these nozzles could or could not have applied to a very small nozzle. (For instance, in large nozzle design, one assumes that the gas boundary layer is thin compared to the throat diameter. This assumption might have been invalid for a nozzle with a 0.040-inch throat.) Therefore, it was decided to use the theoretical equations and large-nozzle design techniques for guidance, but to put no confidence in anything except test results.

Symbols—In the nozzle design analysis, the following symbols were used:

- P_c =chamber pressure, i.e., pressure just upstream of the nozzle
- P_e =exhaust pressure, i.e., pressure at the nozzle exit
- P_a =atmospheric pressure surrounding the nozzle
- A_t =throat area
- A_e =exit area
- θ =half-angle of nozzle exit cone
- $\epsilon = \frac{A_e}{A_t}$ =expansion area ratio
- T_c =chamber temperature, i.e., gas temperature just upstream of the nozzle. Throughout this section, assume that $T_c = 529^\circ \text{R}$.
- K =ratio of specific heats of gas. For nitrogen, $K = 1.404$
- I_{sp} =specific impulse. For nitrogen at 529°R
- $I_{sp} = 79.4 \text{ lb sec/(lb weight)}$
- $\dot{m}$ =mass flow rate
- ϕ =nozzle efficiency, the actual effective specific impulse divided by the ideal specific impulse

F =thrust

R =gas constant. For nitrogen, $R = 1777 \text{ lb-ft/}^\circ \text{R}$, in the ft-slug-sec system

Basic Nozzle Equations—The behavior of gas in a De Laval nozzle is very complicated. However, a fair approximation can be made with the following simplifying assumptions:

1. The gas is ideal in the sense that $PV = RT$, where R is a constant
2. The gas is ideal in the sense that K is independent of temperature and pressure
3. Viscosity effects are negligible
4. The process is adiabatic
5. The gas velocity is constant across any cross-section taken normal to the nozzle axis
6. The velocity of the discharged gas is parallel to the nozzle axis
7. P_e is greater than P_a

With these assumptions, the following equations hold.

Mass flow rate:

$$\dot{m} = A_t P_c \sqrt{\left(\frac{K}{RT_c}\right) \left(\frac{2}{K+1}\right)^{\frac{K+1}{K-1}}} \quad (4.38)$$

Expansion area ratio:

$$\epsilon = \frac{\left(\frac{P_c}{P_e}\right)^{\frac{1}{K}} \left(\frac{2}{K+1}\right)^{\frac{1}{K-1}}}{\sqrt{\frac{K+1}{K-1} \left[1 - \left(\frac{P_e}{P_c}\right)^{\frac{K-1}{K}}\right]}} \quad (4.39)$$

Thrust:

$$\frac{F}{P_c A_t} = \sqrt{\frac{2K^2}{K-1} \left(\frac{2}{K+1}\right)^{\frac{K+1}{K-1}} \left[1 - \left(\frac{P_e}{P_c}\right)^{\frac{K-1}{K}}\right]} + \frac{P_e}{P_c} \epsilon - \frac{P_a}{P_c} \epsilon \quad (4.40)$$

Chamber Pressure—High chamber pressure required an inconveniently small nozzle throat. Low chamber pressure required a difficult pressure regulator design. A compromise was necessary. If

$$I_{sp} = 79.4 \text{ lb-sec/(lb weight)}$$

then 0.05-lb. thrust requires

$$\dot{m} = 1.96(10^{-5}) \text{ slug sec}^{-1}$$

From Equation (4.38), we have

$$A_t P_c = 2.76(10^{-2}) \text{ lb}$$

This means that

- if $P_c = 100$ psi, the throat diameter = 0.019 in.,
- if $P_c = 30$ psi, the throat diameter = 0.034 in.,
- if $P_c = 10$ psi, the throat diameter = 0.059 in.

A throat diameter less than 0.030 in. is difficult to control, so P_c should not be much more than 30 psi.

A simple single-stage pressure regulator can be designed for a pressure ratio of 100, but not for a much higher ratio. Since the gas storage

pressure was 3000 psi, P_c should not be much less than 30 psi. The value $P_c = 30$ psi was selected.

Expansion Area Ratio—Figure 4-75 shows a plot of relative nozzle efficiency as a function of ϵ , derived from Equations (4.39) and (4.40). This plot shows that although a large ϵ is desirable, there is not a great deal to be gained by increasing ϵ beyond 30 or 40.

If P_e is less than P_a , the behavior of the nozzle is unpredictable. In orbit, P_a is negligible, but during laboratory testing it is not. The testing was done in a vacuum chamber of $2(10^6)$ cm³ volume. The nozzle discharged about $2(10^{-5})$ slug sec⁻¹ of nitrogen, so the pressure in the vacuum chamber, P_a , increased at about $2(10^{-3})$ psi sec⁻¹. Figure 4-76 shows the time required for P_a to rise to P_e , as a function of ϵ . In order to allow 20 seconds for a measurement, it was necessary to make ϵ less than 31. It was decided to make ϵ about 31.

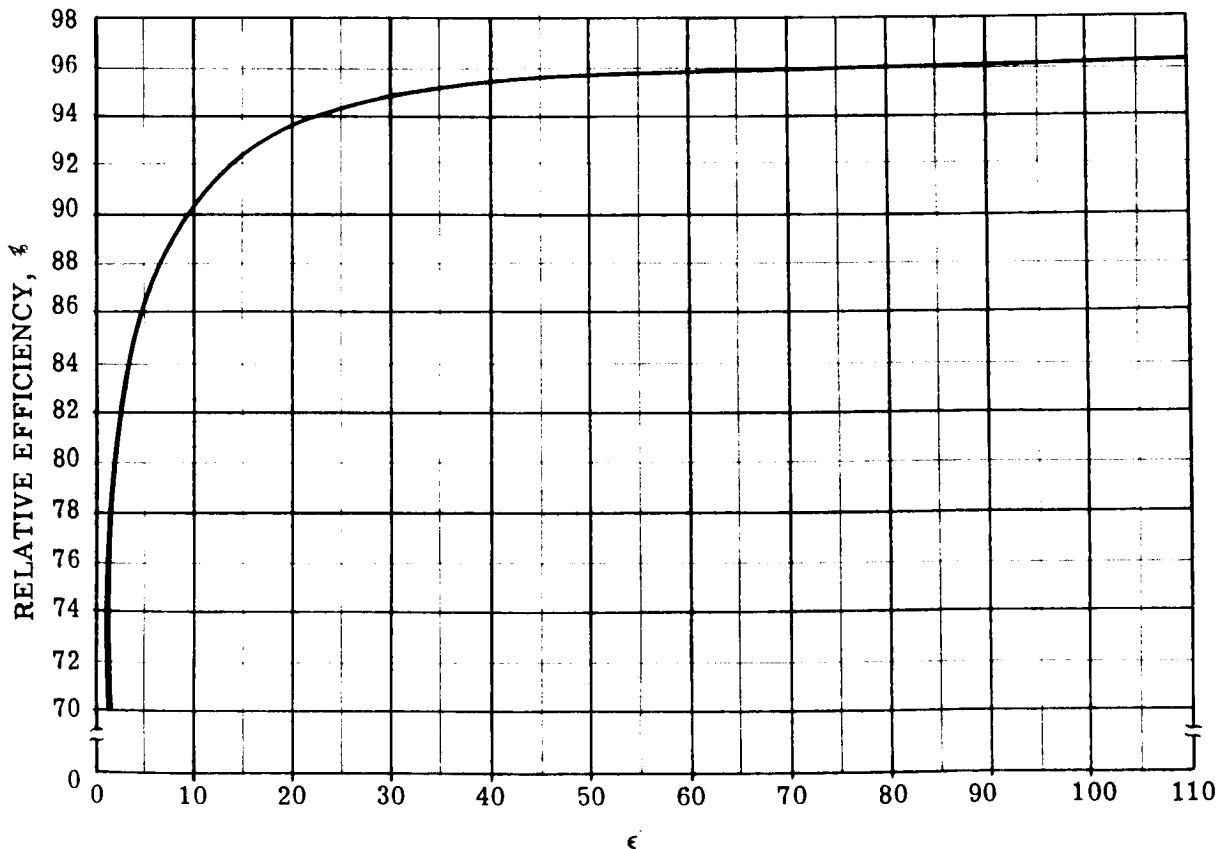


FIGURE 4-75.—Relative nozzle efficiency as a function of expansion area ratio, for a nozzle operating in a vacuum.

$$P_c = 30 \text{ PSIA}$$

$$F_c = 0.05 \text{ LB}$$

NOZZLE IS DISCHARGING INTO A CHAMBER OF $2(10^6)$ CC VOLUME
AT INITIAL PRESSURE OF 0 PSIA.

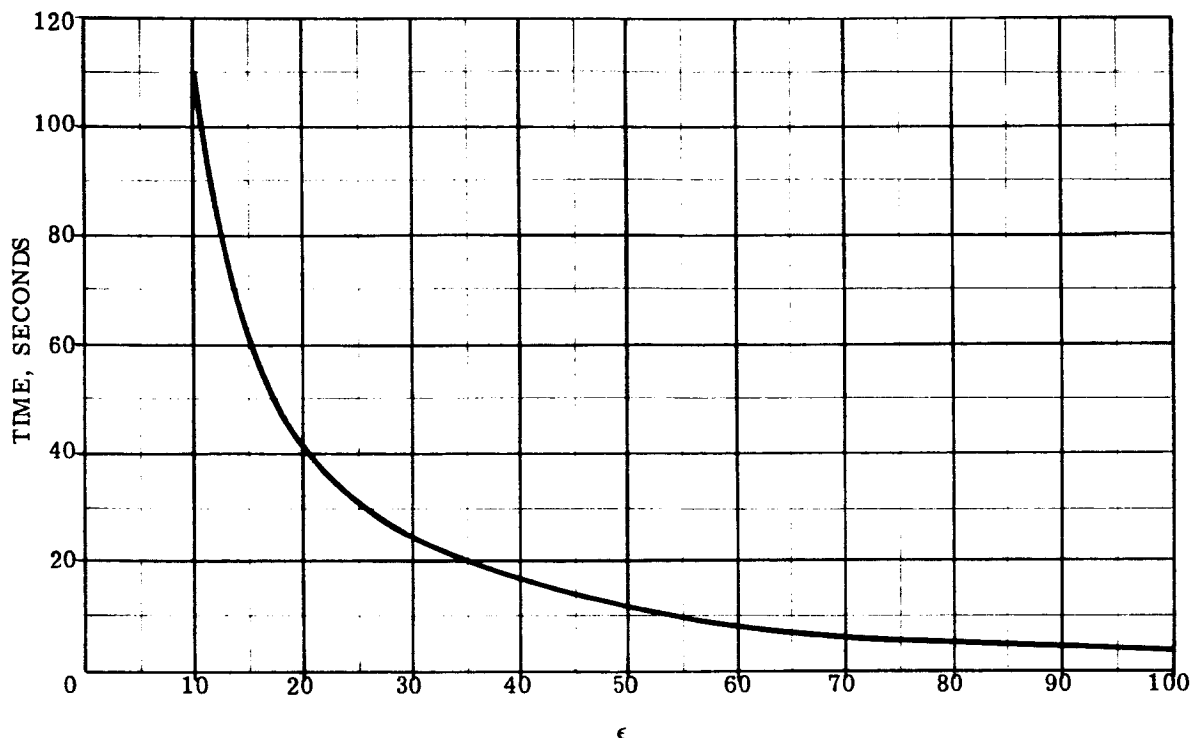


FIGURE 4-76.—Time to reach over-expansion condition as a function of ϵ .

Entrance Cone—Nozzle performance was only very slightly affected by the shape of the convergent section of the nozzle. A right circular cone with 45-degree half-angle was quite satisfactory on many large and small nozzles. Therefore, for the sake of simplicity, a 45-degree entrance cone was chosen.

Throat Shape—In large nozzles, efficiency is bad unless the throat is carefully contoured to provide a smooth path for gas flow. It was felt that this contouring would probably be unnecessary in a very small nozzle. It was decided to use a straight cylindrical throat with no rounding at the ends.

It was shown earlier in this section that if $P_c=30$ psia and $F=0.05$ lb, and if the nozzle is 100 percent efficient, the throat diameter should be 0.034 in. If the nozzle efficiency were 90 percent, then a throat with an effective diameter

of 0.036 inch would produce 0.05 lb. thrust at $P_c=30$ psia. However, it was felt that a throat with a sharp entry would behave as though its diameter were less than its actual value. Therefore, to get 0.036 in. effective diameter, it was decided to try 0.039 in actual diameter.

Exit Cone Shape—If the expanding section of the nozzle is a right circular cone, the exhaust gas stream is divergent. The thrust is not quite as great as if each gas particle were moving parallel to the axis. To correct for this stream divergence, the thrust indicated by Equation (4.40) must be multiplied by $\frac{1}{2}(1+\theta)$, where θ is the half-angle of the exit cone. This function is plotted in Figure 4-77.

In applications where a small increase in efficiency is important, it is customary to contour the expanding section of the nozzle to

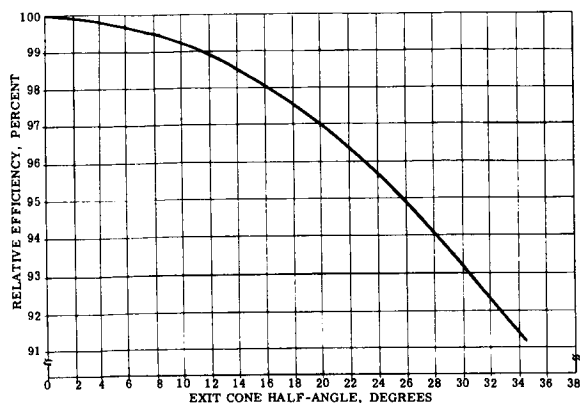


FIGURE 4-77.—Correction for stream divergence.

minimize stream divergence. Since an exit-cone half-angle of less than 16 degrees was anticipated, where contouring could not be expected to give more than 1 percent improvement, a simple straight cone was chosen.

The effective throat diameter of 0.036 inch and ϵ of 30 required that the nozzle exit have a diameter of 0.20 inch.

The optimum angle for the exit cone was determined experimentally. Several nozzles were made with similar entrance cones and throats, but with different exit-cone angles. These nozzles were run with $F=0.05$ lb. and with $F=0.10$ lb. In both cases, $\theta=12.5$ degrees gave the highest efficiency.

Nozzle Efficiency—Several nozzles as shown in Figure 4-78 were tested at 0.05 lb. thrust with $P_a=0.03$ psia (approximately). The average flow was $m=2.22 (10^{-5})$ slug sec^{-1} . Figure 4-79 shows that in vacuum the flow for $F=0.05$ lb. would be $2.18 (10^{-5})$ slug sec^{-1} , so the efficiency of the actual nozzle, operating in vacuum, is 90 percent. The same efficiency was measured with the nozzle operating at 0.10 lb. thrust.

Evaluation of Design—Figure 4-79 shows that the theoretical limit for efficiency of nozzles with $\epsilon=30$ is 93.4 percent. Figure 4-77 shows that the theoretical limit for nozzles with $\theta=12.5$ degrees is 98.75 percent. The limit, then, for nozzles with $\epsilon=30$ and $\theta=12.5$ degrees is 93.2 percent. The nozzle chosen gave 90 percent efficiency. It is not known

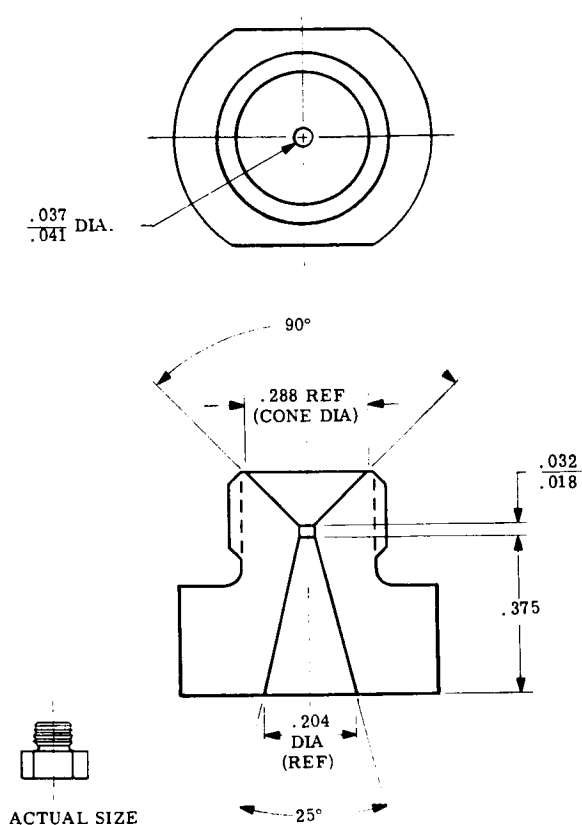


FIGURE 4-78.—Nozzle assembly.

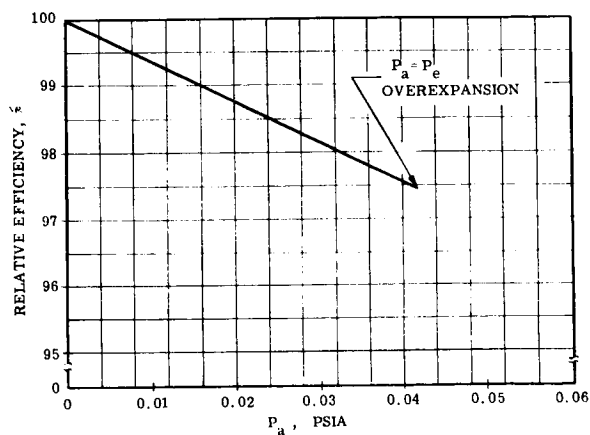


FIGURE 4-79.—Theoretical limit for efficiency of nozzles.

how much of this 3 percent discrepancy was due to gas viscosity, how much was due to the non-contoured throat, or how much was due to

other factors, but it is clear that there was not room for significant improvement.

There were no highly critical dimensions in the nozzle. Efficiency is insensitive to changes in throat diameter, throat length, entrance angle, exit angle, and exit diameter. This made the nozzle easy to build and inspect.

Thrust measurements confirmed the estimate as to the effective throat diameter. In vacuum, the nozzle gave 0.05 lb. thrust at $P_c=30.0$ psia and 0.10 lb. thrust at $P_c=60.0$ psia.

This nozzle was completely satisfactory for its purpose.

PERFORMANCE

The performance discussed in this section is in some cases necessarily that of the complete control system loop since this is all that the information transmitted from the spacecraft can report.

Control Systems

Azimuth Positioning

The accuracy of azimuth positioning was designed to be ± 1 arc minute with the actual design requirement being ± 5 arc minutes. The system was adjusted for zero pointing error on the ground with the wheel section spinning at 0.5 rps.

It was anticipated that when orbit was achieved and the upper structure became weightless, the azimuth error would be other than zero and in the direction opposing wheel rotation due to reduced bearing friction. It

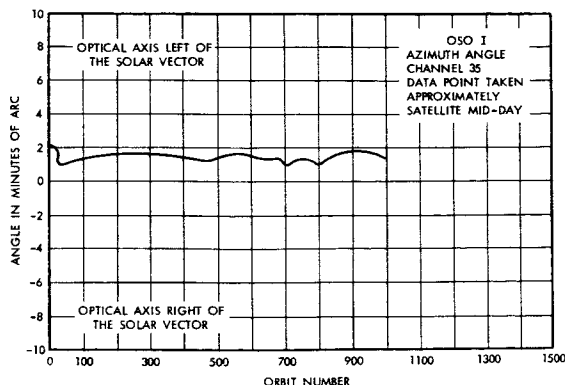


FIGURE 4-80.—Azimuth pointing, 1000 orbits.

was anticipated that the bearing friction might increase slightly after a number of orbits and thus reduce the anticipated error toward zero.

As can be seen from the curve of Figure 4-80, the initial error was not in the direction anticipated, but the long-term error was as expected. The initial error was about 2 minutes of arc. This was reduced to about 1 arc-minute after a few orbits and was from that point on about 1.5 ± 0.5 arc-minutes. The constancy of the positioning demonstrates the long-term reliability of the positioning control system.

One possible explanation of the initial error being in the direction opposite to that anticipated is that the dry lubricated bearings may have been *run-in* at the 1 g load ground test position and may have had to be *run-in* again at the orbit unloaded position. The fact that the error was reduced over several orbits from the initial error supports this reasoning. An interesting hump appears in the curve of Figure 4-80 near orbit 600. The azimuth control system power usage, shown in Figure 4-81, shows the increase in power required to maintain point during this time.

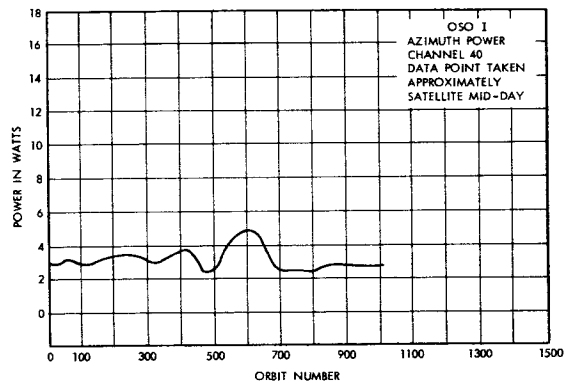


FIGURE 4-81.—Azimuth power, 1000 orbits.

The average power usage for the azimuth control system, as shown in Figure 4-81, was about 3 to 3.5 watts. A nominal value of 2.5 watts was anticipated; thus, the power required was increased by the increase in bearing friction over that expected.

The engineering experiment eye block on the alternate pointed experiment had an azimuth error detecting eye pair consisting of two of the

six eyes in the block. The data from this eye pair corroborated the positioning accuracy as read from the readout-detector eye pairs. This agreement assured that the errors recorded were real positioning errors and were not due to movement of the pointed experiments relative to their set positions on the ground or due to system electronics gain or null point drifts.

One other interesting point displayed in the data is that of the small changes in the azimuth positioning error. The servo drives were designed such that there is a drive *dead zone* for small error signals. As previously discussed a small error must exist to produce drive (see Figure 4-37) and for the azimuth servo this error is always present due to bearing, slip rings, and frictional drag of the torque motor brushes. Thus it was anticipated that the positional error in azimuth would be that error required to produce a drive at one edge of the dead zone. Furthermore, for a constant frictional drag, it was anticipated that the error would be due to a variation in the friction level. The variations seen in the positional error then were due to small variations in the friction level.

It is this variable friction problem that must be solved when more accurate positioning is required. Liquid lubricants can produce a much lower and a much more constant bearing friction level, and for this reason are being considered for use on future spacecraft of the same design.

Elevation Positioning

Elevation positioning was designed to be $\pm 1\frac{1}{2}$ arc-minutes with the design requirement being ± 5 arc-minutes. The system was adjusted on the ground to zero error and it was anticipated that the positioning error would drift between the dead zone set points of $\pm 1\frac{1}{2}$ arc-minutes with perhaps one direction of error predominating due to magnetic coupling effects or to flexible lead unbalanced *spring* action. As can be seen from Figure 4-82 this was definitely not the case.

Figure 4-83 which shows the pitch angle and the elevation error, quickly explains why the anticipated conditions were not achieved. The only mechanical connections between the space-

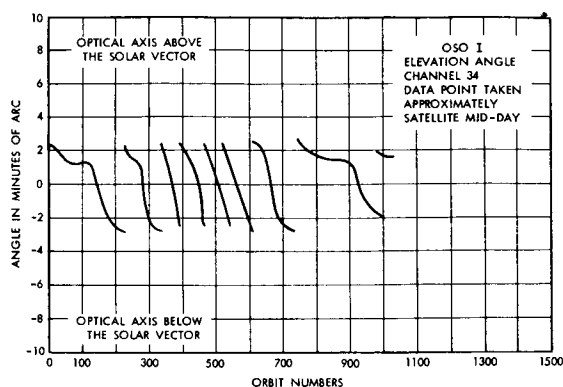


FIGURE 4-82.—Elevation pointing, 1000 orbits.

craft structure and the pointed experiments were the gimbal bearings and the flexible leads to the experiments. If the elevation bearings were brinnelled during the launch vibration, the elevation error would be as seen in the curves somewhere near the zero pitch angle, but would be constant elsewhere; and furthermore, no power would be required to hold the experiments pointed in elevation at any points other than the edges of the brinnelled flats on the bearings. As can be seen from the figure, this is not the case.

This leaves only the flexible leads to produce the error which is obviously a function of the pitch angle. The flexible leads were painted with the same white paint that was used on the back of the solar array to dump heat. The flexible leads then must have been very cold and thus were probably considerably stiffer than when the pointed experiments were dynamically balanced on the ground at room temperature. It is assumed then that the flexible leads acted as springs attempting to keep the pointed experiments in a position normal to the spin axis and that the elevation control system was forced to use considerable power, as seen in Figure 4-84 to maintain point against the spring action of the flexible leads.

This problem has been solved for future spacecraft by using much longer and more flexible leads and by making the surfaces of the flat leads shiny to keep their temperature near room temperature.

CONTROL SYSTEMS

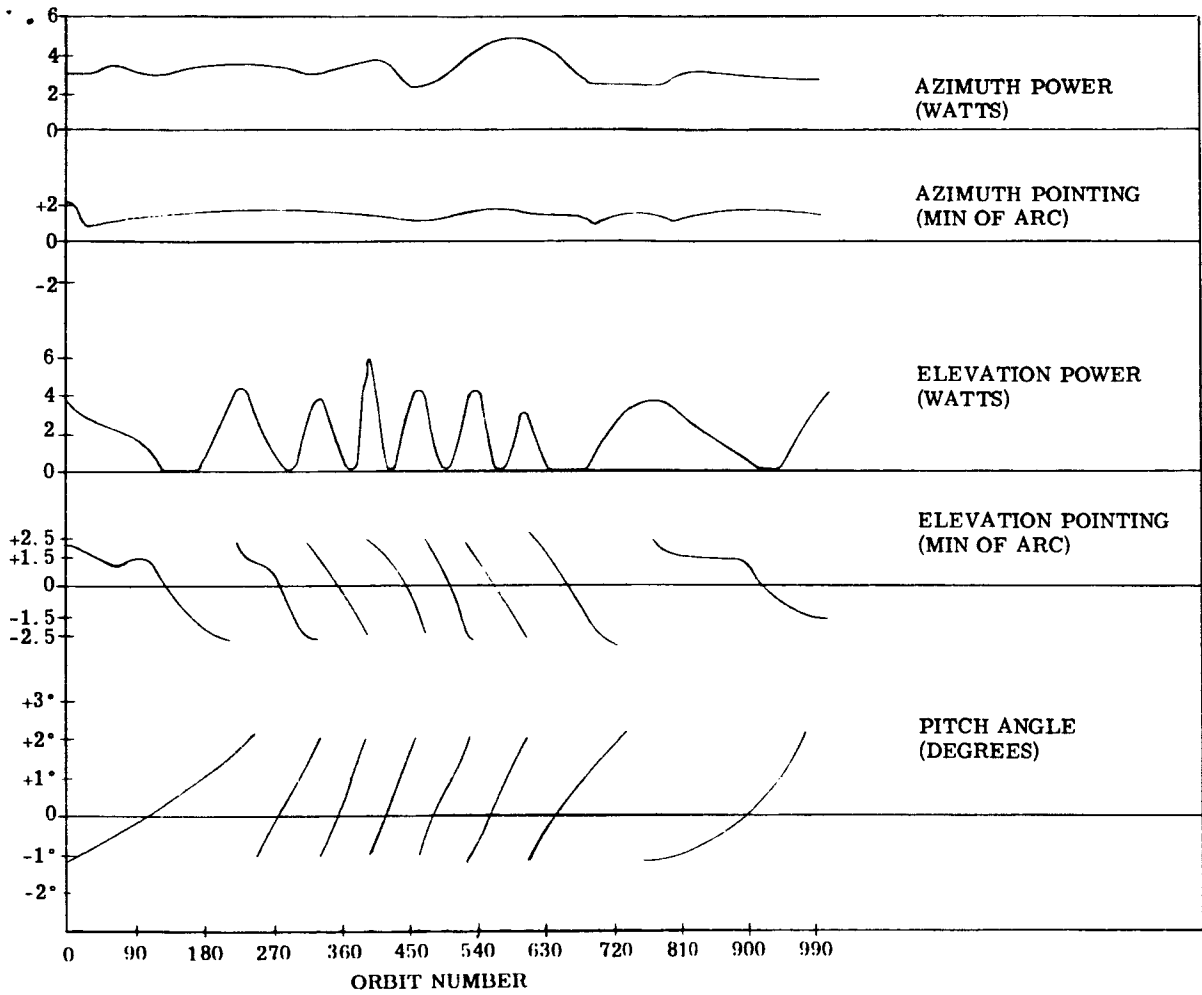


FIGURE 4-83.—Composite performance data, 990 orbits.

The average elevation positioning power was about 2 watts.

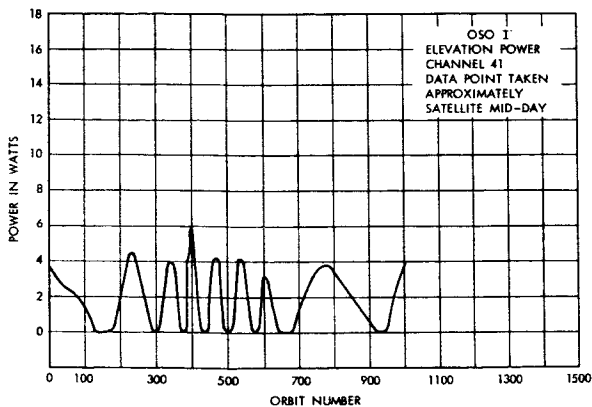


FIGURE 4-84.—Elevation power, 1000 orbits.

Pitch Control

The curves of Figure 4-85 show that the pitch control system, composed of electronic and gas components as previously described, performed very well. Here again, however, a small deviation from the anticipated performance is observed in that the set points for pitch correction were slightly changed from the ground checkout values. The control system was designed to initiate a correction at a 3° error and to overcorrect to 1° beyond the zero-error point.

As previously described, the input signals for the electronics were derived from cadmium sulphide photo detectors. There was a problem associated with these detectors in this application. The cells were used with no filter and

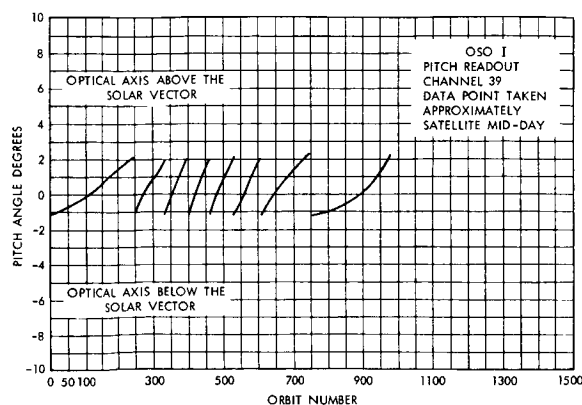


FIGURE 4-85.—Pitch readout, 1000 orbits.

were thus responsive to sunlight illumination as a direct function of their own response characteristics. In order to determine the actual space output, one must know precisely the expected gain increase due to moving above the atmosphere through which the detectors see the sun during ground measurements and checkout. The gain increase from ground to orbit apparently was greater than anticipated, causing the detector output to be high enough at less than 3° pitch error to trigger the pitch control electronics. Some difference in the set points in orbit was anticipated, and since there was no requirement for great accuracy (the solar cells become shaded by the wheel at pitch-up angles greater than 4°) and since it was anticipated that any change would be in a direction to slightly narrow the control range, it was decided to avoid additional development for greater accuracy.

It can be seen in the curves of Figure 4-83 that the pitch angle corrections were made for angles somewhat greater than two degrees and the overshoots were slightly greater than one degree.

Silicon detectors, whose characteristics are much more stable and whose gain increase in orbit is much more predictable, particularly when used behind narrowband filters whose transmission is in an *atmospheric window*, are to be used on future spacecraft.

Spin Control System

The previous performance discussions reveal small deviations from anticipated performance

that had negligible effects on the overall performance of the spacecraft. The problem experienced with the spin control system was of a much more serious nature.

As noted in Chapter 1, the spin-down and spin-up sequencing during initial acquisition was perfect. Furthermore, spin-up control was proper for many orbits thereafter. However, beginning with orbit 895 the spin control system produced several bursts of gas when none were required to maintain spin rate. The last burst occurred near orbit 1150. There have been none since. A spin rate curve is shown in Figure 4-86.

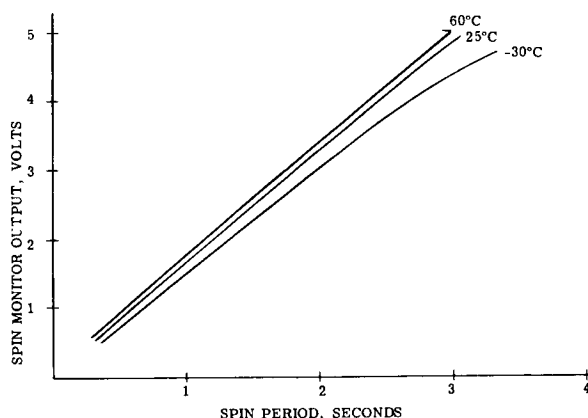


FIGURE 4-86.—Spin rate curves.

During this time, the spin rate was increased from the nominal 0.5 rps to about 0.8 rps. At this increased spin rate, the azimuth servo could no longer position the upper structure to the sun. The reason for this was that the back emf of the azimuth torque motor increased with spin rate, and that the effective net drive for acquisition was reduced as the spin rate increased due to the coarse eye characteristic output signals and the coarse servo lead network properties.

The batteries were discharged quite rapidly with the azimuth servo drawing high power in trying to acquire continuously during spacecraft day. The undervoltage switch switched the load from the batteries as soon as they were discharged. The batteries charged up then to about 10 percent of their capacity at which point the undervoltage switch re-applied the

load. This sequence of events was continuously repeated until the spin rate gradually was reduced by atmospheric drag to about 0.74 rps at which time, during orbit 1622, the azimuth servo regained control.

During the period of time from orbit 1139 to orbit 1622, the only data recorded were during those periods when the spacecraft was not in the undervoltage condition and was within range of a receiving station.

The undervoltage condition described above was repeated after orbit 1822 until about orbit 3320 at which time the azimuth servo again acquired control at about 0.66 rps spin rate.

The reason for loss of control after orbit 1822 was that magnetic coupling effects between on-board magnets and the earth's field caused a sinusoidal variation in spin rate. This variation added to the spin rate after the azimuth servo regained control to again spin the wheel too fast for servo pointing control to be maintained. This coupling effect amounted to an increase of only 0.05 rps, but was enough, when added to the already high spin rate, to cause the nonoriented condition to occur again. The spacecraft was magnetically compensated before flight. Had this not been done, the coupling effects would have been much greater.

The most probable reasons for the second acquisition being at 0.66 rps rather than 0.74 rps after the initial spin-up problem are build-up of brush material on the torque motor commutator causing higher apparent commutator resistance and thus reducing torque, and/or an increase in bearing, slip-ring, and motor-brush frictional drag.

In the following paragraphs some probable and improbable causes of the spin-up malfunction are discussed. Refer to the circuit of Figure 4-44.

It is difficult to find a place for a malfunction beyond the burst generator. It is thus assumed that pulses triggered the burst generator.

Stray pulses could have triggered the burst generator. No such triggering occurred in prelaunch checkouts or for the first 895 orbits. If this were the malfunction cause, the pulses either grew more severe and then less sensitive

with time. This cause of malfunction is improbable.

A missing trigger pulse for the integrator circuit is a probable cause of malfunction. There are three likely possibilities that are all functions of the detectors. These are as follows:

1. *Gradual Reduction of Detector Illuminated Sensitivity:* There are two reasons why this is thought to be an unlikely explanation of the malfunction. First, the detectors of similar type ("on target" detectors and "day-night turn-on" detectors) have not shown a similar problem. Second, if this had been the problem, several spin-up cycles over a short period of time would have been the result. This was not the case.

2. *Detector Continuously Illuminated:* The detectors were used in such manner that only minimum illumination was required to produce a resistance low enough to trigger the pulse integrator. This was done to insure that the turn-on eyes would give a dark signal to turn off spin-circuit power at an illumination level considerably higher than that required for reliable spin-circuit operation. The reason for this was to prevent the spin circuit from missing a pulse just before power turn off. A missing pulse would result in a spin-up burst at every spacecraft dusk. If the pulse shaper were saturated continuously for one revolution of the wheel, a spin-up burst would be generated. A continuously illuminated detector would have produced this result. No possible earth-sun-spacecraft geometry could have produced the condition of an illuminated earth such that the spin detectors could see the earth continuously for one revolution. Thus, a continuously illuminated detector was an improbable cause of the malfunction.

3. *Detector Dark Resistance Reduced:* If the dark resistance of the detectors were reduced from a nominal 500K ohms or greater to the range 200K-280K ohms, the pulse shaper would have been saturated. If the dark resistance were decreased gradually (a possibility if in a radiation environment), the pulse shaper would gradually have become insensitive as the circuit neared saturation. Just before permanent saturation, the pulse shaper would

occasionally miss pulses and cause spin-up bursts. There was another effect that could become important under these conditions, that is, the flip-flop to which the shaped eye pulses were fed was unsymmetrical as can be seen by $6.8 \mu\text{f}$ and 220K tending to hold the base of one flip-flop transistor on. Furthermore, if there were feedback from the solenoid driver when none was wanted, it would have caused the flip-flop to be hard to change from the *no spin-up pulse* state such that the condition of *no spin-up pulses* would be favored. It is thought that this reduced dark current theory best explains the malfunction.

The failure was apparently temporary since the circuit worked properly when the spin rate was reduced to the nominal spin-up set point.

Silicon detectors whose characteristics are much more reliable are being used on future spacecraft designs as spin-circuit input elements. Some circuit modifications are also being made for future applications.

15-Volt Regulators

The readout data from the regulators is shown in Figures 4-87, 4-88 and 4-89. As is shown,

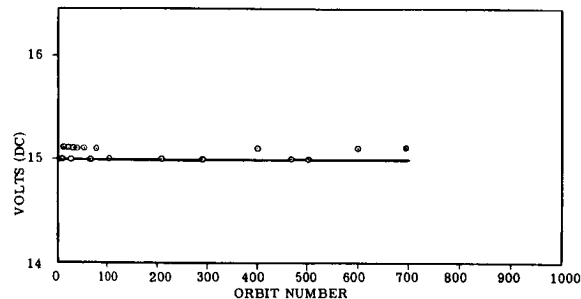


FIGURE 4-87.—15-volt power amplifier monitor channel 45, readout data.

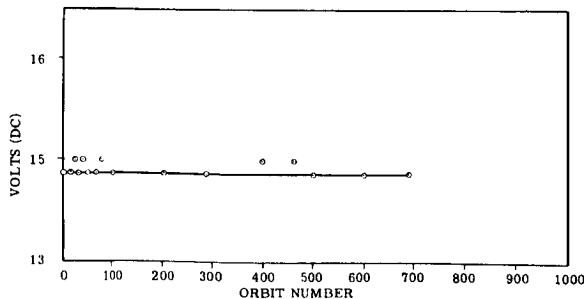


FIGURE 4-88.—15-volt linear amplifier monitor channel 44, readout data.

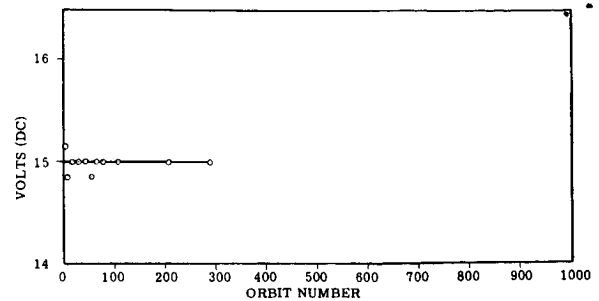


FIGURE 4-89.—15-volt wheel monitor channel 25, readout data.

the regulators regulated well. Further evidence of their excellent performance is in the good performance of the control systems operated from the regulated voltages supplied.

Day Power Turn-On

The day power turn-on circuit worked very well. The circuit never failed to reliably turn power off at night and on during the day. One problem was exposed, however, in the operation of the circuit. The sensitivity *hysteresis* band between the illumination level required to switch power on and that reduction in level required to switch power off was not set to be wide enough. When the spacecraft was in the dawn and dusk portions of the orbit, the illumination level was for about two seconds, such that the power was cycled on and off about five times before either staying off at dusk or staying on at dawn. This switching was no consequence to any spacecraft functions, but was unfortunately very confusing to the University of Minnesota experiment. The experiment used circuitry that detected and counted each power on-off cycle as one orbit. The experiment then had been counting orbits in very rapid groups of 5 or 6 which made data reduction more difficult than anticipated.

Silicon detectors of much more reliable characteristics are being used on future spacecraft designs, and only one eye is used, thus eliminating the need for a hysteresis band.

Undervoltage Switch

Unfortunately, as described fully in the spin-circuit performance description, the undervoltage circuit was given ample opportunity to

demonstrate excellent performance. The circuit worked properly several hundred times. The circuit was designed to give a last measure of life to the spacecraft. As it happened, the life of the spacecraft without the protection would have been only two and one-half months.

Monitor Circuits

As can be seen from the curves of this section, all of the monitor circuits performed well. Data was permanently lost from most of the monitors after the commutators failed.

Sensor Monitoring Experiment

The space environment effects upon the Ball Brothers engineering experiment detectors are listed below.

Detector Type	Degradation (percent per month)
Open cell.....	3.6
Fine pointing.....	1.8
Fine pointing.....	1.2
Coarse pointing.....	0

These degradation figures were derived over the first 1000 orbits and were, for this period of time, linear functions of time. It was assumed that the data could be extrapolated to 6 months with some confidence if the electron flux problem was ignored.

The open-cell detector exposed the silicon cell directly to solar irradiation. The degradation was due to direct ultraviolet irradiation damage.

The fine pointing detectors, with lenses and narrow transmission band filters, were the type used for the servo-positioning-control error detectors. The two detectors were not spectrally matched. This is the reason for the slightly different degradation rates. The servo gain decrease in 6 months (9 percent average) due to eye output reduction caused a positioning error of the same percentage applied to the initial error. That is, with a 1.0 arc-minute initial control accuracy, the error after 6 months was increased by 0.1 arc-minute. The variation in positional accuracy caused by the different degradation rates was negligible.

The coarse pointing detector was identical to the coarse detectors used for the positioning control system. No degradation was measured.

The detector output varied with good agreement to the theoretical variation in the solar constant. This information indicated that the degradation measured for the fine pointing detectors was due entirely to effects on the lens glass (probably ultraviolet-radiation-induced color center formation causing some attenuation), and that the degradation of the cells was negligible when protected by the filter glass.

Of considerable interest would be data concerning the effects of high energy electron flux on the detectors. Unfortunately, the detector outputs were reduced to the lowest level that was readable by about orbit 2000. This was about the time the 9 July 1962 high-altitude nuclear device, which caused the electron flux, was exploded. Since then the coarse detector eye did not continue to give on-scale readings after that time, as would have been anticipated since the solar constant was on the increasing half-cycle, it was assumed that some radiation change had occurred. This change would have had to be only about 3 percent to produce the effects measured, thus no real limits can be estimated. It is thought that the effect of electron flux on the detectors would be evidenced only in the lens glass from the x-rays caused by the electrons impinging on the lens light shields. It is thought that these x-ray effects may be of greater magnitude than the ultraviolet effects. The effects of the x-rays on the silicon cells is thought to be negligible since the energy levels of the x-rays produced are less than the destructive energy level threshold for silicon semiconductors.

The azimuth position readout pair of detectors performed well. After the upper commutator failure about orbit 1900, this detector pair became the only readout available to allow determination of azimuth pointing accuracy.

A slight shift of about $\frac{1}{2}$ arc-minute in relative azimuth pointing direction of the pointed experiments occurred during launch. After this shift had occurred, agreement was found unchanged between the actual azimuth pointing readout detectors and the pair used on this experiment. After 5000 orbits the experimental pair was reading azimuth positioning errors of less than one arc-minute.

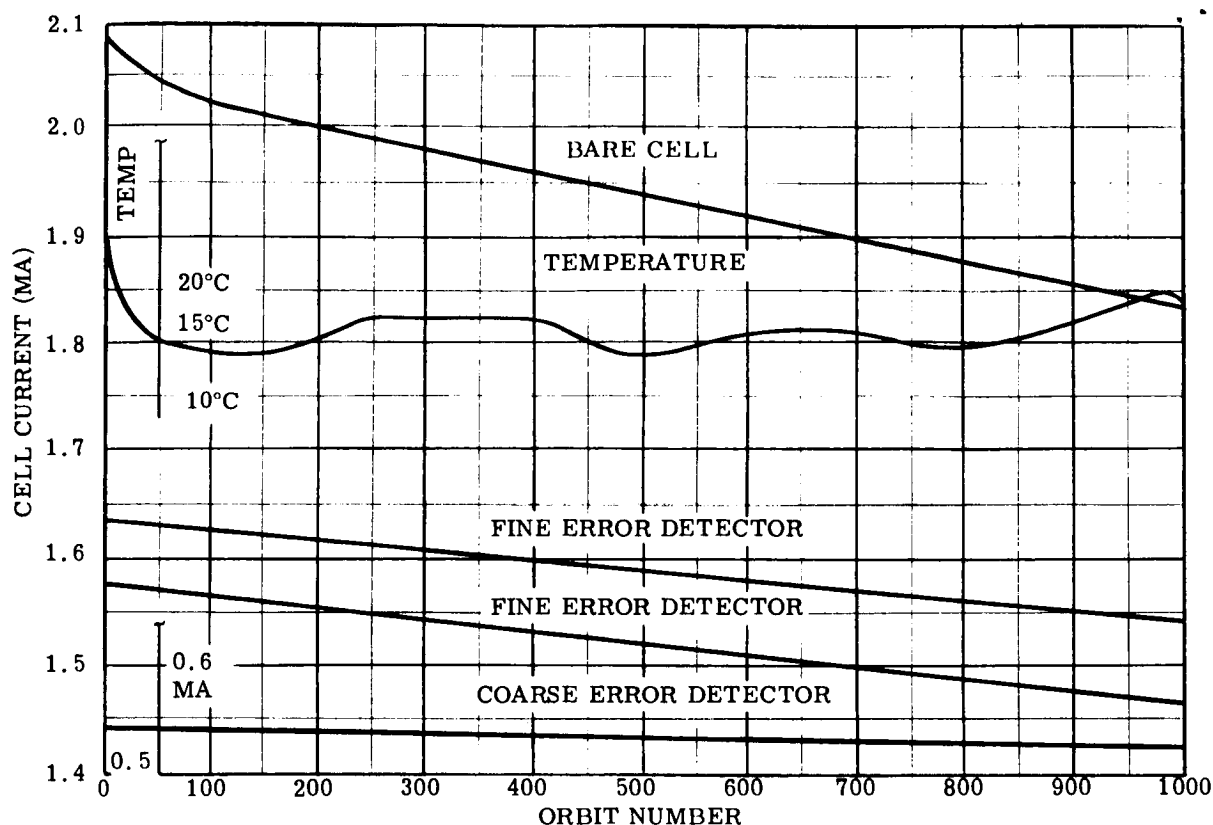


FIGURE 4-90.—Error detectr stability experiment.

Two curves of interest are presented concerning the experiment. The curves of Figure 4-90 show the combined effects of degradation and solar constant variation over the first 1000 orbits. The solar constant change was a decrease of 3.2 percent over this period. The coarse detector, as noted previously, followed this change very closely.

The curve of Figure 4-91 shows the coarse detector output variation over one orbit. The peaks after dawn and before dusk were of variable amplitude and varied somewhat in shape and in time as the spacecraft-earth-sun orientation changed. The peaks were due to earth light inputs to the coarse detector. The curve plotted is typical.

Of value to the circuit designers would have been a time-attenuation curve for solar intensity as the spacecraft emerged from and went into the earth's shadow. Due to the sensitivity of the fine detector readout, a decrease (first few

orbits) of about 10 percent in output produced *low-end-of-scale* readings. Thus, this curve could not be obtained. Of some interest, however, is the fact that the detectors did decrease this 10 percent over the period of time from about 21 seconds prior to power turn-off to about 13 seconds prior to power turn-off. This indicates that there is some atmospheric attenuation in the narrow infrared bandwidth to which the detectors are sensitive prior to that attenuation due to the spacecraft entering the earth's penumbral shadow.

Commutators

Four commutators were incorporated in the OSO I. These were the two spacecraft house-keeping channel commutators, providing 30 points of information in the wheel section and 22 points in the upper section, the alternate pointed experiment 20-point commutator, and the Rochester 20-point commutator.

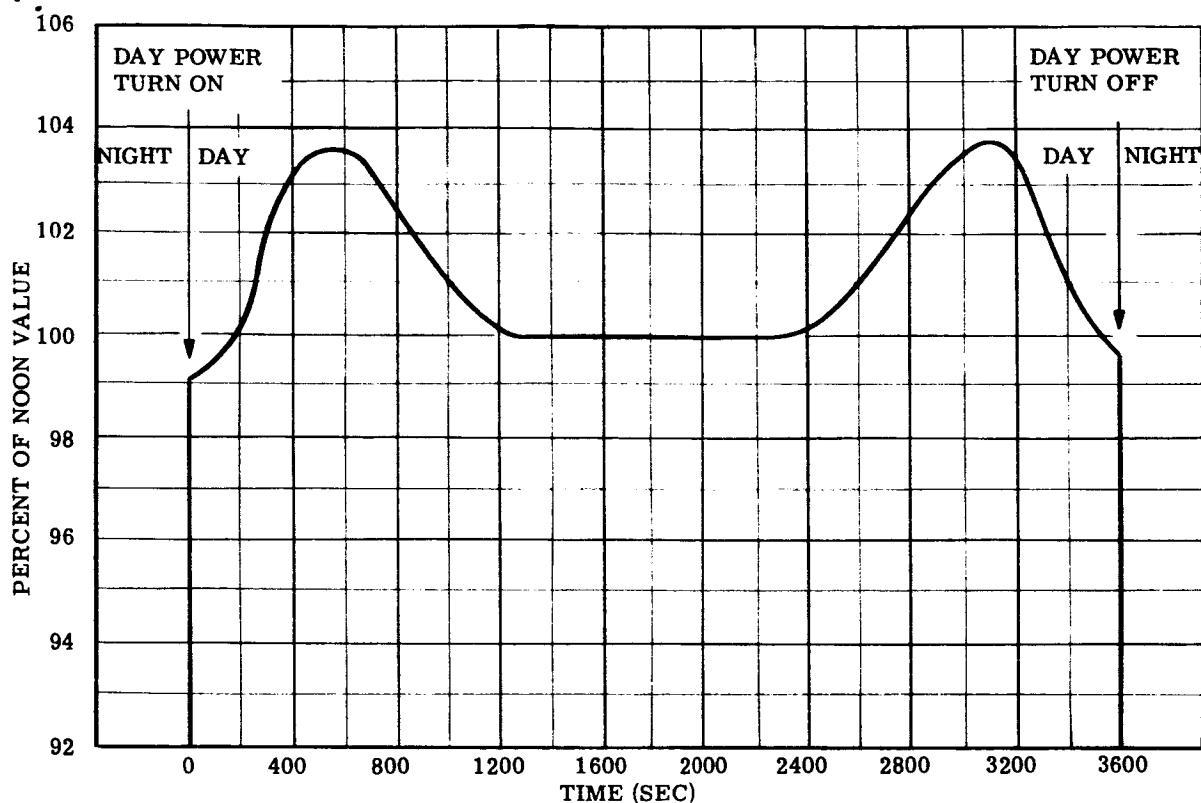


FIGURE 4-91.—Error detection stability experiment.

Of these four devices, two have failed. The spacecraft lower commutator failed during orbit 341, causing the loss of 22 of the 32 lower commutator data points. The spacecraft upper commutator failed after orbit 1900 with all of the upper commutator points being lost. The experiment commutators were working properly after 3500 orbits.

The housekeeping data commutators were designed such that the failure of a switching relay to make contact would cause the device to reset to channel 1 and would thus allow cycling from channel 1 to the channel switched by the bad relay. This was accomplished by a circuit that allowed the relays 0.5 seconds to throw and would cause a reset if relay closure

was not accomplished in this time. The lower commutator has been operating in this sequence since the failure of the relay that switched the eleventh data point.

The reason for the upper commutator failure was not understood. The commutator was not cycling. The method of operation of the two commutators was as follows. The lower commutator ran its sequence in 32 seconds with 1 second dwell time data sampling and then reset to point one. At the reset time the output line from the upper commutator, which was cycling continuously, was switched into the output line through the lower commutator. The lower commutator completed its sequence while the upper commutator was read out.

Chapter 5

DATA ACQUISITION AND COMMAND SYSTEM

INTRODUCTION

The data acquisition system on the OSO I was designed to perform the following functions:

1. Accept the analog outputs of seven* data channels and one reference channel and combine them into a frequency multiplex of eight subcarrier oscillators.

2. Record the frequency-multiplexed signal on a tape recorder for 90 minutes.

3. Upon ground command, play back in 5 minutes the recorded multiplex at 18.35 times the record speed. The playback signal frequency modulates a 1.75 watt transmitter.

4. Radiate the transmitter RF signal via a near-isotropic antenna system.

The command system on the OSO I was designed to perform the following functions:

1. Receive a tone-modulated command carrier via the spacecraft antenna system.

2. Demodulate and select the particular tone command channel via the receiver and decoder, respectively.

3. Actuate 10 spacecraft command functions.

The data acquisition and command system required a total of 4.6 watts of power at 19 volts. Two systems, a primary and alternate, were used, having a total weight of 23 pounds packaged in a volume of 442 cubic inches.

SYSTEMS DESIGN

The Data Transmission Problem

Satellite Range

The orbit was circular at 300 nautical miles altitude. The inclination to the equator was approximately 33 degrees and the period about

*In some cases several experimental measurements were combined into a single channel.

95 minutes. The maximum range at which data was received from the satellite was 1000 miles. The time allotted for the transmission of stored data was 5 minutes. To remain within the range and time requirements, the boundaries of the area for data transmission were as follows: for a direct overhead pass generally from west to east, the 5-minute transmission interval started at a range of 700 miles when the satellite was 32 degrees above the horizon and ended at the same range and elevation angle above the opposite horizon; for passes other than directly overhead, the requirements for 1000-mile range and 5-minute transmission interval were satisfied so long as the orbital path was no farther than 645 statute miles on either side of the overhead orbital path.

The following computations were made to determine whether the combination of transmitter power, bandwidth, spacecraft antenna patterns, and receiving station parameters would yield a satisfactory signal for data analysis.

Given Parameters

The following known parameters are used in the data transmission problem:

Maximum range: 1000 miles

Satellite antenna gain: unity with maximum short-angle nulls of 6 db. The nulls are determined by plotting the composite of linear and horizontal polarization

Receiving antenna gain available: 19.2 db

Receiver bandwidths available:

Model R-220: 8 kc, 45 kc, 70 kc, 180 kc

Minitrack Mod I: 10 kc, 30 kc, 100 kc, 300 kc, 1 Mc

Receiver of preamplifier noise figure: 3 db

Frequency: 136.74 Mc

Transmitter output and radiated power during conditions indicated:

Power out:

Playback..... 1.75 watts

Real-time..... 300 milliwatts

Radiated power: (2 db loss in antenna system)

Playback..... 1 watt

Real-time..... 200 milliwatts

Fades† (136.74 Mc)

Due to refraction: ± 10 db less than 10 percent of the time

Due to absorption: less than 0.3 db

Due to multipath effects: negligible at angles of 23 degrees or greater above the horizon

Galactic Noise

Worst sky condition: 5000°K

Average sky condition: 300°K

Signal Threshold

The most important factor in determining whether an FM-FM transmission system will yield satisfactory data is the required ratio of signal-to-noise to achieve threshold.

The following calculations are based upon the fact that once threshold has been achieved, satisfactory information can be obtained from the telemetry records.

It has been experimentally determined† that in an FM system with large modulation index, the signal-to-noise improvement threshold of the receiver occurs at a point where the ratio of signal-to-noise power is 16, or 12 db. Since in OSO I the modulation index was composite and approximately 2, setting the threshold at 12 db provided an extremely conservative margin. The 12-db figure was conservative because it did not consider the additional improvement realized in the subcarrier discriminators.

When 12 db was used as the signal-to-noise criterion, the receiver threshold occurred at 109 dbm (0.8 μ v) at 100 kc bandwidth and at

114 dbm (0.44 μ v) at 30 kc bandwidth. These values were calculated using a receiver noise figure of 3 db and were also achieved experimentally in the BBRC laboratory.

Calculations for only the wide-band case are as follows: let S equal signal power to achieve 12 db S/N threshold.

12 db = power ratio of 16

N_t = total noise power = source noise + receiver noise

$$N_t = k T_o BW + k T_R BW = k BW (T_o + T_R) \quad (5.1)$$

k = Boltzman's constant, 1.38×10^{-23} Joules/°K

T_o = reference temperature of antenna = 290°K

T_R = effective temperature of 3-db receiver = 290°K

$T_o + T_R = 580^\circ\text{K}$

$BW = 100$ kc

$S/N_t = 16$

$$S = 16 k BW (T_o + T_R) \quad (5.2)$$

$$= (16)(1.38 \times 10^{-23})(10^5)(580)$$

$$= 1.28 \times 10^{-14} \text{ watts}$$

$$= 109 \text{ dbm}$$

$$= 0.8 \mu\text{v across } 50 \text{ ohms}$$

The Transmission Problem Solution

Having determined the signal required to achieve the necessary ratio of S/N, it remains to be shown that with a constant signal and under varying conditions of increased noise, the high power transmission system will operate with sufficient margins of safety. The safety margins of the low-power transmission system can be determined by the ratios of power and bandwidth since these two factors are the only parameters which change.

Given the fixed parameters above, the noise power will be varied simulating ideal, average, and worst conditions.

Ideal Conditions:

Range, d	1000 miles
Frequency.....	136.74 Mc
Wavelength, λ	1.37×10^{-3} miles
Total antenna gain, G_t	19.2 db (ratio 83)
Bandwidth, BW	100,000 cps
Radiated power, P_t	1 watt
Receiver noise figure.....	3 db

†*Propagation Studies for Satellite Systems*, Lockheed Missile and Space Division, Document No. 917040, Phase II, LSMD Document No. 916943

‡"Frequency Modulation Noise Characteristics," Murry G. Crosby, *Proceedings of the IRE*, Vol. 25, No. 4, April 1937

• • Signal-to-noise ratio achieved under ideal conditions:

$$S/N = \frac{P_t \cdot G_t}{k T_i BW} \cdot \frac{\lambda^2}{(4\pi d)^2} \quad (5.3)$$

where:

k = Boltzman's constant

T_i = Summation of source and receiver temperatures

$$= T_{sky} + T_o + T_R$$

$$T_i \text{ (ideal)} = (0) + 290 + 290 = 580^\circ K$$

$$\frac{\lambda^2}{(4\pi d)^2} = \text{Path attenuation}$$

$$= \frac{(1.34 \times 10^{-3})^2}{(12.56)^2 10^6} = 1.19 \times 10^{-14}$$

$$S/N = \frac{(1.0)(83)}{(1.38 \times 10^{-23})(580)(10^5)} = 1.19 \times 10^{-14}$$

$$= 1.24 \times 10^3$$

or, expressed as a power ratio,

$$S/N = 31.5 \text{ db}$$

The safety margin for assumed ideal conditions then equals 31.5—12

$$= 19.5 \text{ db}$$

Average conditions: If it is assumed that all conditions remain the same except that the sky temperature increases to 300°K, then the safety margin is calculated as follows:

$$T_{i(av.)} = T_{sky} + T_o + T_R$$

$$= 300 + 290 + 290 = 880^\circ K$$

$$S/N = 1.24 \times 10^3 \times \frac{580}{880}$$

$$= 0.825 \times 10^3$$

$$= 29.16 \text{ db}$$

However, if the full 19.2 db receiving antenna gain is not utilized due to improper tracking, 3 db should be deducted. In addition, deduct 6 db for spacecraft antenna pattern nulls.

$$S/N_{(av.)} = 20.16 \text{ db}$$

$$\text{Safety Margin} = 8.16 \text{ db}$$

Worst Conditions: By increasing the sky temperature to 5170°K (the remaining parameters being unchanged), the margin of safety becomes:

$$T_{i(worst)} = 5170 + 290 + 290$$

$$= 5750^\circ K$$

$$S/N = 1.24 \times 10^3 \times \frac{580}{5750}$$

$$= 1.25 \times 10^2$$

$$= 20.96 \text{ db}$$

Deducting tracking antenna errors of 3 db and spacecraft antenna pattern nulls of 6 db, the safety margin is essentially zero.

Performance

In flight, better than average conditions of reception did prevail at least 90 percent of the time. The OSO I transmission system provided signals 8 to 15 db above threshold.

RF System

RF Multiplexing System

As shown in Figure 5-1, it was necessary to operate two transmitters and two command receivers using one spacecraft antenna system. Two transmitters were alternately connected into the antenna feed line by a coaxial switch. The switch was actuated by ground command in conjunction with the selection of the primary or alternate transmitter.

To permit the operation of the selected transmitter and the command receivers from the same antenna system, a diplexer was necessary which provided high isolation between the transmitter and the receiver.

For redundancy, two command receivers were operated in parallel. A hybrid junction permitted such operation without the necessity of switching between the two receivers. The input to the hybrid junction originated at the diplexer.

Transmitter

The transmitter consisted of two parts: the modulated driver and the power amplifier. Both units used solid-state active elements.

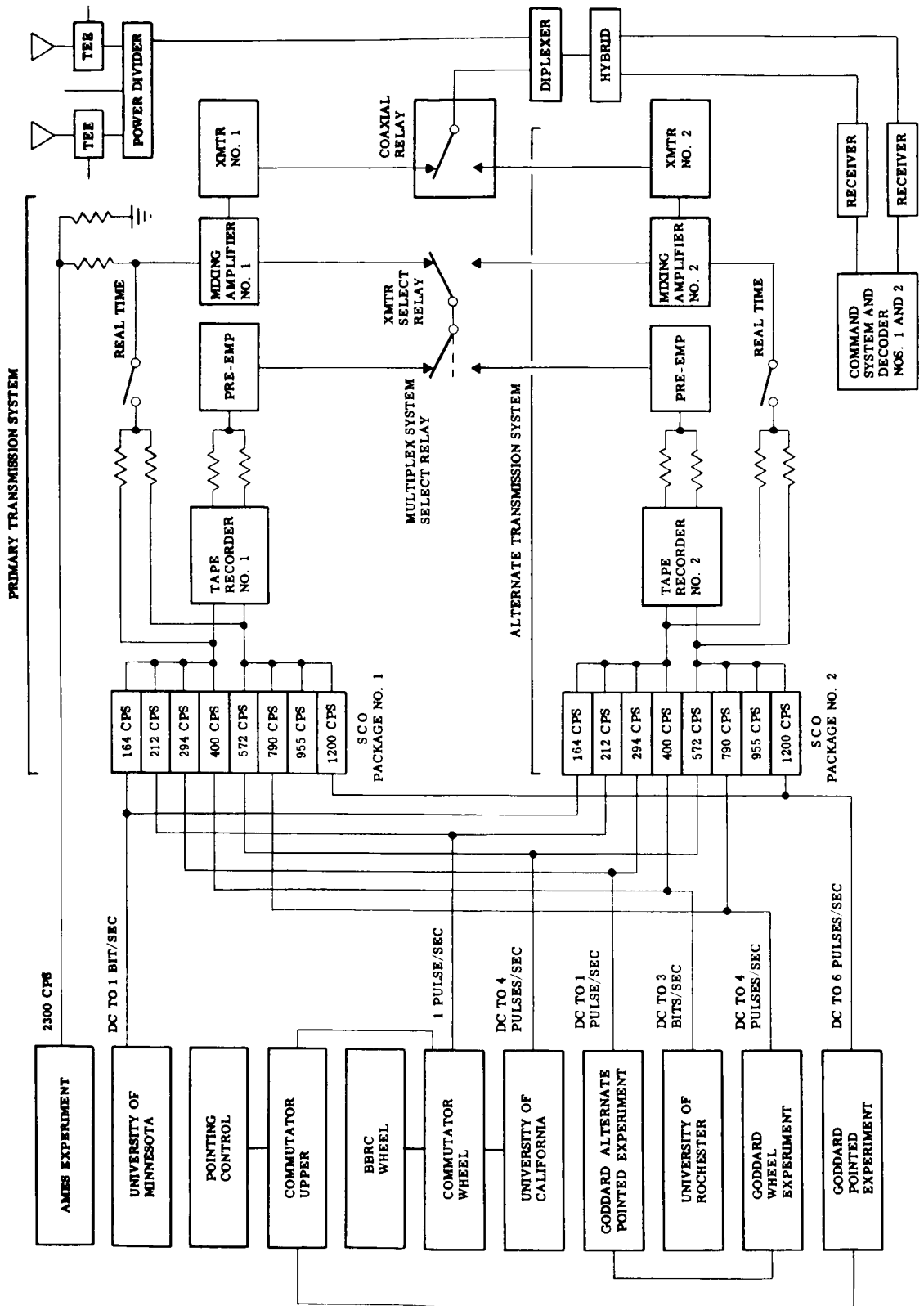


Figure 5-1.—Telemetry system block diagram.

During the 90-minute phase of the orbit, the transmitter output was reduced to 300 milliwatts and the low-frequency, real-time subcarriers were transmitted. When the satellite was within receiver range, the transmitter output was increased to 1.75 watts and the high frequency subcarriers from the tape recorder were transmitted.

The power amplifier was designed to operate without spurious oscillations into a tuned load, such as the diplexer, with a VSWR as great as 2:1.

Antenna System

Considering the vector sum of the vertical and horizontal polarization components, the antenna patterns exhibited nulls no greater than -6 db for 95 percent of all aspect angles. The vertical and horizontal polarization components were received at the Minitrack ground station via two parallel receiving channels. As the spacecraft rotated at $\frac{1}{2}$ revolution per second, antenna nulls were realized in one polarization component or the other. A post-detection diversity combiner performed the summing of the two signals and provided a smoothed video output continuously as great as the stronger of the two signals.

FM Multiplexing System

The OSO I telemetry system was an FM-FM system (Figure 5-1). The outputs from eight nonstandard low-frequency subcarrier oscillators were multiplexed into a magnetic tape recorder for orbital storage. The tape recorder was interrogated and played back the recorded signal when in range of one of the Minitrack ground stations.

At the same time that the subcarrier multiplex was being stored, it was also being fed to the transmitter for continuous real-time transmission. When the tape recorder was played back, the real-time subcarrier signals were removed from the transmitter and the stored frequencies were transmitted. At the end of playback, the real-time frequencies were again connected.

The subcarrier input data rates from the sources indicated below were dictated by a

nominal modulation index of 5. The choice of subcarrier frequencies was predicated upon limited response of the tape recorder in conjunction with tape storage capacity and the necessity to operate at a reasonable transmission bandwidth to conserve transmitter power.

Channel 1: Center frequency 164 cps University of Minnesota—PCM, 1 bit per sec.

Channel 2: Center frequency 212 cps OSO I spacecraft status—52 PAM channels, subcommutated at 1 sample per second.

Channel 3: Center frequency 294 cps Goddard alternate pointed experiment—20 PAM channels, subcommutated at 1 sample per second.

Channel 4: Center frequency 400 cps University of Rochester—20 PDM channels, $\frac{1}{2}$ pulse per second.

Channel 5: Center frequency 572 cps University of California—pulse rate information at 4 pulses per second.

Channel 6: Center frequency 790 cps Goddard wheel experiment—pulse rate information at 4 pulses per second.

Channel 7: Tape speed compensation reference oscillator, fixed at 955 cps.

Channel 8: Center frequency 1200 cps Goddard pointed experiment—pulse rate information at 6 pulses per second.

When the tape recorder played back at 18.35 times record speed, channels 1 through 8 became 3 kc, 3.9 kc, 5.4 kc, 7.35 kc, 10.5 kc, 14.5 kc, 17.5 kc, and 22 kc, respectively. The playback ratio was critical since the subcarrier frequencies had to remain within the ± 7.5 percent bandwidth of the ground station filters.

During playback, an additional channel at 2.3 kc was mixed with the above frequencies at the primary transmitter input. The signal originated at a subcarrier in the Ames emissivity experiment.

To increase reliability, two sets of subcarrier oscillators were provided. Only one set was turned on. No switching was required at the input of the oscillators since each of the data channels was connected to each of the redundant oscillator pairs.

The subcarriers were recorded and played back at equal amplitudes. Pre-emphasis taper for the stored frequencies was provided by a suitable filter at the input of the transmitter (see Figure 5-2). It should be noted that due to transmitter background noise, little taper was possible at the real-time subcarrier frequencies.

The subcarriers were required to operate at less than 24 milliwatts and weigh less than 0.15

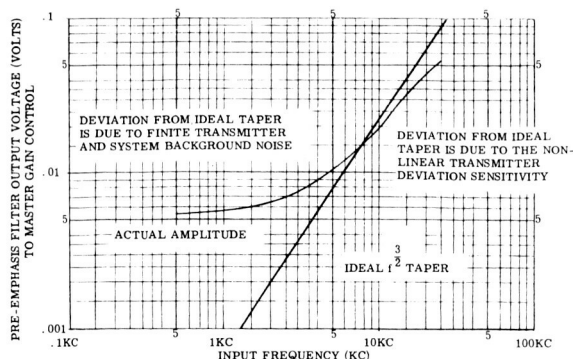


FIGURE 5-2.—Pre-emphasis curve.

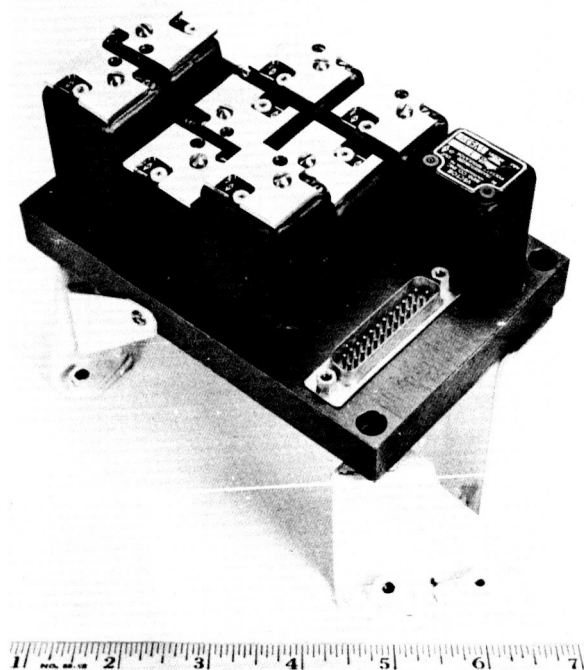


FIGURE 5-3.—Subcarrier oscillator package.

pounds per channel. Figure 5-3 shows one set of subcarrier oscillators, including the reference oscillator and voltage regulator.

Data Storage System

Orbital storage was necessary to permit total recovery of data taken while the satellite was beyond the range of the NASA Mini-track receiving stations. The tape recorder used was Raymond Engineering Laboratory, Model No. 1515.

The record speed of the tape recorder was 0.65 ips. This speed was dictated by the highest frequency to be recorded and the tape storage capacity of 300 feet was required for the 90-minute storage interval.

The pulse averaging discriminating technique permitted effective tape speed compensation in the receiving station discriminators which were tuned to the bands from 3.0 kc to 22 kc.

The tape recorder frequency response provided near uniform amplitudes for channels 1 through 8. This was necessary to insure a uniform signal-to-noise ratio for all channels recorded.

The tape storage system was required to operate with a power consumption of less than 1.1 watts, and it weighed less than 5 pounds.

Airborne Command System

Receiver

The receiver was crystal-controlled AM with -100 dbm sensitivity at 75 percent modulation. It was limited to receiving one tone at a time and was diplexed to the spacecraft antenna with the transmitter. The hybrid junction permitted two receivers and decoders to operate in parallel (see Figure 5-1).

Airborne Decoders

The decoder, consisting of a series of seven bandpass filters, was designed so that the transmission of two tones in succession was required to actuate a given command channel. Tone 1 permitted the selection of the OFF condition in preparation to actuate tones 3 through 7. Tone 2 permitted the selection of the ON condition in preparation to actuate tones 3 through 7. Once a given condition was estab-

lished in the spacecraft it remained until a subsequent command changed it.

The following commands were performed on the OSO I.

Command No. 1: Tape recorder playback ON. Performed by actuating tone 2, then tone 3.

Command No. 2: Playback OFF, tape recorder power ON and alternate transmitter power ON. Performed automatically by the tape recorder when playback is completed or by actuating tone 1, then tone 3.

Command No. 3: Primary transmitter power select performed by actuating tone 2, then tone 4.

Command No. 4: Alternate transmitter select and transmitter power OFF performed by actuating tone 1, then tone 4.

Command No. 5: Primary multiplex system select performed by actuating tone 2, then tone 5.

Command No. 6: Alternate multiplex system select performed by actuating tone 1, then tone 5.

Command No. 7: Wheel experiments ON and tape recorder power OFF. Performed by actuating tone 2, then tone 6.

Command No. 8: Wheel experiments OFF. Performed by actuating tone 1, then tone 6.

Command No. 9: Pointed experiments ON and playback ON. Performed by actuating tone 2, then tone 7.

Command No. 10: Pointed experiments OFF. Performed by actuating tone 1, then tone 7.

Receiving and Command Stations

Mod I Minitrack Receiving System

The data acquisition system for the OSO I was initially designed to be compatible with the Motorola R-220/URR receiver. The bandwidths were adjusted to 8 kc and 70 kc during real-time and playback, respectively. At the time the spacecraft was launched, the GSFC Mod I Minitrack receivers had been installed at the prime receiving stations. The bandwidths were readjusted to 30 kc and 100 kc, respectively.[§]

[§] For more detailed information, refer to the NASA *Satellite Instrumentation Network and Facilities Report*, pages 23 and 24.

Discriminators

Blossom Point, Maryland; Fort Myers, Fla.; Lima, Peru; Quito, Ecuador; Antofagasta, Chile; and Santiago, Chile, were equipped with EMR Model 97 tunable discriminators and Model 95 low-pass output filters. In addition, Fort Myers had a diversity combiner and two EMR Model 189F fixed discriminators with plug-in filters. The 17.5 kc discriminator was used in conjunction with the EMR Model 96F tape-speed compensator and compensated for airborne tape recorder flutter. Flutter variations of 1.5 percent peak-to-peak up to rates of 300 cps were eliminated in the ground station discriminators by the EMR Model 96F. To eliminate flutter, a 955-cps fixed frequency was multiplexed and recorded on the spacecraft tape. Upon playback, this 955-cps signal became 17.5 kc. Any variations in the frequency of this signal were detected by the 17.5-kc discriminator and were fed via the tape speed compensator and appropriate delay lines to the remaining discriminators. The second discriminator had two plug-in filters, one at 212 cps and one at 3.9 kc. The output of the discriminator with either of the two filters was applied to a Sanborn pen recorder for real-time display of the spacecraft functional data. The outlying stations in South America received the signal and used the tunable discriminator and a Sanborn recorder to display and read quick-look data. Neither tape speed compensation for the playback signal nor diversity combining was available at these stations.

Ground Tape Recorder

A 7-track Ampex FR-100 was used at each receiving station to record the following:

Track 1: Receiver Output No. 1.

Track 2: Receiver Output No. 2.

Track 3: Station Tape Recorder Compensation Frequency.

Track 4: Receiver No. 1 AGC.

Track 5: Receiver No. 2 AGC.

Track 6: Voice Annotation and Command Tones which were sent.

Track 7: Time.

Further information on the ground station tape recorders is available in the NASA docu-

ment *Satellite Instrumentation Network and Facilities Report*, page 26.

Receiving and Command Antennas

The receiving and command antennas¹¹ used were 9-element Yagi systems designated YGCI-1A/136-8S. This was a nine-element broad-side array mounted on a remotely controlled azimuth and elevation rotator system atop a 15-foot tower. The system was utilized for both telemetry and command purposes. Each element of the array was in itself an eight-element cross-polarized Yagi antenna. The Yagi used for command was mounted in the center of the array, and the input connections were brought out separately so that transmitted power could be fed to it. The vertically and horizontally polarized outputs from the receiving array were brought out separately; these were passed through notch filters which rejected the command frequency and thereby protected the receiver preamplifiers which were also mounted on the tower. From the preamplifiers the two outputs were brought into a polarization differentiation unit which allowed selection of all possible modes of polarization, either simultaneously or one at a time. To assure correct polarization differentiation, the electrical length and relative phase of both transmission lines, from the antenna output terminals to the polarization selector box, were made precisely equal by means of a line stretcher.

Receiving and commanding antenna requirements were as follows:

Telemetry:

Center frequency: 136.5 Mc
 Frequency range: 135-138 Mc
 Gain: 19.2 db above isotropic
 Polarization: horizontal, vertical, right and left circular
 "E" plane pattern: 19.5° at half-power points
 "H" plane pattern: 19.5° at half-power points
 VSWR: under 1.1 at 136.5 Mc under 2.0 from 135-139 Mc
 Nominal impedance: 50 ohms

Command:

Center frequency: 123 Mc
 Frequency range: 115-131 Mc
 Gain: 13.2 db above isotropic
 Polarization: right circular
 VSWR: under 1.5 from 118-130 Mc
 Power handling capability: 400 watts input
 "E" plane pattern: 42 degrees at half-power points
 "H" plane pattern: 42 degrees at half-power points
 Nominal impedance: 50 ohms

The antenna azimuth scan was 740° and elevation was $\pm 80^\circ$ from vertical at 1 rpm. Azimuth and elevation synchro indicators were provided at the remote control box.

Diversity Combiner

In order to permit quick-look capability at Fort Myers, the video output of each of the two Mod I receivers was combined in a Defense Electronics Model TDC-1 diversity combiner. The composite signal was sent to the EMR discriminators where it was reduced to the Sanborn display.

The two video signals were also recorded on channels 1 and 2 of the station tape recorder. They were subsequently combined in a similar combiner at Greenbelt prior to formal data reduction.

Command Transmitting System

All stations referred to previously had command capability for the OSO I. The Collins 242G-2 VHF transmitter was used for interrogation purposes at these ground stations. The transmitter was crystal-controlled and amplitude modulated. It was capable of operation over a frequency range from 108 to 152 Mc. Special coders were used to supply the modulating signal necessary to actuate the spacecraft command functions. An output power of 200 watts was available. A command timer was used which turned the transmitter on for a predetermined length of time (2 seconds) after the switch was thrown and permitted transmission of the selected command for the timed interval.

The transmitter operated into a load connected through a 52-ohm transmission line

¹¹NASA *Satellite Instrumentation Network Facilities Report*, pages 15 and 16

having a standing wave ratio of 2 to 1 or less. The command antennas on the telemetry nine-element Yagi arrays met the following requirements:

- Frequency range: 108 to 152 Mc
- Power output: 200 watts
- RF output impedance: 52 ohms nominal
- Modulation: AM, plate, and screen

The Crosley J-183110 encoder was installed at the ground stations and used to amplitude-modulate the command transmitter with the selected command tone. The encoder contained the seven crystal-controlled tone generators which performed the command functions.

To command the spacecraft, two commands had to be sent. The first command selected the ON or OFF state, and the second represented the execute command. Each tone modulated the transmitter for the timed interval described previously (2 seconds).

SUBASSEMBLY DESIGN AND QUALIFICATION

Subcarrier Oscillator Subassembly

A block diagram of this subassembly is given in Figure 5-4.

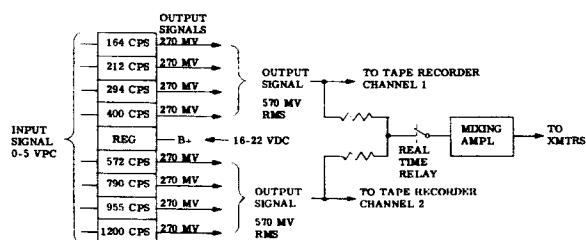


FIGURE 5-4.—Subcarrier oscillator subassembly block diagram.

Design Specifications

The following design specifications provided oscillators which operated properly within channel bandwidths for a period of six months during qualification testing and nine months in orbit.

1. Power requirements per oscillator: 24 milliwatts at 6 volts.

2. Subcarrier system power requirements: 1.1 watt at 19 volts.
3. Subcarrier oscillator center frequencies:

164 cps	572 cps
212 cps	790 cps
294 cps	1200 cps
400 cps	
4. Reference oscillator frequency: 955 cps
5. Input impedance: 250 K ohms.
6. Modulation sensitivity: adjusted for ± 6 percent. (See Figure 5-5 for channel bandwidth apportionment and Table 5-1.)

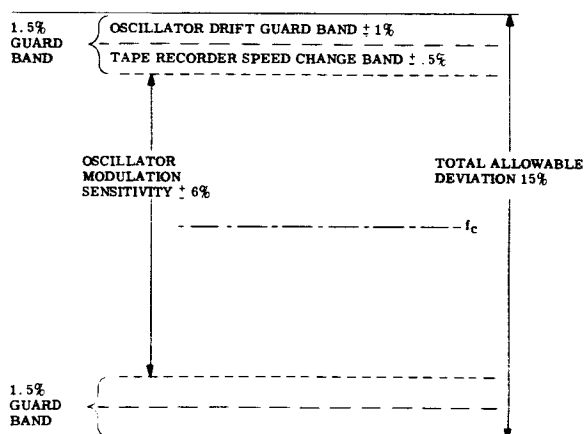


FIGURE 5-5.—Subcarrier oscillator channel bandwidth apportionment.

7. Signal overvoltage protection: the input sensitivity characteristic limited the deviation to $\pm 7\frac{1}{2}$ percent.
8. Total harmonic distortion: less than 1 percent.
9. Linearity: 3 percent of design bandwidth from the best straight line.
10. Long term drift: see Table 5-2.
 - Reference oscillator—less than ± 0.25 percent of the reference frequency.
 - Subcarrier oscillator—less than ± 0.75 percent of their center frequency.
11. Output signal: 270 mv rms.
12. Temperature: -20°C to $+55^{\circ}\text{C}$.
13. Weight and size: $2\frac{1}{2}$ oz.; 1.88 cubic inches.

Table 5-1

Subcarrier Oscillator Requirements

Channel No.	-6%	f_c (cps)	+6%	Bandwidth +6% (cps)	Long term drift limits +cps	Intell. required of each channel	Playback frequency (f_c) $\times$ 18.35 (cps)	IRIG channels (playback) f_c
2	154	164	174	20	1. 48	1 bit/sec-----	3000	8
3	199	212	225	26	1. 91	1 pulse/sec-----	3900	9
4	276	294	312	36	2. 64	1 pulse/sec-----	5400	10
5	376	400	424	48	3. 60	3 bits/sec-----	7350	11
6	538	572	606	69	5. 15	4 pulse/sec-----	10500	12
7	743	790	837	95	7. 10	4 pulse/sec-----	14500	13
9	1128	1200	1272	144	10. 8	6 pulse/sec-----	22000	14
1	Ames Research Center-----						2300	7
8	955 Reference oscillator $\pm$ 2.39						17500	-----

Table 5-2

Subcarrier Oscillator Drift History

		164 cps S.N. 1580	212 cps S.N. 1547	294 cps S.N. 1327 and 1324	400 cps S.N. 1310	572 cps S.N. 1287	790 cps S.N. 1280	955 cps S.N. 1284	1200 cps S.N. 1299
Date	Test time (days)	f_c	f_c	f_c	f_c	f_c	f_c	f_c	f_c
4/26/61	0	163. 8	212. 4	292. 3	-----	572. 6	790. 0	955. 2	1197. 5
7/14/61	79	163. 9	212. 4	292. 5	□387. 2	575. 2	790. 6	952. 3	1197. 2
9/ 6/61	133	164. 2	212. 8	292. 7	387. 6	576. 5	790. 4	953. 0	1198. 4
9/21/61	148	164. 2	212. 6	293. 6	387. 3	576. 2	789. 6	952. 7	1197. 6
10/ 3/61	160	*163. 8	*212. 0	*292. 7	*399. 3	*571. 7	*790. 1	*954. 5	*1198. 0
10/18/61	175	163. 8	211. 8	293. 9	399. 2	571. 9	789. 8	955. 5	1198. 3
10/31/61	188	163. 7	211. 6	295. 7	399. 0	572. 0	789. 6	955. 7	1196. 5
12/28/61	246	163. 7	211. 3	289. 5	398. 1	570. 4	789. 6	956. 3	1097. 1
2/ 5/62	285	163. 8	211. 5	S.N. 1324 *294. 4	398. 3	570. 7	789. 4	956. 1	1196. 8

*Oscillators readjusted.

□ Adjusted for 0-11 input signal, later changed to 0-5 vdc.

Subcarrier Pre-Emphasis Design

In a double FM system, it is desirable to achieve threshold not only at the receiver detector, but also at the subcarrier discriminators. An additional requirement is that each of the subcarrier signals achieve threshold simultaneously. This will be true only if the transmitter deviation per subcarrier is properly adjusted. To accomplish this, it was necessary to transmit the high-frequency subcarrier signals with more energy than the low-frequency subcarrier signals. This was due to the noise characteristics

in an FM-FM system. The magnitude of the random noise spectrum follows a curve proportional to subcarrier frequency taken to the $3/2$ power ($f^{3/2}$). Thus, it was necessary that the transmitter deviation follow an $f^{3/2}$ taper. Figure 5-5 shows the ideal $f^{3/2}$ taper curve compared to the actual curve used. The ideal curve could not be realized at low transmitter deviations due to system background noise.

The requirement for the $3/2$ power taper comes about from two distinct concepts of noise amplitude as a function of frequency: (1) noise

voltage per unit bandwidth at the output of the FM carrier demodulator is directly proportional to the modulating frequency, $N \approx f$, and increases with frequency at a constant rate of 6 db per octave; and (2), the subcarrier bandwidth in an FM-FM system is a fixed percentage of the subcarrier frequency and will, therefore, increase as the subcarrier frequency increases ($BW \approx f$). Since noise voltage increases as the square root of the bandwidth, $N \approx (BW)^{1/2}$, and bandwidth is proportional to frequency, then there is a second increase in the noise voltage, $N \approx f^{1/2}$.

The total noise introduced into the input of the subcarrier discriminator is then the product of these two sources. $N \approx f \cdot f^{1/2}$ or $N \approx f^{3/2}$. Therefore in order to maintain a constant signal-to-noise ratio over the multiplex spectrum, the deviation of the transmitter per subcarrier signal had to follow the $f^{3/2}$ taper.

The first steps in pre-emphasis design were accomplished using theoretical data, but the final adjustment was a matter of trial and error. Table 5-3 shows the ideal pre-emphasis. Figure 5-2 shows how the ideal had to be modified to keep the low frequencies above the transmitter background.

Table 5-3

Ideal Pre-Emphasis Schedule for the OSO I Transmission System

f	$f^{3/2}$	Percentage of total deviation
2,300 cps-----	110, 000	1. 11
3,000 cps-----	165, 000	1. 66
3,900 cps-----	290, 000	2. 92
5,400 cps-----	400, 000	4. 03
7,350 cps-----	630, 000	6. 35
10,500 cps-----	1, 080, 000	10. 9
14,500 cps-----	1, 750, 000	17. 6
17,000 cps-----	2, 200, 000	22. 2
22,000 cps-----	2, 300, 000	33. 2
	9925×10^3	99. 97

The next step in pre-emphasis design was to adjust the amplitudes of the subcarriers to permit the percentage deviations indicated.

Due to the nonlinearity of the transmitter deviation sensitivity curve (Figure 5-6) the percentages given did not indicate the relative amplitudes. The amplitude providing a prescribed deviation is a function of the transmitter sensitivity at the deviating frequency.

On the OSO I, a filter was used to obtain the relative amplitudes. A master gain control was then used to vary the amplitude of the composite signal to control the deviation of the transmitter within the restraints of bandwidth.

It was determined chiefly by empirical methods that the multiplex described could be contained in a bandwidth of 100 kc with negligible loss of sidebands during peak deviation. In adjusting deviation to conform to bandwidth restrictions, doppler shift as well as transmitter and receiver drift were considered.

There was no indication of drift in the subcarrier oscillators during the first nine months of orbital life. The nonlinearity of the alternate 212-cps oscillator was increased beyond the design specifications during this period. No other changes were noted during this time. A copy of the data obtained from the spacecraft in orbit is shown in Figure 5-7.

Tape Recorder Subassembly Design

Design Specifications

Since the spacecraft was beyond the range of the receiving station for 90 minutes of its 95-minute orbit, it was necessary to store data. The multiplexed subcarrier oscillator tones plus a fixed reference tone were recorded on tape for 90 minutes. The tape recorder was a single-reel, continuous-loop device that recorded continuously until commanded to play back. During the playback interval of 5 minutes, the entire tape loop passed over the playback head. The recorder then reverted to the record state automatically. The *record time to playback time* ratio of the tape recorder caused the recorded tones to be transmitted at the frequencies indicated in Table 5-1. The frequencies were standard IRIG FM-FM telemetry frequencies. The reference tone, 17.5 kc, was used for airborne tape-speed compensation.

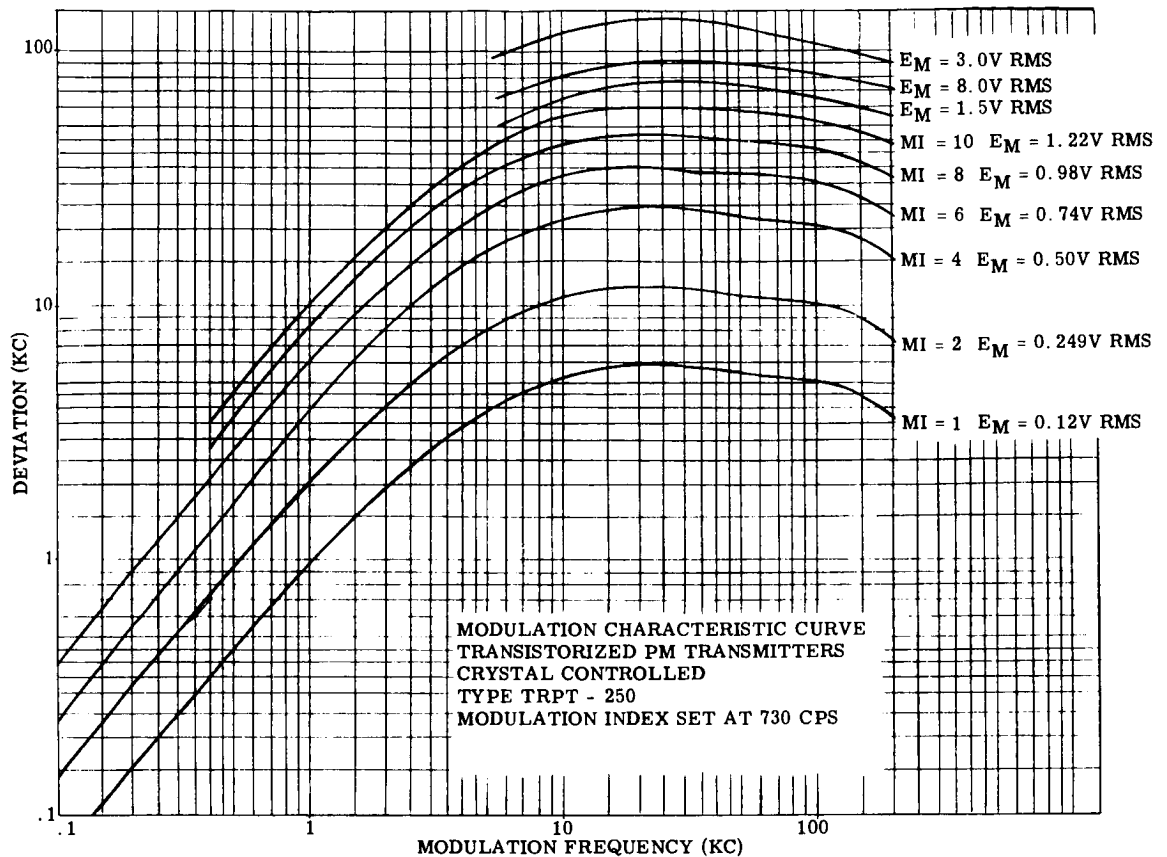


FIGURE 5-6.—Deviation sensitivity.

The following design specifications provided tape recorder storage which met all systems' requirements adequately.

1. Power requirements at 19 volts dc:
Record operation: 1.03 watt (90 minutes)
Playback operation: 1.4 watts (5 minutes)
Average orbit power: 1.05 watts (95 minutes)
2. Record-to-playback ratio: 18.35 ± 2 percent.
3. Tape Speed:
Record speed: 0.65 ips.
Playback speed: 11.9 ips.
4. Tape transport: continuous loop, single reel.
5. Motor: the motor used in the tape recorder started as a dc motor. Centrifugal force removed the starting brushes and the motor ran as a synchronous machine from an inverter power supply.
6. Tape capacity: 300 ft.

7. Wow and flutter: less than $1\frac{1}{2}$ percent peak-to-peak and less than 300 cps.
8. Input impedance: 20K.
9. Input signals:

Channel 1: Input consisted of 4-frequency-multiplexed subcarrier oscillators. Center frequencies are 164 cps, 212 cps, 294 cps, and 400 cps.

Channel 2: Input consisted of 4 frequency-multiplexed subcarrier oscillators. Center frequencies are 572 cps, 790 cps, 955 cps, and 1200 cps.

The signal level for both channels was 540 millivolts *rms*.

10. Erase head: a permanent magnet.
11. Record method: signal superimposed on a bias oscillator.
12. Output signal both channels: 1.07 volts *rms*. across a 5K load.

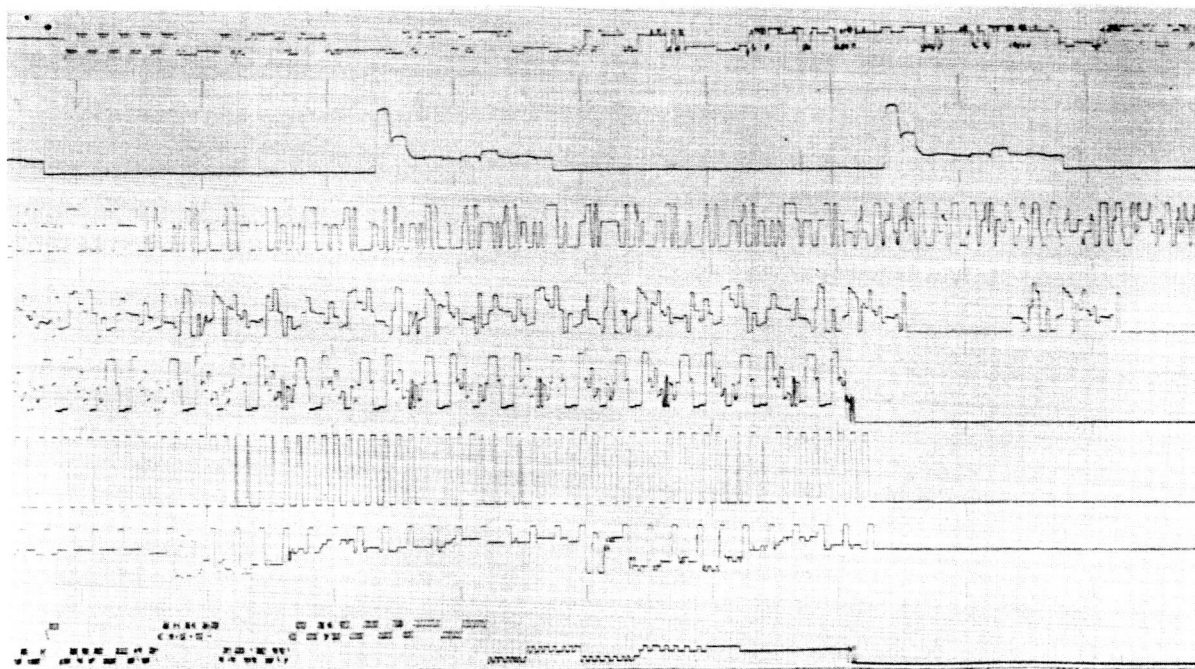


FIGURE 5-7.—Data format.

13. Amplitude modulation: less than 10 percent.
14. Harmonic distortion: less than 3 percent per subcarrier channel.
15. Frequency response:
 - Channel 1: ± 3 db from 1 kc to 8 kc.
 - Channel 2: ± 3 db from 9 kc to 24 kc.
16. Signal-to-noise: less than 30 db on either channel.
17. Weight and size: the tape recorder weight was 5.1 lb. and packaged into a container 7 inches in diameter and 3 inches high as shown in Figures 5-8 and 5-9.
18. Life: six months overall.
 - Tape: 7,000 record-playback cycles.
 - Motor: 6 months continuous operation with no more than 25 on-off cycles during this period.
19. Tape type: the continuous-loop type of transport required that the tape used be lubricated to reduce tape drag to a minimum. Tape type LR 1220 manufactured

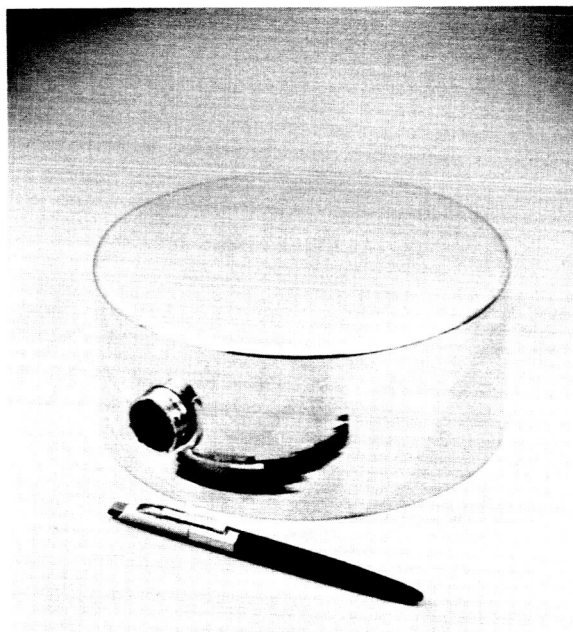


FIGURE 5-8.—Tape recorder.

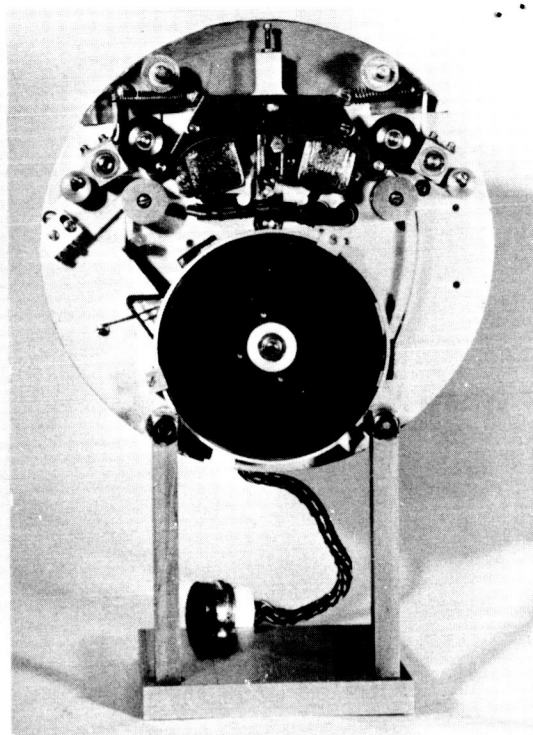
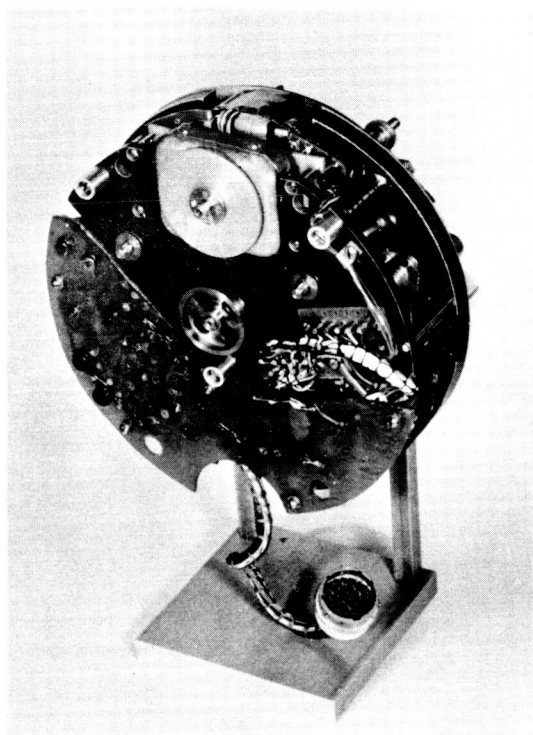


FIGURE 5-9.—Tape recorder motor, electronics, transport drive and reel.

and lubricated by Minnesota Mining and Manufacturing Company was used.

Transmitter Subassembly

The transmitter subassembly consisted of two parts: the driver and the power amplifier. Both units were initially obtained from the Vector Manufacturing Company. The driver, Figure 5-10, was capable of delivering 250 milliwatts to the power amplifier. The power amplifier was capable of 1.75 watts output.

There were two modes of operation for the transmitter subassembly, low power during 90 minutes of the orbit and high power during commanded playback of the tape recorder. The change to low power output was accomplished by switching a resistor in series with the B+ line to the exciter. The resulting reduced drive to the power amplifier provided an output of 0.3 watts. During high power transmission—lasting five minutes of each orbit—the B+ to the driver was normal and the power amplifier output was 1.75 watts.

During preliminary testing it was determined that the Vector power amplifier was incompatible with the ESCO diplexer. To correct this problem, a lumped-constant diplexer was purchased from Vector. However, the power amplifier continued to radiate spurious signals unless the VSWR was precisely adjusted. The adjustments were considered to be too critical



FIGURE 5-10.—Transmitter subassembly driver.

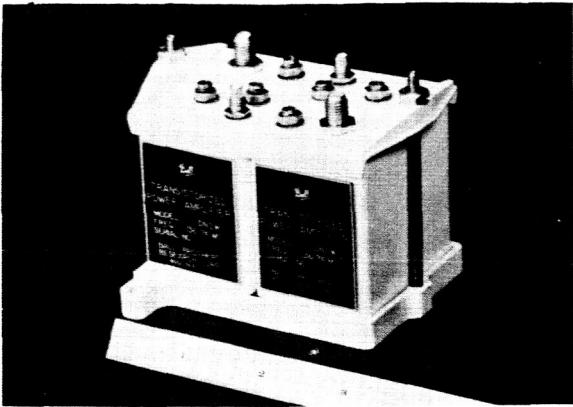


FIGURE 5-11.—Transmitter subassembly power amplifier.

and BBRC designed, fabricated, and qualified the Model DN2W shown in Figure 5-11.

RF Power Amplifier

Two Western Electric 2N1645 transistors operating in parallel provided an extremely conservative output of 1.75 watts. Broad-band RC neutralizing was incorporated to stabilize the amplifier when subjected to mismatched loads and to tuned loads such as the lumped-constant diplexer.

A pi-section input filter solved the problem of matching the driver output to the low input impedance of the 2N1645's.

The case of the 2N1645 transistors was connected to the collectors. This required the case to be electrically isolated from the chassis. The heat conduction problem was resolved by

use of a large copper output coil which had one end connected to the transistor case and the other to the chassis. The schematic of the power amplifier is shown in Figure 5-12.

Design Specifications

The transmitter subassembly with the BBRC power amplifier was qualified according to the following specifications:

Frequency: 136.74 mc ± 0.005 percent.

Power output: High power: 1.75 watts.

Low power: 0.3 watts.

Power consumption at 19 volts: High power: 4.8 watts. Low power: 2.1 watts.

Efficiency: High power: 36.5 percent.

Low power: 14.3 percent.

Supply voltage variation: 16 to 22 volts; 19 volts nominal.

Modulation: Indirect FM.

Deviation sensitivity: See Figure 5-6,

Frequency response: 10 cps to 100 cps.

Spurious output: 40 db below carrier.

Load mismatch: Spurious outputs must be less than 40 db when operated into a continuously varying VSWR from 1:1 to 2:1.

After nine months in orbit both the primary and alternate transmitter subassemblies had shown no determinable degradation in output level or significant frequency shift.

Command Subassembly

The OSO I command system was a tone system using tones which were classified

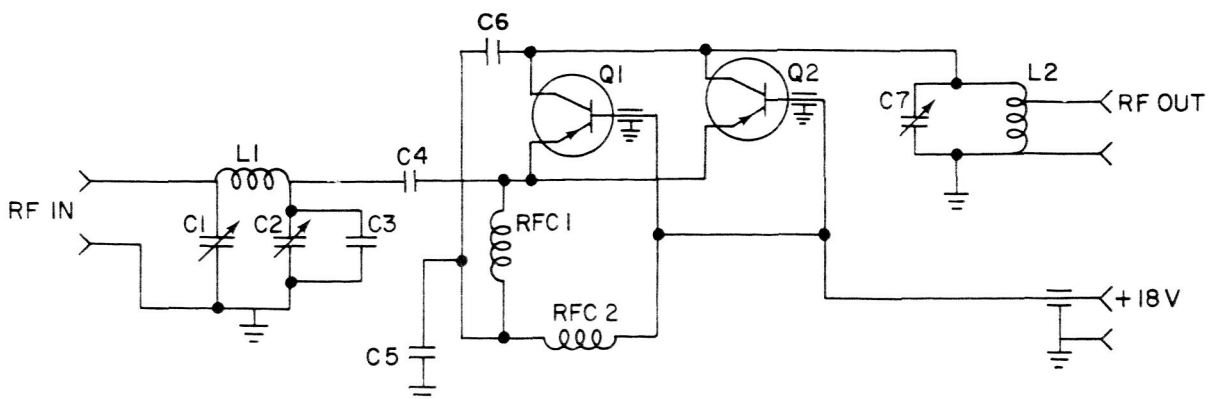


FIGURE 5-12.—OSO I power amplifier schematic.

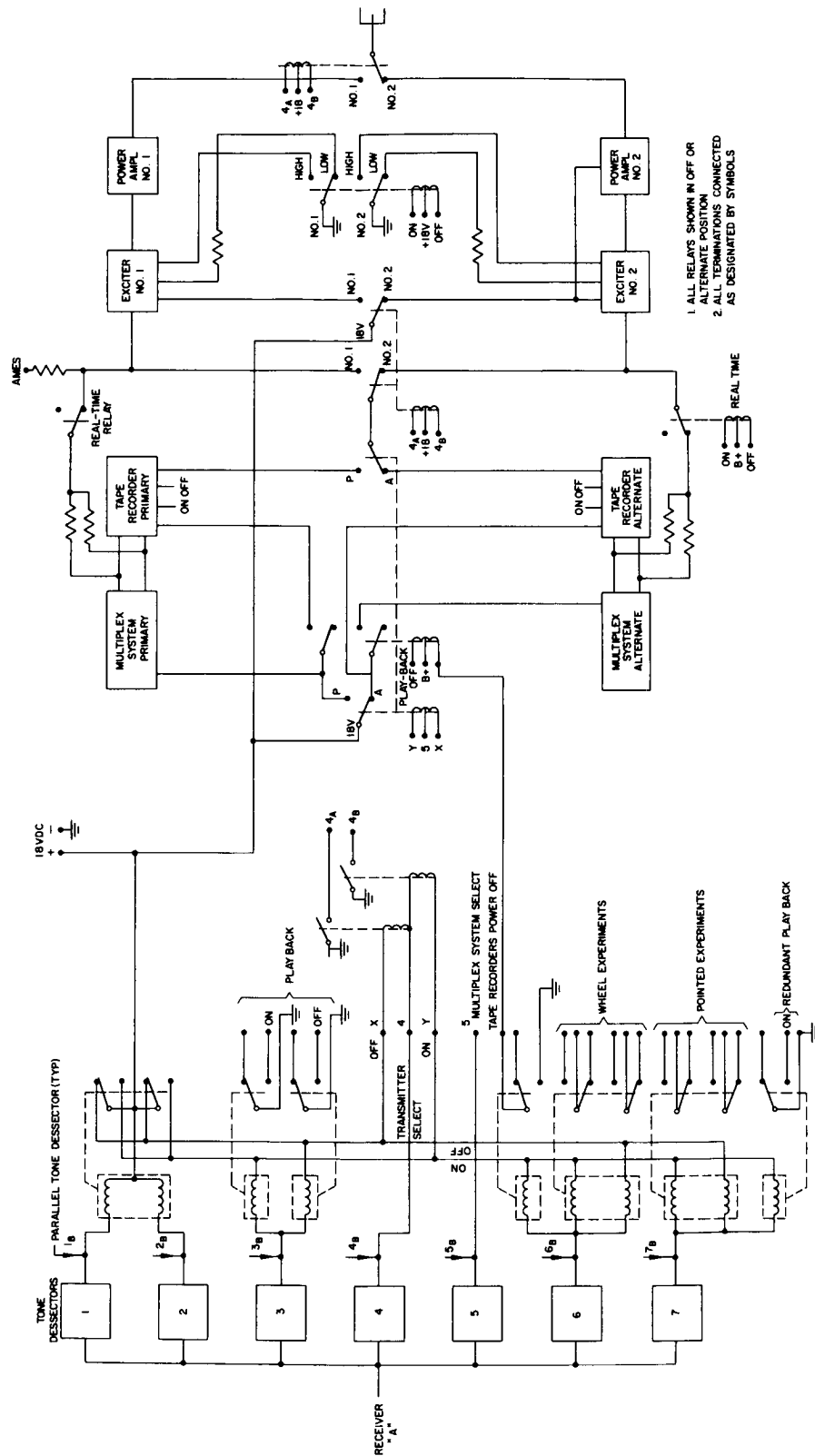


FIGURE 5-13.—Command decoder operation.

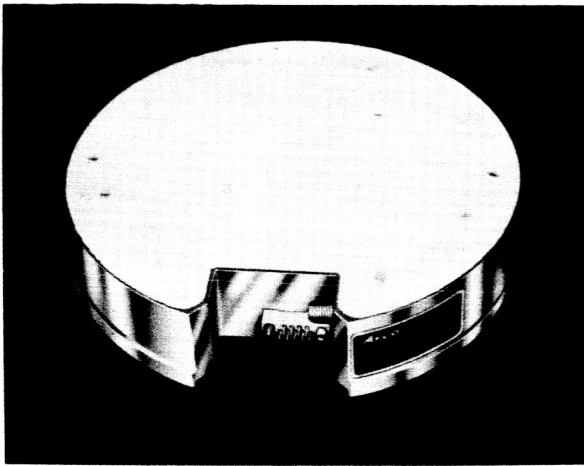


FIGURE 5-14.—OSO I receiver.

confidential. Seven tones actuated seven channels. The receiver output was filtered by highly selective bandpass filters at the input to the decoders. When the presence of the particular channel tone was detected, the relay driving circuit of the decoders was biased into conduction and connected one end of the driven relay field to ground. This was a momentary connection during the presence of the command tone.

Figure 5-13 illustrates the decoder. Its operation is as follows. Each of the tone disector channels is connected to either one end of a single relay field (channels 1 and 2), or to a pair of relay fields (channels 4 and 5), or in the case of channels 6 and 7 to three relay fields. Channels 1 and 2 are used to select the ON or OFF function by applying B+ to the proper parallel bank of relays. Channels 6 and 7 each have an additional relay to actuate for the ON condition. The channel 6 and 7 decoder relay drivers have sufficient current-carrying capacity to operate two relays in parallel.

Receiver

The receiver was procured from the AVCO Corporation. It was Model No. AD183114 (Figure 5-14) and had the following characteristics:

Frequency: 110–150 Mc.

The tuning is fixed and is set to a specified frequency at the manufacturer's plant

Modulation: AM

Input impedance: 50 ohms.

6-db bandwidth: 35 kc.

Sensitivity: $-dbm$ (75 percent mod.).

Dynamic range: 2 to 2000 microvolts minimum.

Image rejection: 80 db.

Spurious response rejection: 60 db.

Local oscillator radiation: 200 $\mu\mu w$ max.

Local oscillator stability: ± 2 kc.

I.F. stability: ± 2 kc.

Audio amplifier response: $\pm 1db$ 3 kc to 7kc.

Audio output: 50 milliwatts (5v rms across 500 ohms).

Standby power: 150 milliwatts (19 vdc source).

Power during interrogation: 250 milliwatts.

Input connector: SM male.

Decoders

The decoders were procured from AVCO Corporation. They were operated in parallel to the point of attachment to the relays (see Figure 5-15). One decoder contained the

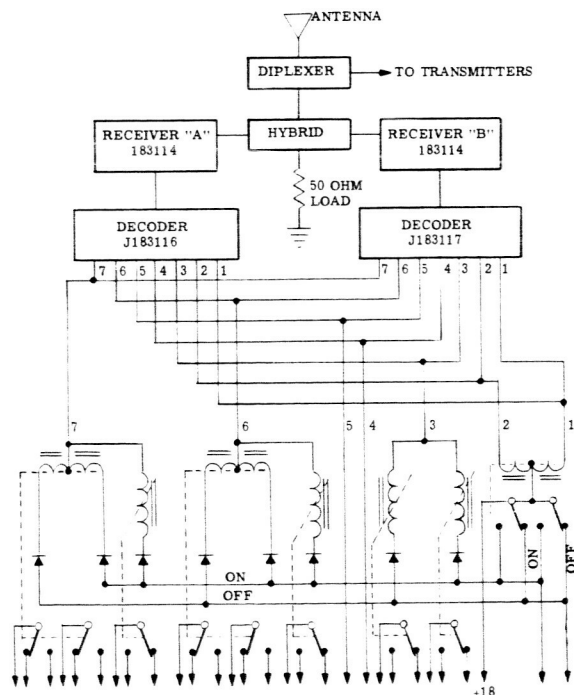


FIGURE 5-15.—Command system diagram.

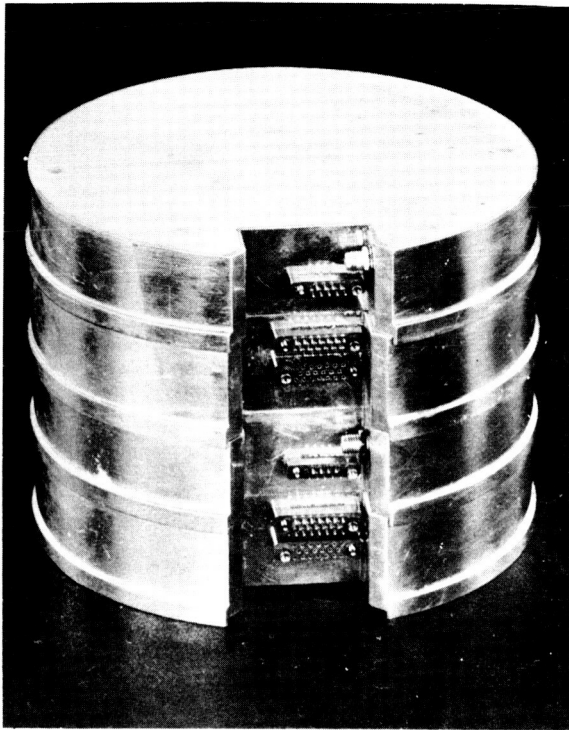


FIGURE 5-16.—OSO I receivers and decoder.

relays shown while the other had none; thus, the two part numbers.

The specifications for the decoders were as follows:

Tone frequencies: (seven, classified).

Input impedances: 500 ohms.

6-db channel bandwidth: 5 percent of tone frequency.

20-db channel bandwidth: 10 percent of tone frequency.

Audio input power: 50 mw.

Noise immunity: no channel will operate due to random noise.

Standby power: 10 microamps max. at 18 vdc.

Interrogate power: 700 mw nominal.

The decoders and receiver were mounted in a stacked arrangement as shown in Figure 5-16.

Ground Station Encoder

The encoder was compatible with the Collins 242F-2 transmitter. The command tone to be sent was selected by a pushbutton on the front of a 19-inch rack-mounted chassis.

The chassis contained seven crystal-controlled tone generators with short and long-term stabilities of 1 part in 10^4 and ± 1 percent, respectively.

The output of the encoder amplitude modulated the transmitter. The transmitter was keyed ON simultaneously with the command tone. A timer inside the transmitter turned off the RF carrier after 2 seconds.

Antenna and RF Multiplexing Subassembly

The spacecraft was stabilized with respect to the sun. Since there was no stabilization with respect to the earth, it was necessary that the radiation pattern be omnidirectional.

Three components are necessary for RF multiplexing:

1. A coaxial relay directs the RF output from one of two transmitters to the diplexer.
2. A diplexer allows the single antenna to be used for both the transmitter and receiver.
3. A hybrid is used to divide the received command power and provide isolation between the two receivers.

The two receivers and decoders were operating simultaneously. Figure 5-17 is a block diagram of the RF multiplexing system.

Specifications for the antenna and RF multiplexing system were prepared by Ball Brothers Research Corporation. The specifications required losses of less than 3 db in the

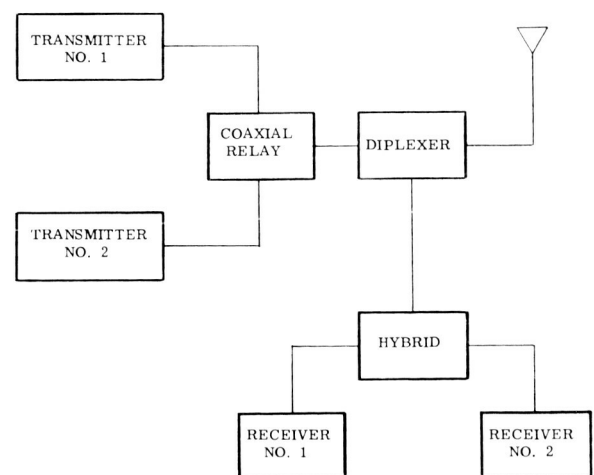


FIGURE 5-17.—OSO I RF multiplexing system.

• multiplexing system, isolation from transmitter to receiver of 40 db, isolation from receiver to receiver of 30 db, and an energy pattern within -6 db of a unity gain antenna. Small angle nulls were allowed. The entire antenna and RF multiplexing system had to weigh less than 3 lb.

Antenna

Preliminary antenna design was accomplished with a one-third scale model of the Orbiting Solar Observatory. Antenna patterns were submitted for a *two-driven-element* array and for a *three-driven-element* array. The radiation characteristics of the two arrays were similar. For simplicity and reliability, the *two-driven-element* array was chosen. Impedance matching was then accomplished using a full-scale model of OSO. Principal-plane antenna patterns were recorded and submitted for approval. The antenna patterns recorded on the BBRC antenna range are included in Appendix A. These include 18 transmitter patterns and 3 principal-plane receiver patterns.

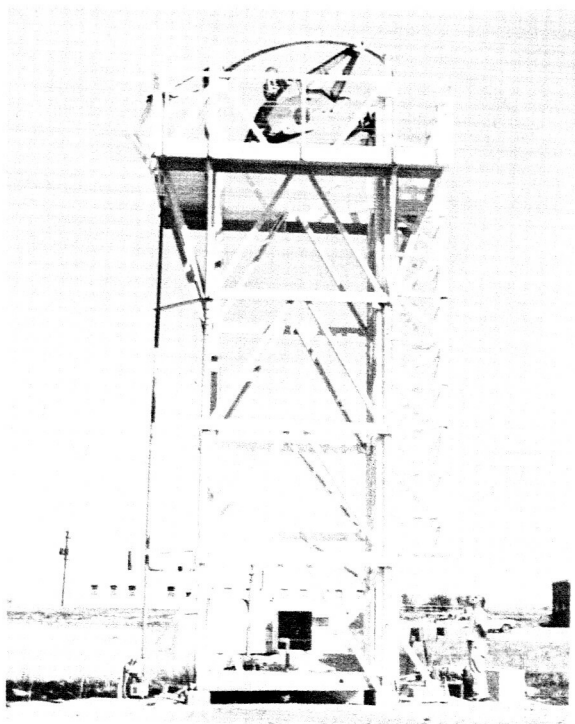


FIGURE 5-18.—Antenna tower.

A special fiberglass test fixture was used on the BBRC antenna tower. This fixture was essentially a modified gimbal which allowed the recording of antenna patterns at angular position increments of 10 degrees. The fixture and tower used for antenna pattern measurements are shown in Figures 5-18 and 5-19.

Figure 5-20 is a pictorial view of OSO I, including the antenna elements. The spacing between stubs was approximately one-half wavelength. Each antenna element consisted of a support bracket and a stub. The support brackets also supported the extendable arms of the spacecraft. Each stub was electrically connected to its respective support bracket to form a three-element array. The three elements were completely insulated from the spacecraft by the use of nylon and fiberglass epoxy material. Two of the elements were excited and the third was parasitic. The element excitation was applied at the point where the support bracket connected to the spacecraft arm. The electrical length of each element was approximately one-half wavelength.

It was decided that the spacecraft transmitter should radiate during the launch for tracking purposes. Since the arms were folded down during this phase, the antenna configuration was much different from that of the orbital position and the resulting radiation pattern was severely distorted. This was accompanied by a high loss of radiated power. In spite of this distortion and loss of power, adequate gain was available for tracking and the mismatch of impedances caused no harm to the RF power amplifiers.

The antenna system was utilized for reception as well as transmission with the two frequencies differing by 12 percent. Impedance matching was accomplished by single-stub matching techniques. Single-stub matching is a narrow-band method, but it was possible to obtain a VSWR of less than 1.5:1 at the transmitter frequency and less than 2:1 at the receiver frequency.

Diplexer

A helical resonator bandpass diplexer was designed by ESCO. During system integration, it was found that the Vector solid-state

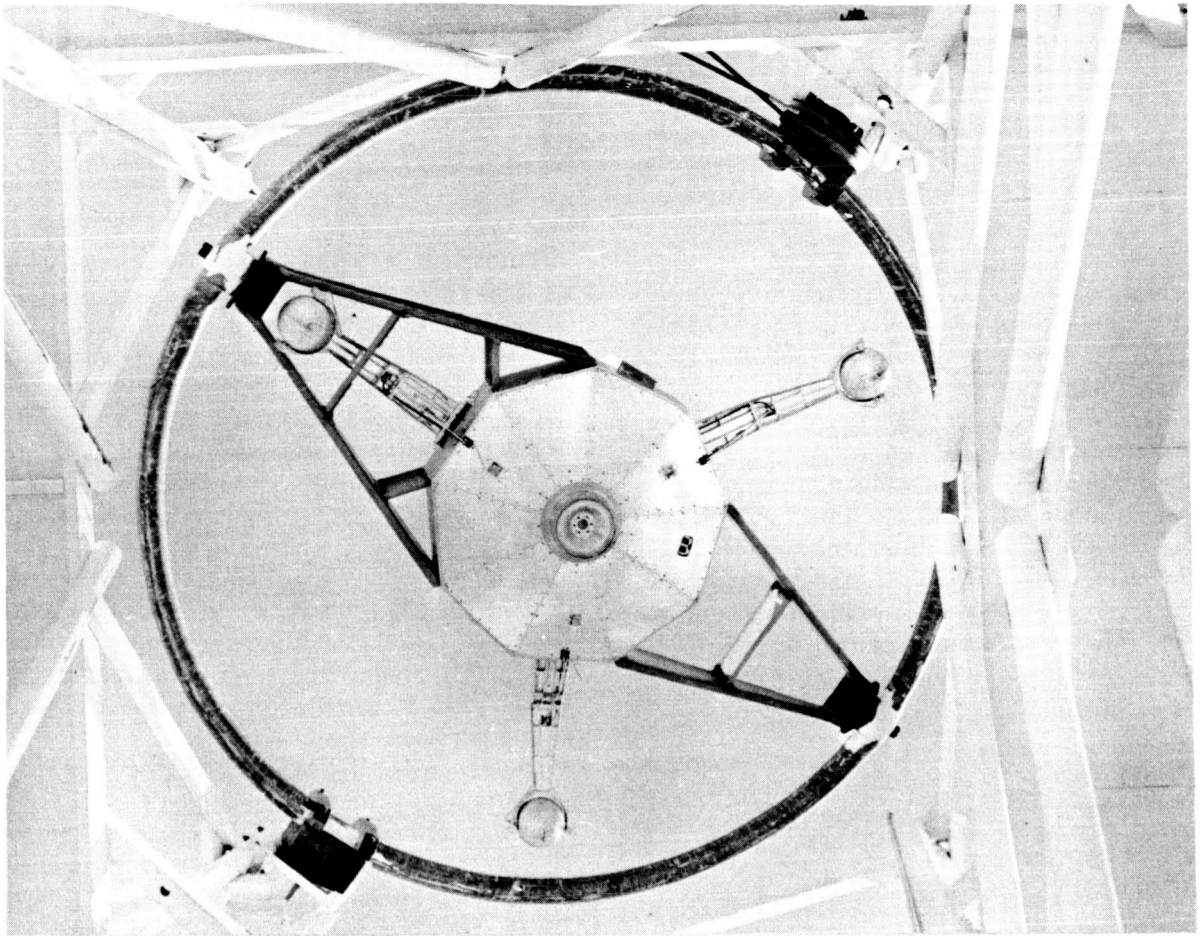


FIGURE 5-19.—OSO I mounted in the antenna fixture.

RF amplifiers occasionally radiated spurious signals when working into the high-Q load presented by the diplexer.

This problem resulted in the procurement from the RF amplifier manufacturer (Vector) a lumped-constant diplexer using high and low pass filters. The lower cutoff frequency of the lumped-constant diplexer as opposed to the helical resonator diplexer reduced the spurious signal generation, however, it was necessary to use a different power amplifier to completely overcome the spurious signal problem.

The characteristics of the lumped-constant diplexer were as follows:

1. An isolation of greater than 40 db.
2. An insertion loss of less than 1 db at the transmitter frequency and less than 2 db at the receiver frequency.

A photograph of this diplexer is shown in Figure 5-21.

Hybrid

A lumped-constant hybrid circulator was designed by ESCO. No problems were encountered with this unit.

The characteristics of the hybrid were as follows:

1. An isolation of greater than 35 db between receivers.
2. An insertion loss of less than 1.5 db.
3. A power split of within 0.2 db.

Figure 5-22 is an illustration of this unit.

Coaxial Relay

One coaxial relay was used to connect the desired telemetry transmitter to the antenna.

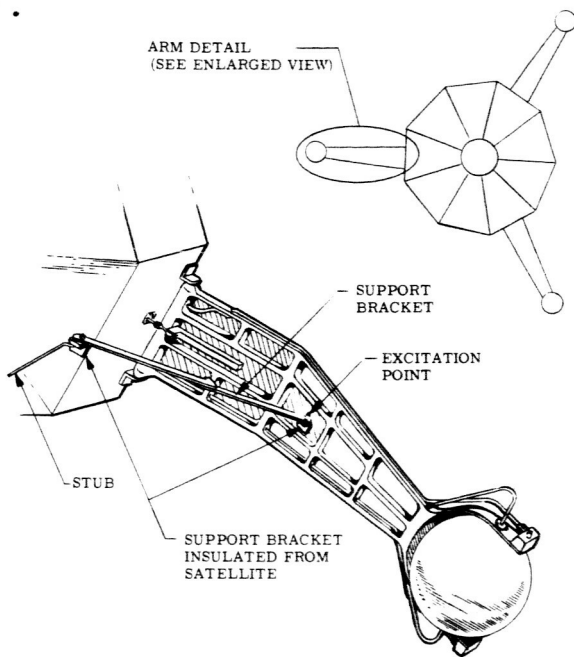


FIGURE 5-20.—OSO I antenna elements.

Flight relays passing all specifications were selected from a group of twelve relays supplied by the subcontractor.

The relay specifications were as follows:

Insertion loss: less than 0.25 db.

Mismatch: less than 1.2 VSWR

Crosstalk: no less than 40 db.

Figure 5-23 is an illustration of this unit.

OPERATIONS

General

The worldwide Minitrack network was responsible for tracking and for receiving data from the Orbiting Solar Observatory.** The OSO subsatellite plot is shown in the world map, Figure 5-24.

**NASA-GSFC Operations Plan 1-62 Orbiting Solar Observatory.

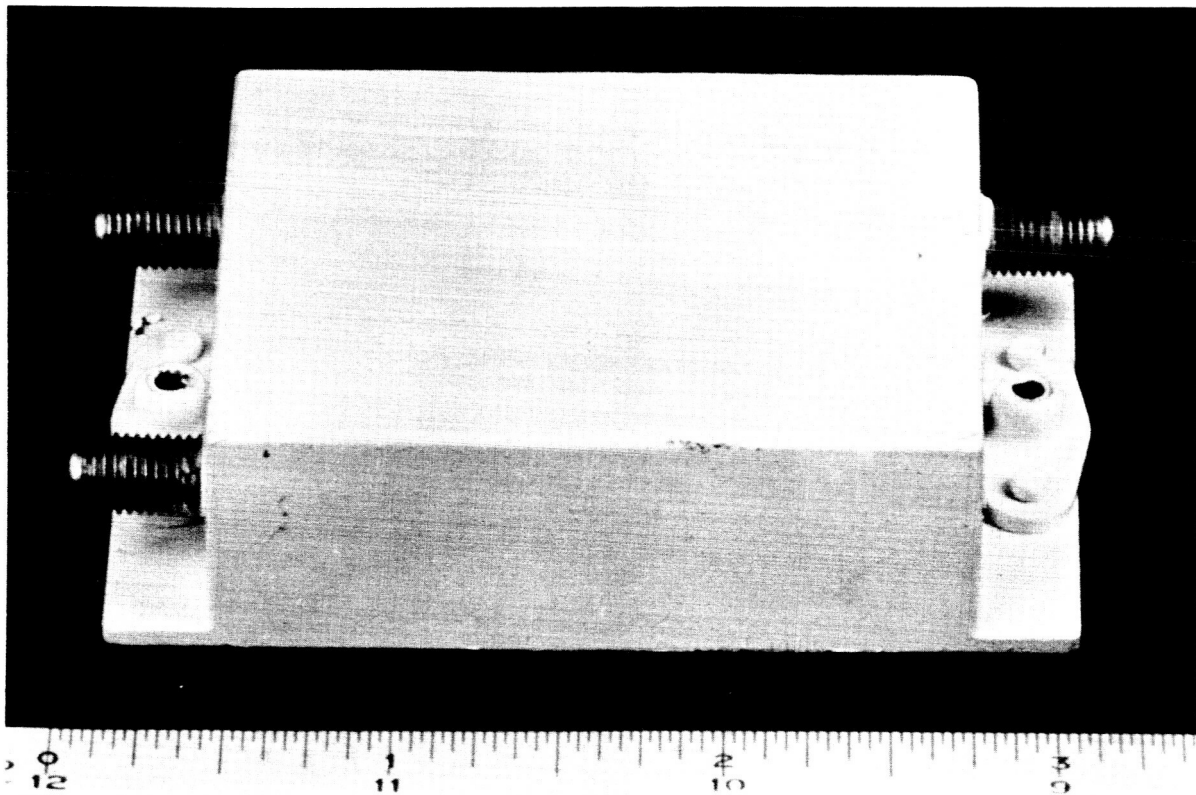


FIGURE 5-21.—OSO I diplexer.

ORBITING SOLAR OBSERVATORY

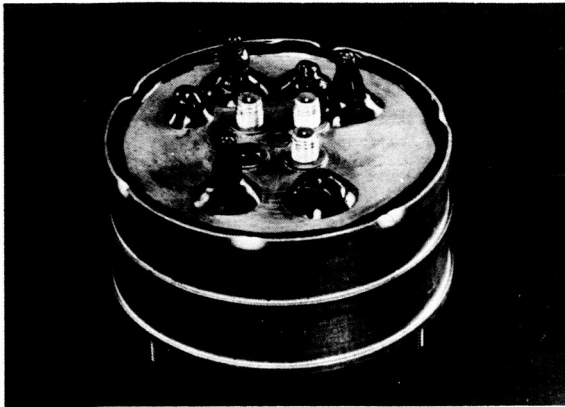


FIGURE 5-22.—OSO I hybrid.

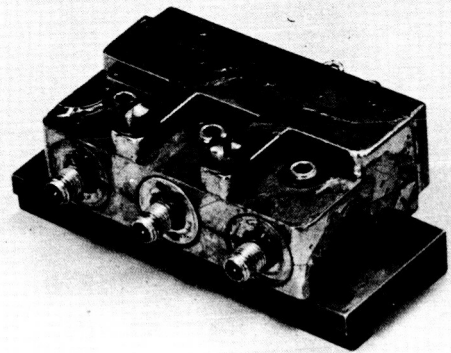


FIGURE 5-23.—OSO I coaxial relay.

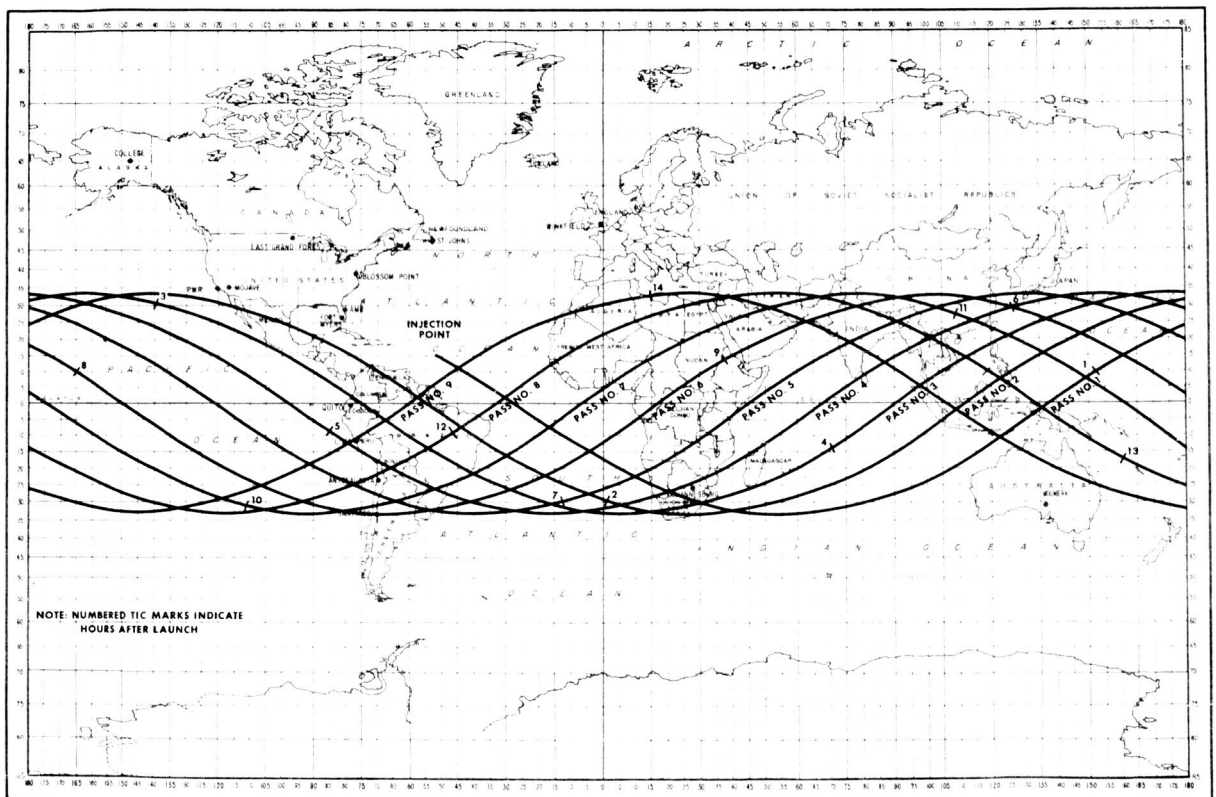


FIGURE 5-24.—Orbiting solar observatory sub-satellite plot: 300 nautical miles-inclination 33° nominal orbit.

• • Telemetry recordings were required from the stations listed below:

Primary

Antofagasta, Chile
Blossom Point, Maryland
Fort Myers, Florida
Lima, Peru
Quito, Ecuador
Santiago, Chile

Secondary

Mojave, California
Woomera, Australia
Johannesburg, South Africa (early orbit only)

NOTE: Secondary telemetry acquisition stations were used only during early orbits or during those occasions where conflicts developed and a primary station was unavailable to command and record the telemetered data.

There were two types of data transmitted by the Orbiting Solar Observatory. These were real-time and stored data. Stored data, recorded in the spacecraft during one transit around the earth, were played back and transmitted upon command once per orbit by one of the above stations. Recovery of real-time telemetry data was not required at any of the stations until both of the spacecraft recorders failed to function.

In order that the status of the spacecraft could be evaluated on a current basis, a special quick-look procedure was established for the analysis of the "housekeeping" channel.

During the first two weeks after launch, Fort Myers was the primary station, and all housekeeping data received there were also reduced there. The outlying stations could read only chart deflection of the selected functions listed below and TWX this information to Fort Myers.

After two weeks, GSFC at Greenbelt received all quick-look housekeeping data in the form of

centimeter deflections. After conversion to functions, the 12 channels were TWX'd to Boulder daily.

The following functions indicated the status of the spacecraft on a quick-look basis:

<i>Channel</i>	<i>Parameter</i>
2	Battery temperature.
5	Calibrate 0 volts dc.
6	Calibrate 2.5 volts dc.
7	Calibrate 5.0 volts dc.
8	Spacecraft voltage monitor.
34	Elevation positioning accuracy read-out
35	Azimuth positioning accuracy readout.
36	+ 5.0 volts dc calibration.
37	2.5 volts dc calibration.
38	0 volts dc calibration.
40	Azimuth positioning power monitor.
41	Elevation positioning power monitor.

Launch and Early Operation

The first station to command the Orbiting Solar Observatory to play back its recorded data was the Fort Myers Minitrack station. This station was provided with special tape-speed compensation and diversity-combining equipment to provide a detailed quick-look analysis of the spacecraft and experiment status immediately after launch. BBRC personnel and the GSFC spacecraft controller were present at this station during the first two weeks of orbit to monitor the spacecraft operation and to advise a course of action in case of a spacecraft emergency.

The stations listed in Table 5-4 were able to receive telemetry signals in the order listed and in the manner indicated during the launch and early orbit period. They were equipped to make a Sanborn recording in real-time from one antenna polarization and TWX PAM deflections (in centimeters) to the Fort Myers station.

TABLE 5-4.—OSO I Readout and Command Schedule for the First 19 Hours After Launch

Start Time	Station	Length of Pass (Min.)	Max. Elev. (deg.)	Slant km	Range S-Mi	Type Command	Required to record telemetry	Type Data
T+0hr. 19'	JOBURG	12	51	691	429	None	Yes	*R-T
T+1hr. 18'	MOJAVE	12	49	721	448	None	Yes	R-T
T+1hr. 27'	FTMYRS	11	32	966	600	Playback	Yes	†REC
T+2hr. 01'	JOBURG	11	30	999	620	None	Yes	R-T
T+2hr. 59'	MOJAVE	11	20	1321	820	None	Yes	R-T
T+3hr. 13'	QUITOE	12	67	594	369	Playback	Yes	R-T
T+3hr. 15'	LIMAPU	11	22	1242	772	None	Yes	R-T
T+3hr. 42'	JOBURG	12	40	800	497	None	No	
T+4hr. 55'	QUITOE	10	15	1539	956	None	No	
T+4hr. 56'	LIMAPU	12	49	706	439	None	No	
T+4hr. 59'	AGASTA	12	54	667	414	Playback	Yes	REC
T+5hr. 01'	SNTAGO	12	16	1468	912	None	No	
T+5hr. 23'	JOBURG	12	61	614	382	None	No	
T+5hr. 39'	LIMAPU	09	11	1768	1098	None	No	
T+5hr. 40'	AGASTA	12	37	851	529	None	No	
T+6hr. 41'	SNTAGO	12	50	697	433	Playback	Yes	REC
T+7hr. 05'	JOBURG	10	13	1623	1008	None	No	
T+8hr. 22'	SNTAGO	12	86	551	342	Playback	Yes	REC
T+8hr. 23'	AGASTA	11	22	1212	753	None	No	
T+10hr. 03'	SNTAGO	12	65	597	371	Playback	Yes	REC
T+10hr. 04'	AGASTA	12	29	1013	629	None	No	
T+11hr. 45'	AGASTA	12	74	563	350	Playback	Yes	REC
T+11hr. 45'	SNTAGO	11	25	1114	692	None	No	
T+11hr. 46'	LIMAPU	11	12	1653	1027	None	No	
T+13hr. 27'	AGASTA	10	17	1394	866	None	No	
T+13hr. 27'	LIMAPU	11	56	641	398	Playback	Yes	REC
T+13hr. 29'	QUITOE	10	14	1548	962	None	No	
T+14hr. 33'	OOMERA	09	13	1608	999	None	No	
T+15hr. 08'	LIMAPU	11	17	1402	871	None	No	
T+15hr. 09'	QUITOE	12	75	556	345	Playback	Yes	REC
T+16hr. 12'	OOMERA	12	49	709	441	None	No	
T+16hr. 54'	FTMYRS	11	20	1279	795	Playback	Yes	REC
T+17hr. 53'	OOMERA	13	69	587	365	None	No	
T+18hr. 34'	FTMYRS	12	73	574	357	Playback	Yes	REC
T+18hr. 37'	BPOINT	11	21	1257	781	None	Yes	REC

*R-T—Real Time.

†REC—Recorded.

Normal Operations

The normal operations started approximately two weeks after launch. All stations assigned command and telemetry acquisition responsibilities during this period were expected to record a maximum amount of stored data during the playback period for which they were assigned responsibility. Each station, in addition, was required to submit quick-look information (Sanborn deflections in centimeters) to GSFC at Greenbelt where the housekeeping channels were reduced and TWX'ed to Boulder for analysis.

DATA REDUCTION

Data reduction consisted of editing and analyzing 52 PAM housekeeping channels.

Key Channels—Quick-Look Reduction

Certain of the 52 channels contained the key performance information of the spacecraft. Each receiving station was required to forward centimeters deflection of these channels to the Operations Control Center at GSFC. After TWX'ed the function status to Boulder on a conversion to the function, the Control Center daily basis.

The key channels are:

- Battery No. 2 temperature.
- Spacecraft voltage monitor.
- Spin gas pressure.
- Elevation readout.
- Azimuth readout.
- Pitch readout.
- Azimuth power monitor.
- Elevation power monitor.

Semi-Automatic Comprehensive Reduction of Housekeeping Channels

Magnetic tapes supplied by GSFC were used to generate paper strip charts. Sanborn pen and Heiland Visicorder records were generated simultaneously. The Visicorder records were used for housekeeping data reduction and the Sanborn records were used to qualitatively determine the condition of experiments.

Oscar K Readout

Oscillographic records were read by the Bensen Lehrner, Oscar K, oscillogram semi-automatic reader. The output of the Oscar K reader was automatically recorded on an electrotyper. Recorded data included time, channel number, frame number, and data.

The data were recorded as a decimal digit from 0.00 to 5.00 volts. These data were not in a form for graphical presentation in that scalar mathematics had to be performed to convert voltages to quantities of temperature, pressure, angular degrees, etc.

An additional output of the Oscar K provided an octad format which was recorded on punched paper tape.

Computer Reduction

The CDC 160 computer with a Fortran program was used to process the octad format tape from the Oscar K. Special computer program tapes were prepared which contained coefficients of calibration curves. The data tape and the calibration tape were fed to the computer. The computer output tape was fed to a Frieden Flexowriter. The Flexowriter typed out the magnitude of eleven channels in eleven rows across the page. Ninety data samples per channel were taken per orbit. The

Flexowriter typed out data sample numbers 1 through 90 down the page. The result of this tabulation then was "Function Magnitude versus Orbit Position," since sample number and orbit position could be correlated.

By editing the oscillographic records, thirty-three of the fifty-two channels were selected for computer processing in each orbit. The process was repeated three times with three sets of data and calibration tapes. The computer required 15 minutes to process one orbit.

OVERALL PERFORMANCE

General

The FM-AM telemetry system of the spacecraft performed well for over 1000 orbits. The signals received from the spacecraft at the receiving station during tape recorder playback were at least 20 db above threshold and free of noise resulting in data of excellent quality. A record of a data transmission received at Fort Myers, Florida, receiving station compared favorably with one made in the laboratory during spacecraft checkout. It was possible to receive real-time data transmission from the spacecraft in Boulder, Colorado. The signal was of sufficient quality that information concerning the operation of the spacecraft and the experiments was routinely being gathered at Boulder.

The communications system continued to give excellent performance through the tenth week during which time 100 percent of the data recorded was usable. An average of 10 to 15 percent of the data could not be recorded due to spurious commands received by the spacecraft as well as minor ground station problems.

RF System

The RF systems (both primary and alternate) continued to operate satisfactorily. At the time of this writing there had been no significant change in transmitter frequencies or receiver signal-to-noise ratio since launch.

The slant range†† versus the available signal level at the receiver input terminals is shown in Figure 5-25. As can be seen, the combined

††Taken from NASA-GSFC Operations Plan 1-62, Orbiting Solar Observatory (S-16).

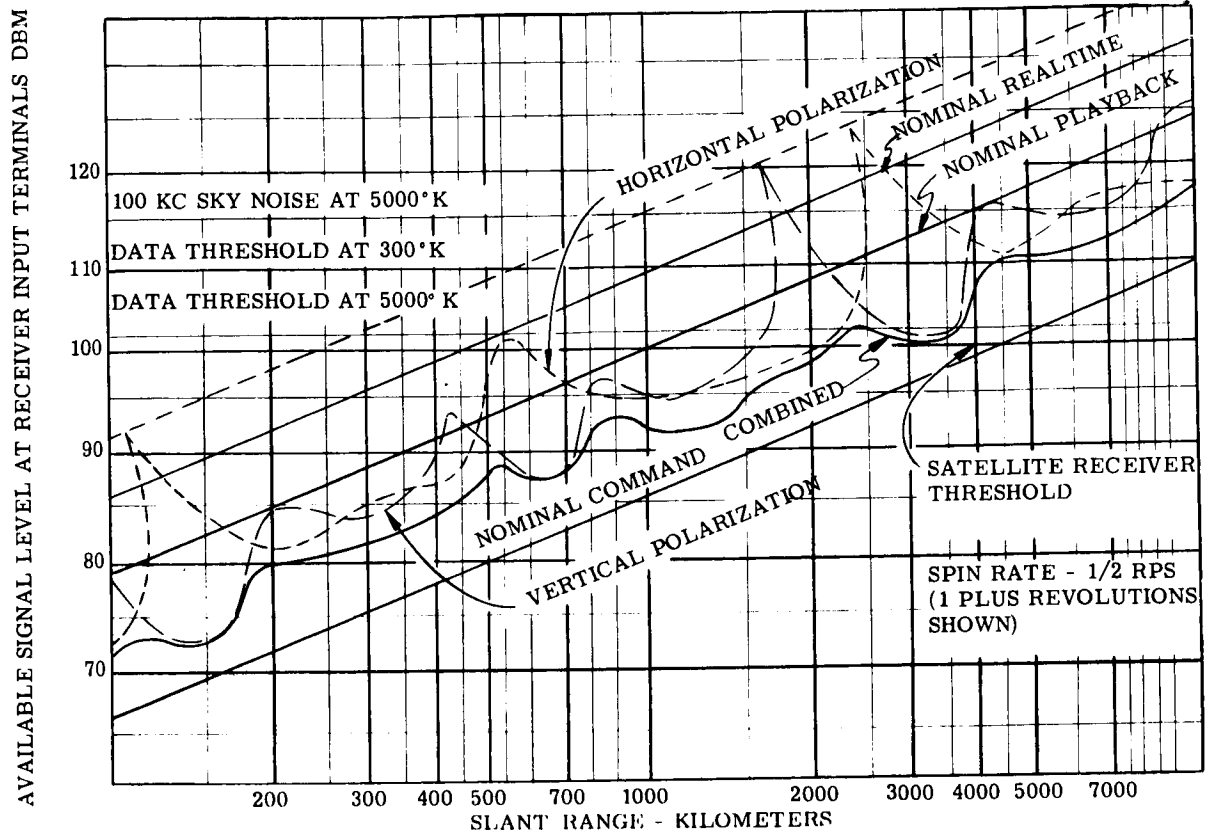


FIGURE 5-25.—Comparison of slant range versus signal level.

polarization curve lay within the data threshold at 300°K and 5000°K up to a slant range of 2000 miles.

FM Multiplex System

No indication of drift in either of the sub-carrier oscillator packages was received during the first nine months of orbit. Transmitter power and frequency remained well within specification. Over half of the playback transmissions over the tracking stations had a signal strength of -100 dbm, or better. Only 1 percent had signal strengths of less than the threshold value of -120 dbm.

Data Storage System

Prior to the failure of the primary tape recorder, the high frequency channel amplitude decreased from a normal level during orbit 364

to nearly zero during orbit 372. The low-frequency channel was not affected. The decrease in amplitude could have been caused by oxide buildup on the playback head. The primary tape recorder failed completely after 383 orbits. The nature of the failure indicated that the spurious commanding received over North Africa had simply worn out the starting mechanism of the motor by many turn-on/turn-off cycles. The recorder performance in orbit was comparable to the performance in the laboratory until the time it failed to start. The loss of the primary tape recorder necessitated the use of the alternate system.

The alternate tape recorder failed to provide playback modulation during orbit 1038. Indications were that the basic recorder, timer, and motor continued to operate normally. Data obtained during orbits 3472 and 3489 indicated

that the motor and driver were still operating properly. One of the following items apparently failed:

1. Bias oscillator.
2. Latching relay.
3. 15-volt regulator.
4. Mylar belt (tape drive).

Command System

Soon after the launch of the spacecraft, it became apparent that spurious commands were being received when the spacecraft was between 20° and 30° east longitude and between 15° and 35° latitude (see Figure 5-24). The specific source of the spurious command signals was not determined. The spurious commanding could

have been caused by a modulated carrier or by two carriers heterodyning and generating the exact command tones.

GSFC transported a transmitter to the particular region of North Africa over which the signals were received. The transmitter was modulated with the ON command tone. This modulation desensitized the receiver and decoder to any OFF commands sent by the source of interference. In this way, the sequence of ON-OFF commanding which was disastrous to the primary tape recorder was eliminated.

No problems other than spurious commanding were apparent with the command system. All commands were received properly by the spacecraft.

Chapter 6

POWER SUPPLY

INTRODUCTION

Electrical power for the spacecraft was converted from sunlight by a sun-oriented array of silicon solar cells. Part of this power was used to charge a storage battery which stabilized the voltage, supplied peak demands, and provided power during the dark part of the orbit. The solar-cell array and battery constituted the power supply.

POWER AND TEMPERATURE ESTIMATES

It was estimated that at the nominal voltage of 19 volts, the power requirements would be:

	Day (Watts)	Night (Watts)
Control system.....	6.6	2.8
Communications system.....	4.5	4.5
Experimental instruments.....	6.5	4.3
Total.....	17.6	11.6

The thermal analysis (Chapter 7) predict a solar-cell array operating temperature of about 70° C and a battery temperature of about 10° C.

SOLAR CELL ARRAY

The solar-cell array had to provide 17.6 watts day power, and also had to provide enough battery charge to support the 11.6-watt night load. Assuming that the battery efficiency was 75 percent on an ampere-hour basis in a 76 percent sunlight orbit, the solar-cell array had to deliver 22.5 watts. In a 63 percent sunlight orbit the solar-cell array had to deliver 26.7 watts.

The array comprised 1872 silicon solar cells, Hoffman Type 120CG. Each cell had a 0.006-inch-thick cover glass with an ultraviolet re-

flective coating on one side and a red antireflective coating on the other. The purpose of these coatings was to transmit the light to which the cell was sensitive (to increase power output) and reject the light to which it was insensitive (to decrease temperature).

The cells were soldered together into modules of four and five cells each, and the modules were interwired to form strings of 52 cells in series. There were 36 such strings.

Solar-cell modules occasionally fail by open-circuiting, but rarely by shortcircuiting. If modules are connected in simple series strings, a failure of one module takes an entire string out of the circuit. However, if parallel crosswiring is provided between series strings, the failure of a single module is much less serious. The current that should flow through that module is carried by those in parallel with it, in addition to their own normal current. In this particular array, if 6 or more series strings were cross-connected the failure of a module would have cost only the amount of power normally contributed by that module.

If all the series strings had been crossconnected, then a short circuit across part of one string (caused, say, by meteorite damage) would have wiped out the entire array. Therefore, this array was divided into five sections, three containing 6 series strings and two containing 9 series strings. The strings within each section were crosswired, and each section had its own blocking diodes.

Current-voltage curves for the completed array are shown in Figure 6-1. Curve A is an experimental curve taken in sunlight in Boulder, Colorado. Curve C is this experimental curve corrected to the expected orbital conditions. Curve C shows that the array will deliver 29.4 watts at 19 volts at 70° C in space sunlight.

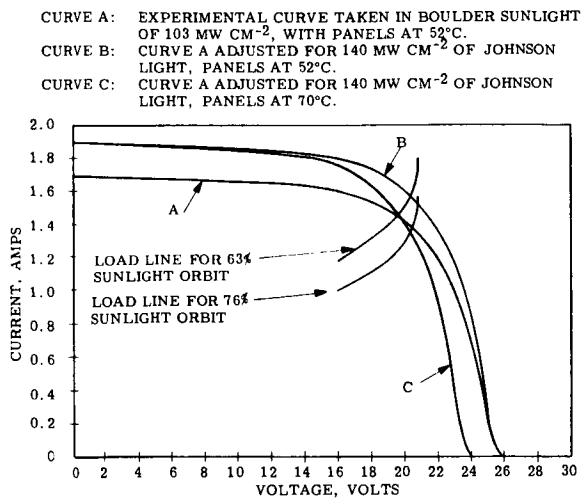


FIGURE 6-1.—Current-voltage curves for solar cell array.

BATTERY

At the time this spacecraft was designed, the most common reasons for the failure of Nicad cells were degradation of the separator material due to overheating, and case bulging due to overcharging. Therefore, it was decided to use a battery with such great overcharge capacity that overcharging was impossible, and to pot the cells into cavities in a solid metal block to provide mechanical reinforcement and excellent heat conduction.

Since the solar-cell array produced about 1.55 amperes at 19 volts, and the control system uses 0.35 amperes in keeping the array oriented, the battery had to be able to absorb a continuous charge current of at least 1.2 amperes.

The cell selected for use in this battery was a Sonotone Type F hermetically sealed Nicad cell, with two plates and a polypropylene separator. This cell had a nominal capacity of 5.6 ampere-hours above 1.0 volts, and a measured capacity of at least 4.2 ampere-hours above 1.2 volts. It could stand continuous overcharge at 0.45 amperes with a maximum voltage of 1.48 volts.

The battery consisted of 42 cells, arranged in 3 series strings of 14 cells each. The battery capacity was 16.8 ampere-hours above 14.0 volts and 12.6 ampere hours above 16.8 volts. It could stand continuous overcharge 1.35

amperes with a maximum voltage of 20.7 volts.

In normal operation, the battery was cycled to a discharge depth of about 2 percent. It was thought that this would provide excellent cyclic life. If the solar-cell array had failed completely, the battery alone could have powered the spacecraft for about 10 orbits.

POWER MARGIN

Assuming that the voltage swing during any one orbit was negligibly small, it was possible to plot approximate load lines for the solar-cell array. This has been done in Figure 6-1. The load lines curve upward because the battery efficiency decreases radically as the voltage rises above the normal full-charge voltage.

In Figure 6-2, the data of Figure 6-1 are replotted to show the excess current available from the solar-cell array as a function of operating voltage, for four different operating conditions. These curves show that even if our estimate of solar-cell performance were 25 percent high, or if the array were degraded by 20 percent, or if our estimate of the load were 20 percent low, the operating voltage would still have been above 18 volts. Further, no matter how far in error the estimates may have been, the operating voltage would not have risen above 20.8 volts.

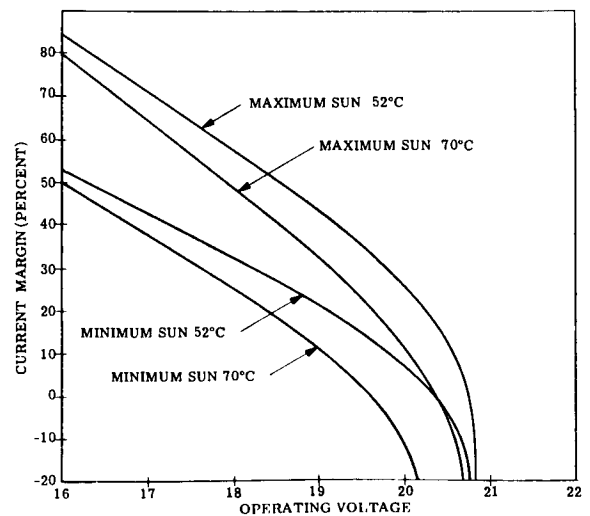


FIGURE 6-2.—Power margin curves.

The day power switch, as described in detail in Chapter 4, was provided to remove power from those circuits and experiments not required at night.

The voltage regulators, also described in Chapter 4, were provided for those spacecraft functions critical to supply voltage variations such as the servo amplifiers and most of the spacecraft instrumentation circuitry.

Fuses were installed in series with the power lines to each of the experiment and spacecraft systems which could fail without destroying the usefulness of the remaining systems. The fusing was set at a level such that a drastic overcurrent demand was required to remove the circuit from the supply. The fuses were derated for vacuum use based upon actual vacuum test data.

The pointed and wheel experiments could have been removed from the power lines by command should the power supply capability have been reduced, or if a malfunction had occurred that was insufficient to blow the fuse.

POWER SUPPLY PERFORMANCE

Preflight and Initial Orbital Results

The actual power usage for the spacecraft, using the last preflight checkout data and the data measured in orbit, was as follows:

	Day (watts)	Night (watts)
Control electronics	6.2	1.2
Control drive	*5.5	0
Communications	4.1	4.1
Experiments	7.2	3.4
Total	23.0	8.7

*3.5 watts azimuth, 2 watts elevation drive.

Test results from battery charge cycling indicated an 80 percent charge efficiency. Thus, the array was actually required to provide 23 watts day power and enough additional charge to maintain the 8.7-watt night load. For a minimum sunlight orbit, the array had to

produce 29.4 watts, and for a maximum sunlight, 26.4 watts.

The array temperature in orbit was 60°C, 10°C lower than predicted. This reduction in operating temperature allowed the array to produce about four percent more current than predicted so that the estimated initial orbital output of the array was 30.6 watts. Because of this, the minimum sunlight orbit power margin was about 4 percent and the maximum sunlight orbit margin 16 percent.

The reason for the narrow margins was that more control system drive power was required than was originally estimated. The anticipated drive was 2.5 watts and the actual drive was 5.5 watts. Had the anticipated drive values and array temperature been realized, the minimum sunlight orbit power margin would have been 11 percent and the maximum sunlight orbit power margin 27 percent.

Long Term Orbital Results

Voltage measurements made during the first four months of orbital life indicated that the power system was performing as expected and that the array output was sufficient to maintain the entire observatory load.

Measurements of battery voltage were made during orbits 3500 to 3700 about four months after the 9 July 1962 high-altitude nuclear device explosion which caused considerable electron flux intensification at the 300-nautical-mile altitude of the spacecraft. It was determined at that time that the spacecraft array could not supply the required 23 watts day power. However, the array was able to charge the batteries with the 3.4-watt wheel experiment load commanded off. If the following assumptions are made it can be concluded that the array was producing about 21.3 watts, and thus that the array had degraded about 30 percent in the four-month period that the radiation had existed.

Array temperature was still 60°C.

The loads were the same as when the spacecraft was launched.

There was no micrometeorite damage to the panel.

The original 30.6-watt array output estimate was accurate.

Half of the wheel experiment load could have been handled.

There were other considerations relating to the degradation determination outlined above. One of these was that there was some probability that, due to brush material buildup on the commutator of the azimuth torque motor, the torque gain of the motor may have been reduced. This could have had an appreciable effect on degradation, but it was estimated that it caused no greater than a 20 percent increase (0.7 watts) in required azimuth pointing power.

A further consideration was that the pointing power varied over wide limits due to the spring effect of the elevation gimbal flexible leads against which the elevation torque motor had to drive. This effect caused a variation in the elevation drive power from 0 to 6 watts with the average being the 2 watts assumed in the previous discussion. The 30 percent degradation figure arrived at above then could have been as low as 15 percent or as high as 36 percent; but 30 percent is the most probable figure.

Solution of the Degradation Problem . .

In future spacecraft of the same type and mission a more satisfactory answer to the problem of degradation would be one of the following:

1. N on P silicon cells can be used to derive about the same space sun output as the OSO I array and should have a degradation figure low enough (about 11 percent) to be useful for a six-month lifetime. This solution will undoubtedly be the one used.

2. The other approach would be to use higher efficiency P on N cells, to produce as much power as is practical for the given area, anticipate the degradation, and plan to selectively command experiments off when the array can no longer handle the full load.

With no increase in panel area being considered, the first solution has an experiment power limit of about 9 watts (averaged over the orbit) assuming the present basic power requirements exist. If this experiment power is exceeded, there will be no choice other than to increase the size of the array or to reduce the six-month life expectation for full-time experiment utilization.

Chapter 7

THERMAL CONTROL

INTRODUCTION

In the Orbiting Solar Observatory it was important that temperatures be maintained within specific limits in those sections of the spacecraft containing temperature sensitive equipment.

The efficiency of the silicon solar cells providing electrical power for the spacecraft decreases as temperature increases. For this reason the nonrotating structure was designed to operate at the lowest possible temperature.

The nickel-cadmium storage batteries located in the wheel section operated at good efficiency over the range 35°C to -10°C . Other components and equipment located in this section were temperature limited, but all were capable of operating satisfactorily in the range from -10°C to 35°C . Therefore, this temperature range was specified for all wheel experiments and equipment.

Many of the thermal design specifications established for the OSO were based on fundamental heat transfer considerations and did not require mathematical analysis. For example, by painting all interior wheel surfaces with "thermally black" paint and providing good conduction between adjoining structural members, temperature gradients in the wheel were minimized. This procedure tended to prevent local "hot spots" from developing in components such as the storage batteries and electrical equipment where heat was generated.

Mathematical analysis was performed in support of the overall thermal design problem to provide the following information:

1. Expected operating temperature of the solar-cell array for various orbital conditions.
2. Required values of absorptivity and emissivity for the outer surfaces of the wheel to

maintain the temperature of equipment located in this section within the specified temperature range for all orbital conditions.

3. Temperature profiles around the orbit for the wheel interior and exterior cover panels when the surface radiative properties previously determined, are assumed.

An exact mathematical analysis of the actual physical problem was not possible because of the complex geometry of the two spacecraft sections under consideration. Simplified models of these geometries had to be assumed to provide a feasible solution method, and other simplifying assumptions were made in the analysis that was performed.

This chapter presents the methods and results of the mathematical analysis and the accompanying computer program conducted in support of the thermal design program for OSO I. Derivations of the heat transfer equations for determining spacecraft temperatures are given in detail, including evaluation of the shape factor integrals associated with the heat exchange rates between the various spacecraft outer surfaces and the earth.

The simplified analysis of the wheel was run with an Intercom 500 computer program written for a Bendix G-15 computer. The analyses of the solar-cell array and the shielded model of the wheel were run from Fortran programs written for the CDC1604 computer.

ANALYSIS OF SIMPLIFIED MATHEMATICAL MODEL

In the first solution of the problem, the rotating and nonrotating sections of the spacecraft were assumed to be isolated, isothermal masses, but with the same outside geometry and thermal capacity as that of the actual space-

craft. The presence of the arms and spin-up gas bottles on the wheel structure and the pointed instruments in the upper structure was neglected. The heat transfer rate between the two sections was considered to be negligibly small.

This simplified model of the spacecraft provided a fairly accurate representation of the physical problem since good conduction paths between adjoining structural members are provided by filling all riveted or bolted joints with high-conduction epoxy resin, and all interior surfaces in the wheel are painted "thermally black." These steps provide maximum heat distribution within the two structures and prevent the occurrence of significantly large temperature gradients.

Derivation of Heat Transfer Equations

The rate of change in temperature of a thermally isolated section of the spacecraft is:

$$\frac{dT_{\oplus \rightarrow}}{dt} = \frac{q_i - q_o}{mc} \quad (7.1)$$

where

$T_{\oplus \rightarrow}$ = temperature of the spacecraft
 t = time
 q_i = radiative power absorbed by the spacecraft
 q_o = power radiated from the spacecraft
 mc = the total heat capacity of the spacecraft

The heat absorbed comes from three sources: the sun, the earth, and internal heat sources. Direct solar radiation is received at the rate

$$q_{\odot} \begin{cases} = A_s \alpha_{\odot} E_{\odot} & (\text{in sunlight}) \\ = 0 & (\text{in the earth's shadow}) \end{cases} \quad (7.2)$$

where

A_s = the projected area of the body in the direction from the sun
 $\alpha_{\odot}$ = the coefficient of absorptivity of the body for solar radiation
 $E_{\odot}$ = emissive power of the sun, 442 BTU/hr-ft²

The radiative power received by the body from the earth consists of reflected sunlight and thermal radiation. These heat input rates are given by

$$q_{\oplus a} \begin{cases} = F_a A_{\oplus \rightarrow a} \alpha_{\oplus} r_{\oplus} E_{\odot} & (\text{above sunlit earth}) \\ = 0 & (\text{above dark earth}) \end{cases} \quad (7.3)$$

and

$$q_{\oplus b} = F_b A_{\oplus \rightarrow b} \alpha_{\oplus} T_{\oplus}^4 \quad (7.4)$$

where

F_a = geometric shape factor for reflected sunlight
 F_b = geometric shape factor for thermal radiation from the earth
 $q_{\oplus a}$ = heat input rate due to reflected sunlight
 $q_{\oplus b}$ = heat input rate due to thermal radiation from the earth
 $A_{\oplus \rightarrow a}$ = surface area of the body receiving reflected sunlight
 $A_{\oplus \rightarrow b}$ = surface area of the body receiving thermal radiation from the earth
 $r_{\oplus}$ = coefficient for diffuse reflection from the earth (albedo)
 $\alpha_{\oplus}$ = coefficient of absorptivity of the body for receipt of thermal radiation at 250°K
 $T_{\oplus}$ = the effective black-body temperature of earth = 250°K
 σ = the Stefan-Boltzmann constant, 0.1714×10^{-8} BTU/hr-ft²/°R⁴

The thermal radiation rate from the spacecraft may be written as

$$q_o = A_T \epsilon_{\oplus} \sigma T^4 \quad (7.5)$$

where

A_T = total radiating surface area of body
 $\epsilon_{\oplus}$ = coefficient of emission for thermal radiation from the surface areas

The internal heat generated in the wheel section of the spacecraft is assumed to be:

$$q_{\oplus \rightarrow} = 8.5 \text{ watts (in sun); } 8.5 \text{ watts (in dark)} \quad (7.6)$$

Substituting Equations (7.4) and (7.5) into Equation (7.1) yields

$$\frac{dT_{\oplus \rightarrow}}{dt} = \frac{1}{mc} (q_{\odot} + q_{\oplus a} + q_{\oplus \rightarrow} + F_b A_{\oplus \rightarrow b} \alpha_{\oplus \sigma} T_{\oplus}^4 - A_{T \epsilon_{\oplus \rightarrow}} \sigma T_{\oplus}^4) \quad (7.7)$$

Evaluation of the Shape Factors, F_a and F_b

The radiative power received by the spacecraft through reflection of sunlight from an element dS of the earth's surface is

$$dq_{\oplus a} = I \alpha_{\odot} \cos \beta A_{\oplus \rightarrow a} \frac{\cos \phi_1 \cos \phi_2 dS}{\rho^2} \quad (7.8)$$

where

I = intensity of radiation from dS

β = angle between the solar vector and the normal to dS

ϕ_1 = angle between the direction of the spacecraft and the normal to dS

ϕ_2 = angle between the direction of dS and the normal to $A_{\oplus \rightarrow a}$

ρ = distance between dS and the spacecraft

The intensity of the radiation received by the spacecraft from dS when $\beta = 0$ is

$$I = r_{\oplus} E_{\odot} / \pi \quad (7.9)$$

Substituting Equation (7.9) into Equation (7.8), we have

$$dq_{\oplus a} = \alpha_{\odot} r_{\oplus} E_{\odot} \frac{A_{\oplus \rightarrow a}}{\pi} \cos \beta \frac{\cos \phi_1 \cos \phi_2 dS}{\rho^2}$$

or

$$q_{\oplus a} = F_a A_{\oplus \rightarrow a} \alpha_{\odot} r_{\oplus} E_{\odot} \quad (7.3)$$

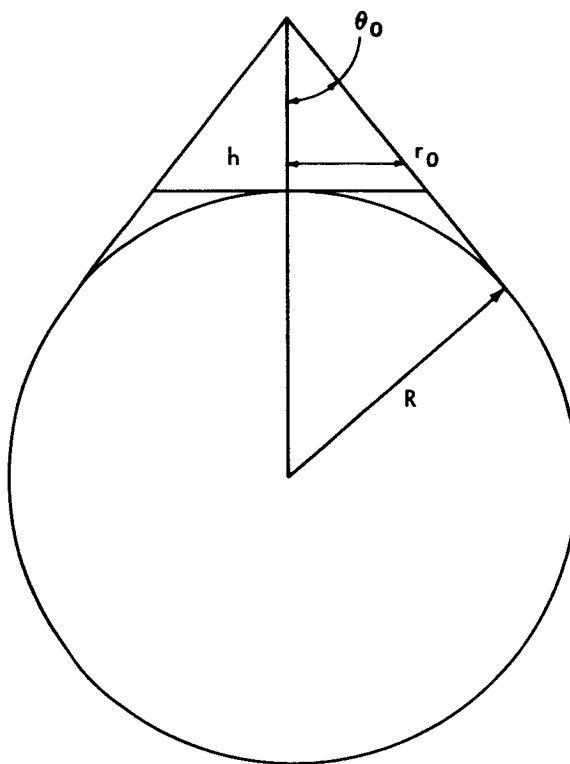
where

$$A_{\oplus \rightarrow a} F_a = \frac{A_{\oplus \rightarrow a}}{\pi} \int_s \frac{\cos \beta \cos \phi_1 \cos \phi_2 dS}{\rho^2} \quad (7.10)$$

To simplify integration of Equation (7.10), the following assumptions are made:

The earth is a flat disk of radius, r_0 , with the spacecraft a distance, h , above the center of the disk, where

h = orbital altitude of the spacecraft
 r_0 = radius of disk "seen" by the spacecraft as shown in figure below.



Then, if $h = 345$ statute miles and $R = 3960$ statute miles

$$\theta_0 = \sin^{-1} \left(\frac{R_0}{h + R_0} \right) = \sin^{-1} (0.9199) = 66^\circ 55'$$

$$r_0 = h \tan \theta_0 = (345)(2.3464) = 809.5 \text{ statute miles}$$

The angle β in Equation (7.10) is constant over the entire surface of the disk at any given time and is equal to its value for the element dS directly below the spacecraft at that time.

With the above assumptions, we have the following relationships:

$$A_{\oplus \rightarrow a} = A_{\oplus \rightarrow b}$$

and

$$F_a = \begin{cases} F_b \cos \beta & \text{for } \cos \beta \geq 0 \\ 0 & \text{for } \cos \beta < 0 \end{cases}$$

Thus, in determining the factor, $F_a A_{\oplus \rightarrow}$, in Equation (7.3) for any particular surface of the

spacecraft, we obtain an expression from which the factor, $F_{\oplus \rightarrow s}$, is readily obtained by simply letting $\cos \beta = 1$. This is equivalent to re-writing Equation (7.7) as

$$\frac{dT_{\oplus \rightarrow}}{dt} = \frac{1}{mc} [q_{\odot} + FA_{\oplus \rightarrow} (\cos \beta \alpha_{\odot} r_{\oplus} E_{\odot} + \alpha_{\oplus} \sigma T^4) + q_{\oplus \rightarrow} - A_{\tau \epsilon_{\oplus \rightarrow}} T_{\oplus \rightarrow}^4] \quad (7.11)$$

for $\cos \beta \geq 0$.

NOTE: The restriction on the value of $\cos \beta$ is used to approximate the total amount of heat received by the spacecraft from reflected sunlight during each orbit. Inclusion of the complete albedo term on the sunlit side of the terminator implies too much reflected light. Dropping the term completely on the other side implies too little reflected light. The two errors nearly cancel.

where

F = shape factor for thermal radiation from the earth

A = surface area of the spacecraft receiving reflected sunlight and/or thermal radiation from the earth.

The shape factor, F , is given by Equation (7.10) as

$$F = \frac{1}{\pi} \int_S \frac{\cos \phi_1 \cos \phi_2 dS}{\rho^2} \quad (7.12)$$

or for any surface area of the spacecraft, $A_{\oplus \rightarrow i}$, receiving radiation from the earth, we have

$$A_{\oplus \rightarrow i} F_i = \frac{A_{\oplus \rightarrow i}}{\pi} \int_S \frac{\cos \phi_1 \cos \phi_2 dS}{\rho^2} \quad (7.13)$$

Because of the abrupt discontinuity in normal direction to the various surfaces of the spacecraft, Equation (7.13) will be evaluated for each of the following surfaces:

- bottom of wheel
- top of wheel
- side of wheel
- front of solar cell array
- back of solar cell support structure

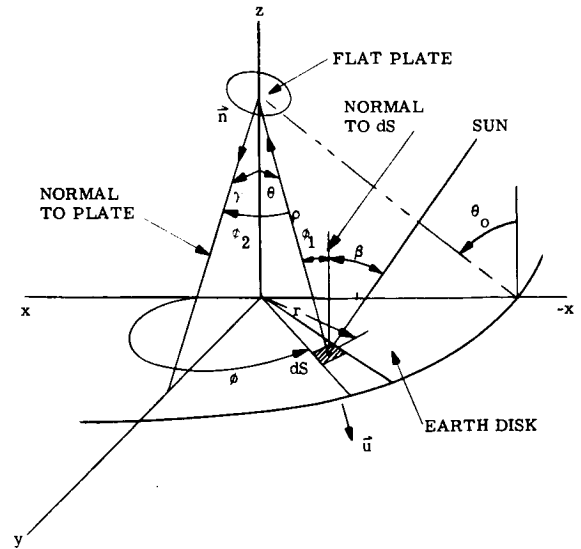


FIGURE 7-1.—Earth disk and plate geometry.

In these calculations, the wheel will be assumed to be a circular cylinder with flat ends, unobstructed by the nonrotating structure in receiving radiation from the earth. The upper structure will be considered as a semicircular flat plate, unobstructed by the wheel in receiving radiation from the earth.

Using Figure 7-1, we can construct a unit normal vector n for one surface of a flat plate of negligible thickness:

$$\vec{n} = \vec{j} \sin \gamma - \vec{k} \cos \gamma$$

Similarly, we can construct a unit vector directed from the plate toward dS :

$$\mu = \vec{i} \sin \theta \cos \phi + \vec{j} \sin \theta \sin \phi - \vec{k} \cos \theta$$

Since the dimensions of the plate will be considered very small compared to ρ , one such vector will serve for the whole plate, and it will not be necessary to integrate over the surface of the plate.

The angle γ is determined by the orientation of the plate in space. Later in the discussion it will be shown how γ is related to the orbital parameters.

Four cases arise with respect to the plate orientation (see Figure 7-2):

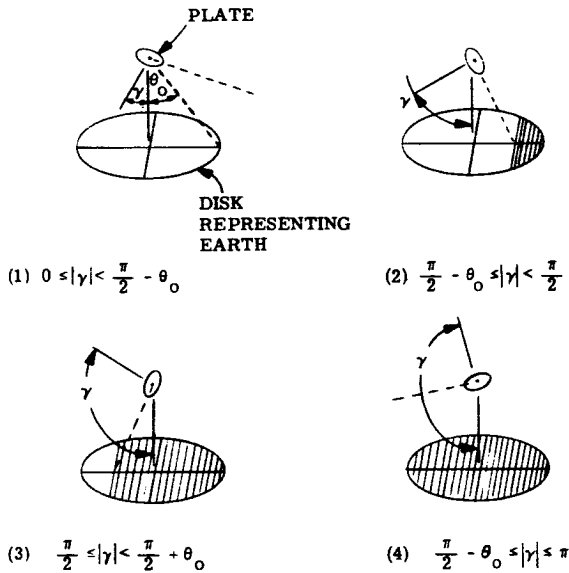


FIGURE 7-2.—Plate orientation.

1. The plate surface "sees" the entire disk representing the earth ($0 \leq |\gamma| < \pi/2 - \theta_0$).
2. The plate surface "sees" more than half, but not all, of the disk ($\pi/2 - \theta_0 \leq |\gamma| < \pi/2$).
3. The plate surface "sees" a small part (less than half) of the disk ($\pi/2 \leq |\gamma| < \pi/2 + \theta_0$).
4. The plate surface "sees" none of the disk ($\pi/2 + \theta_0 \leq |\gamma| \leq \pi$).

The boundary between the regions of the disk is given by

$$\mathbf{u} \cdot \mathbf{n} = \sin \theta \sin \phi \sin \gamma + \cos \theta \cos \gamma = 0$$

or

$$\phi = -\sin^{-1} (\cot \theta \cot \gamma)$$

The direction cosine terms in Equation (7.12) are

$$\cos \phi_1 = \cos \theta$$

$$\cos \phi_2 = \mathbf{n} \cdot \mathbf{u} = \sin \theta \sin \phi \sin \gamma + \cos \theta \cos \gamma$$

θ and ϕ are used as the variables of integration over the disk. Since

$$r = h \tan \theta$$

and

$$dr = h \sec^2 \theta d\theta$$

dS becomes

$$dS = h^2 \sec^2 \theta \tan \theta d\theta d\phi$$

Likewise

$$\rho^2 = h^2 \sec^2 \theta$$

so that the shape factor becomes

$$F = \frac{1}{\pi} \iint (\sin^2 \theta \sin \phi \sin \gamma + \sin \theta \cos \theta \cos \gamma) d\phi d\theta \quad (7.14)$$

evaluated for the appropriate area of the disk. The selection of proper limits and the evaluation is given beginning on page 211. The results are

Case 1 ($0 \leq |\gamma| < \pi/2 - \theta_0$):

$$F = \cos \gamma \sin^2 \theta_0 \quad (7.15)$$

Case 2 ($\pi/2 - \theta_0 \leq |\gamma| < \pi/2$):

$$\begin{aligned} F = & \frac{1}{2} \sin^2 \theta_0 \cos \gamma + \frac{1}{2} \sin^2 \gamma \\ & - \frac{1}{\pi} \cos \theta_0 \sqrt{\sin^2 \theta_0 - \cos^2 \gamma} \\ & - \frac{1}{\pi} \sin^2 \gamma \tan^{-1} \left(\frac{\cos \theta_0}{\sqrt{\sin^2 \theta_0 - \cos^2 \gamma}} \right) \\ & + \frac{1}{\pi} \cos^2 \gamma \tan^{-1} (|\cos \gamma| \sqrt{\tan^2 \gamma \tan^2 \theta_0 - 1}) \\ & + \frac{1}{\pi} \cos \gamma \sin^2 \theta_0 \sin^{-1} (\cot \gamma \cot \theta_0) \end{aligned} \quad (7.16)$$

Case 3 ($\pi/2 \leq |\gamma| < \pi/2 + \theta_0$):

Same expression as Case 2.

Case 4 ($\pi/2 + \theta_0 \leq |\gamma| \leq \pi$):

$$F = 0 \quad (7.17)$$

These functions can be considered for $0 \leq \gamma \leq \pi$ only, since γ will be determined by a cosine expression and the choice of quadrant for γ is thus partly arbitrary. A plot of F versus γ for a flat surface is shown in Figure 7-3.

Shown in Figure 7-3 is a graph of the function:

$$0.846 \cos^2 (0.6345\gamma) \cos (0.6345\gamma) \geq 0$$

$$0.0 \cos (0.6345\gamma) < 0$$

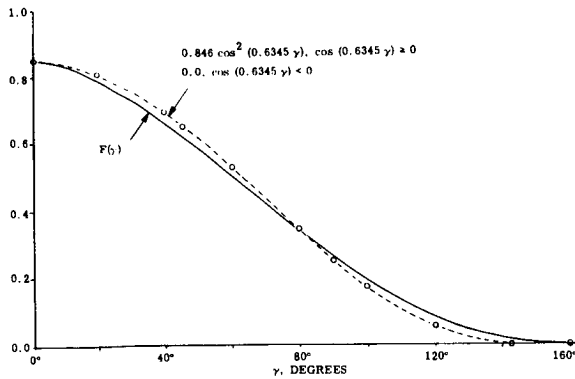


FIGURE 7-3.—Shape functions for flat plate.

which was used in the computer solution to approximate F for a single surface of a flat plate. This approximation is much simpler than the expression for F and is sufficiently accurate. The constants in the approximation were chosen

so that the areas under the two curves are equal.* In Figure 7-4 are graphs of the shape functions of both surfaces of a flat plate versus γ for the bottom surface, together with a shape factor for the plate when both surfaces have the same characteristics and both receive radiation. Also shown in this figure is a plot of the equation

$$F_{t+b} = 0.332 \cos^2 \gamma + 0.514 \quad 0 \leq \gamma \leq \pi \quad (7.18)$$

which was used as an approximation to the composite shape function.

For the sides, a shape function

$$F_s = 0.254 + 0.077 \sin^2 \gamma \quad (7.19)$$

was used, where γ is the angle between local vertical and the axis of the cylindrical surface. The development of this expression is given on page 211.

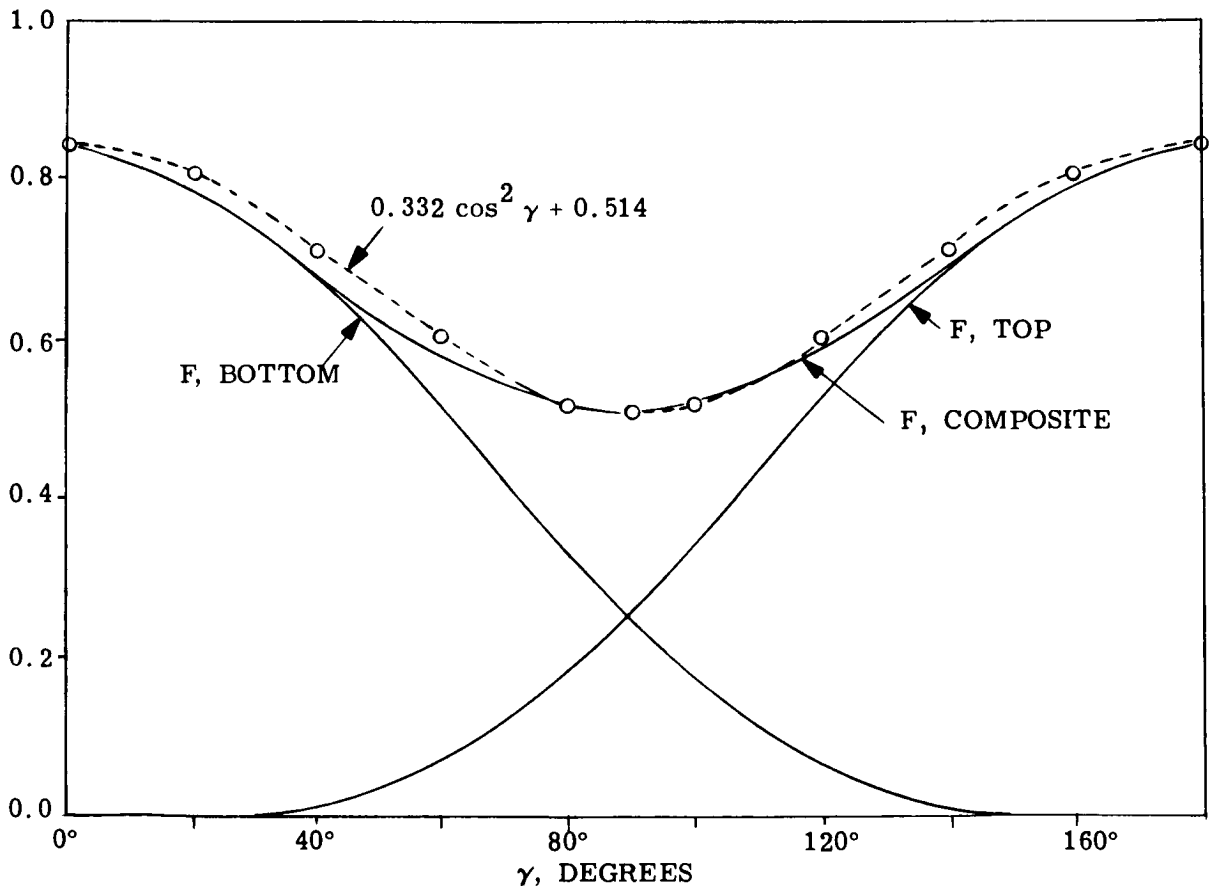


FIGURE 7-4.—Shape functions for both surfaces of flat plate.

Statement of the Problem for Computer Solution

Orbital Geometry

The coordinate system shown in Figure 7-5 was used to relate the angles γ and β in Equations (7.11), (7.18) and (7.19) to the orbital parameters η , ϵ , and ζ . The sun is always taken to be along the x -axis. The angle ϵ then represents the angle between the satellite orbital plane and the ecliptic, which varies from 0° to 56.78° for a 33.33° orbit with respect to the earth's equatorial plane.

A local vertical unit vector is given by

$$\mathbf{v} = \mathbf{i} \sin \eta \cos \epsilon + \mathbf{j} \sin \eta \sin \epsilon + \mathbf{k} \cos \eta$$

Thus, $\cos \beta$, at any point on the illuminated surface of the earth is given by

$$\cos \beta = \mathbf{i} \cdot \mathbf{v} = \sin \eta \cos \epsilon \quad (7.20)$$

whenever $\cos \beta \geq 0$.

Unit vectors for the top and bottom surfaces of the spacecraft are

$$\mathbf{n}_t = \mathbf{j} \sin \zeta + \mathbf{k} \cos \zeta$$

and

$$\mathbf{n}_b = -\mathbf{n}_t$$

Thus for the top

$$\cos \gamma_t = \mathbf{n}_t \cdot \mathbf{v} = \sin \eta \sin \epsilon \sin \zeta + \cos \eta \cos \zeta \quad (7.21)$$

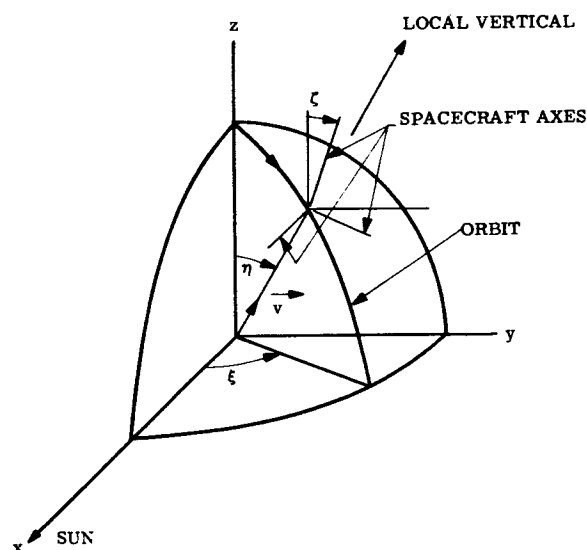


FIGURE 7-5.—Spacecraft orbital coordinate system.

and for the bottom

$$\cos \gamma_b = \mathbf{n}_b \cdot \mathbf{v} = -\sin \eta \sin \epsilon \sin \zeta - \cos \eta \cos \zeta \quad (7.22)$$

Because of the symmetry of the wheel section and the assumption of like surface characteristics of the top and bottom plates, the variation of ζ from 0° to 90° is sufficient to determine the variation of heat input rates due to changes in this parameter. Therefore, a limited number of combinations of values for ζ and ϵ was considered adequate to provide a good approximation of the extreme temperature variations that could be expected for a particular spacecraft configuration. These were

$$\zeta = 0^\circ, 45^\circ, 90^\circ$$

$$\epsilon = 0^\circ, 10^\circ, 20^\circ, 30^\circ, 40^\circ, 50^\circ, 56.78^\circ$$

Change of Independent Variable

In the finite difference form of the heat transfer equation required for a digital computer solution, it is convenient to take equal intervals of change in the independent variable. In this particular problem, the heat input rates to the spacecraft are periodic with respect to orbital time, so it is desirable to divide each orbit into a finite number of equal intervals. For a circular orbit, the position angle, η , of the spacecraft with respect to a given initial position, varies linearly with time. Therefore, the choice of using either equal time intervals or position intervals exists. Orbital time for the 300-nautical-mile circular orbit is not integral, but the position angle varies from 0° to 360° during each orbit and is easily divided into equal intervals of integral value. For this reason, the following change of variable in Equation (7.11) was made:

$$\frac{dT}{dt} = \frac{dT}{d\eta} \cdot \frac{d\eta}{dt} = \frac{dT}{d\eta} \cdot \dot{\eta} = \frac{1}{mc} f(\gamma, \beta, T)$$

or

$$\frac{dT}{d\eta} = \frac{1}{mc\dot{\eta}} f(\gamma, \beta, T) \quad (7.23)$$

where $\dot{\eta}$ = constant angular rate of the satellite about the center of the earth.

Rewriting Equation (7.11) in the form of Equation (7.23) and substituting from Equations (7.18) and (7.19), we have

$$\begin{aligned} \frac{dT_{\oplus \rightarrow}}{d\eta} = & \frac{1}{mc\eta} \{ q_{\odot} + \cos \beta r_{\oplus} E_{\odot} [A_{\oplus \rightarrow i} \alpha_{\odot i} (0.514 + 0.332 \cos^2 \gamma) + A_{\oplus \rightarrow s} \alpha_{\oplus s} (0.254 + 0.077 \sin^2 \gamma)] \\ & + \sigma T_{\oplus}^4 [A_{\oplus \rightarrow i} \alpha_{\oplus i} (0.514 + 0.332 \cos^2 \gamma) + A_{\oplus \rightarrow s} \alpha_{\oplus s} (0.254 + 0.077 \sin^2 \gamma)] \\ & + q_{\oplus \rightarrow} - (2A_{\oplus \rightarrow i} \epsilon_{\oplus \rightarrow i} + A_{\oplus \rightarrow s} \epsilon_{\oplus \rightarrow s}) \sigma T_{\oplus}^4 \} \quad (7.24) \end{aligned}$$

for the rate of change in temperature of the wheel section, and

$$\begin{aligned} \frac{dT_{\oplus \rightarrow}}{d\eta} = & \frac{1}{mc\eta} [(A_{BK} F_{BK} \alpha_{\oplus BK} + A_{FR} F_{FR} \alpha_{\oplus FR}) \sigma T_{\oplus}^4 + (A_{BK} F_{BK} \alpha_{\odot BK} \\ & + A_{FR} F_{FR} \alpha_{\odot FR}) r_{\oplus} E_{\odot} \cos \beta + q_{\oplus \rightarrow} - (A_{BK} \epsilon_{\oplus \rightarrow BK} + A_{FR} \epsilon_{\oplus \rightarrow FR}) \sigma T_{\oplus}^4] \quad (7.25) \end{aligned}$$

for the upper structure. In Equations (7.24) and (7.25),

$$F_{FR} = \begin{cases} 0.846 \cos^2 [0.6345 \cos^{-1} (-\sin \eta \cos \epsilon)], & \text{if } \cos [] \geq 0 \\ 0 & \text{if } \cos [] < 0 \end{cases}$$

$$F_{BK} = \begin{cases} 0.846 \cos^2 [0.6345 \cos^{-1} (\sin \eta \cos \epsilon)], & \text{if } \cos [] \geq 0 \\ 0 & \text{if } \cos [] < 0 \end{cases}$$

$$\cos \beta = \begin{cases} \sin \eta \cos \zeta & \text{if } \sin \eta \cos \zeta \geq 0 \\ 0 & \text{if } \sin \eta \cos \zeta < 0 \end{cases}$$

and

$$q_{\odot} = \begin{cases} A_{s\odot} E_{\odot} & \text{for } \eta \leq \eta_{in} \text{ or } \eta \geq \eta_{out} \\ 0 & \text{for } \eta_{out} > \eta > \eta_{in} \end{cases}$$

The expressions for determining the position angles of the satellite when it goes into and emerges from the earth's shadow are, respectively,

$$\begin{aligned} \eta_{in} &= 57.296 \left[\pi + \tan^{-1} \sqrt{\frac{1 - (R/R+h)^2}{(R/R+h)^2 - \sin^2 \epsilon}} \right] \\ \eta_{out} &= 57.296 \left[2\pi - \tan^{-1} \sqrt{\frac{1 - (R/R+h)^2}{(R/R+h)^2 - \sin^2 \epsilon}} \right] \end{aligned} \quad (7.26)$$

for

$$[(R/R+h)^2 - \sin^2 \epsilon] > 0,$$

and

$$\eta_{in} = \eta_{out} = 270^\circ \quad (7.27)$$

for

$$[(R/R+h)^2 - \sin^2 \epsilon] \leq 0$$

In Equations (7.26), η_{in} and η_{out} are given in degrees from an initial position $\eta = 0^\circ$, and

R = radius of the earth

h = orbital altitude of the spacecraft

Finite Difference Form

Euler's Modified Method was used for the numerical integration of the differential equation. This method is ordinarily not satisfactory for extending a solution very far from a starting point because of the error build-up, but in this case we wished only to carry the solution around a fixed orbit repeatedly until the temperature cycle became stable. The speed of this method, which was a prime consideration, can hardly be exceeded.

The method consists of finding a "zeroth" approximation for T_{i+1} , namely, T_{i+1}^0 , by the use of the relation

$$T_{i+1}^{(0)} = T_i + \frac{\Delta \eta}{mc\eta} f(\gamma_i, \beta_i, T_i) \quad (7.28)$$

This is simply a linear extrapolation of the solution from the point (η_i, T_i) , using the slope of the solution curve at that point. Successively better approximations to (η_{i+1}, T_{i+1}) are then found using the iteration formula

$$T_{i+1}^{(n+1)} = T_i + \frac{\Delta \eta}{2mc\eta} f(\gamma_i, \beta_i, T_i) + f(\gamma_{i+1}, \beta_{i+1}, T_{i+1}^{(n)}) \quad (7.29)$$

This is actually an extrapolation process from (η_i, T_i) using the average slope between the points

$$(\eta_i, T_i) \text{ and } (\eta_{i+1}, T_{i+1}^{(n)}).$$

The criterion used for termination of the iteration was that

$$|T_{i+1}^{(n+1)} - T_{i+1}^{(n)}| \leq \epsilon \quad (7.30)$$

where ϵ was a previously chosen test constant. In general, it was found that iteration would not proceed through more than one or two cycles unless ϵ was on the order of 10^{-3} , or less.

The solution for the wheel was programmed in Intercom 500 language for the Bendix G-15 computer, and for the upper structure in Fortran language for the CDC 1604 computer.

Values of Parameters

The values listed for the following parameters were not changed during the computer program:

$$mc = \begin{cases} 60.8 \text{ BTU/}^\circ\text{R} & (\text{for entire wheel}) \\ 4.84 \text{ BTU/}^\circ\text{R} & (\text{for homogenous shell}) \\ 2.655 \text{ BTU/}^\circ\text{R} & (\text{for solar-cell array}) \end{cases}$$

$$A_s = 3.66 \text{ ft}^2$$

$$E_\odot = 7.37 \text{ BTU/min-ft}^2$$

$$A_{\oplus i} = A_{\oplus b} = 9.56 \text{ ft}^2$$

$$A_{\oplus s} = 11.08 \text{ ft}^2$$

$$A_{BK} = A_{FR} = 4.52 \text{ ft}^2$$

$$\alpha_{\oplus BK} = \epsilon_{\oplus \rightarrow BK} = 0.85$$

$$\alpha_{\oplus FR} = \epsilon_{\oplus \rightarrow FR} = 0.70$$

$$\alpha_{\odot BK} = 0.20$$

$$\alpha_{\odot FR} = 0.80$$

$$T_\oplus = 450^\circ\text{R}$$

$$q_{\oplus \rightarrow} = 0.48 \text{ BTU/min (both in sunlight and dark)}$$

$$\sigma = 2.8567 \times 10^{-11} \text{ BTU/min-ft}^2\text{-}^\circ\text{R}^4$$

$$\eta = 3.7895^\circ/\text{min}$$

$$R = 3960 \text{ statute miles}$$

$$h = 340 \text{ statute miles}$$

$$\Delta\eta = \begin{cases} 15^\circ & (\text{for G-15 program}) \\ 5^\circ & (\text{for CDC 1604 program}) \end{cases}$$

The following parameters were changed in value at various times during the program to

determine their effect on spacecraft temperatures:

$$\alpha_{\odot i}, \alpha_{\odot b}, \alpha_{\odot s}$$

$$\alpha_{\oplus i}, \alpha_{\oplus b}, \alpha_{\oplus s}$$

$$\epsilon_{\oplus \rightarrow i}, \epsilon_{\oplus \rightarrow b}, \epsilon_{\oplus \rightarrow s}$$

$$T_\oplus$$

The values assumed for each of these parameters during each series of computer runs are listed in the discussion of the results of these runs.

Computer Printout Format

Bendix G-15 Program for Wheel—Two different printout formats were used. Examples of each of these are shown in Figures 7-6 and 7-7. Each of the formats had a heading for each orbital run consisting of the fixed values of ϵ , ζ , η_{in} , and η_{out} . The first program then printed the following variables for each η_i around the orbit for as many times as the orbit was run until the temperature stabilized: η_i , $T_{\oplus \rightarrow}$, heat input rate from albedo, heat input rate from earth black body radiation, total heat input rate, and heat output rate. When the program was rewritten to shorten

$T_\odot = 455.890^\circ\text{R}$

HEADING						
	ξ (deg)	ζ (deg)	η_{in} (deg)	η_{out} (deg)		
	56.780	.000	225.720	314.280		
ORBITAL RUN						
η (deg)	$T_{\oplus \rightarrow}$ ($^\circ\text{R}$)	q_{alb} (BTU/min)	q_{bb} (BTU/min)	q_{in} (BTU/min)	q_{out} (BTU/min)	
15.000	455.990	1.495	4.316	15.006	-12.788	
30.000	456.170	2.772	4.142	16.109	-12.810	
45.000	456.410	3.696	3.905	16.796	-12.838	
60.000	456.670	4.252	3.667	17.113	-12.869	
75.000	456.950	4.517	3.493	17.205	-12.902	
90.000	457.230	4.591	3.430	17.215	-12.933	
105.000	457.510	4.517	3.493	17.205	-12.965	
120.000	457.780	4.252	3.667	17.113	-12.996	
135.000	458.040	3.696	3.905	16.796	-13.026	
150.000	458.260	2.772	4.142	16.109	-13.053	
165.000	458.420	1.495	4.316	15.006	-13.073	
180.000	458.500	.000	4.380	13.574	-13.083	
195.000	458.530	.000	4.316	13.510	-13.081	
210.000	458.550	.000	4.142	13.336	-13.084	
225.000	458.560	.000	3.905	13.099	-13.085	
240.000	458.270	.000	3.667	4.147	-13.051	
255.000	457.690	.000	3.493	3.973	-12.986	
270.000	457.100	.000	3.430	3.910	-12.919	
285.000	456.520	.000	3.493	3.973	-12.853	
300.000	455.950	.000	3.667	4.147	-12.788	
315.000	455.680	.000	3.905	13.099	-12.759	
330.000	455.710	.000	4.142	13.336	-12.761	
345.000	455.750	.000	4.316	13.510	-12.766	
.000	455.800	.000	4.380	13.574	-12.772	

FIGURE 7-6.—Example of G-15 printout format—first computer program.

ξ (deg) 0.000	ζ (deg) 0.000	η_{in} (deg) 203.100	η_{out} (deg) 336.910
HEADING	q_{alb} (BTU/min)	q_{bb} (BTU/min)	q_{in} (BTU/min)
η (deg)	0.000	2.020	11.215
0.000	0.000	2.019	14.160
15.000	2.947	2.014	16.703
30.000	5.494	2.008	18.588
45.000	7.385	2.002	19.770
60.000	8.573	1.998	20.369
75.000	9.177	1.996	20.546
90.000	9.355	1.998	20.369
105.000	9.177	2.002	19.770
120.000	8.573	2.008	18.588
135.000	7.385	2.014	16.703
150.000	5.494	2.019	14.160
165.000	2.947	2.020	11.215
180.000	.000	2.019	11.213
195.000	.000	2.014	2.494
210.000	.000	2.008	2.488
225.000	.000	2.002	2.482
240.000	.000	1.998	2.478
255.000	.000	1.996	2.476
270.000	.000	1.998	2.478
285.000	.000	2.002	2.482
300.000	.000	2.008	2.488
315.000	.000	2.014	2.494
330.000	.000	2.019	11.213
345.000	.000	2.020	11.215
360.000	.000		
ORBITAL RUN	q_{out} (BTU/min)	$T_{\oplus}$ (°R)	
η (deg)	-10.987	512.000	
0.000	-10.997	512.110	
15.000	-11.021	512.400	
30.000	-11.058	512.830	
45.000	-11.104	513.360	
60.000	etc.	etc.	

FIGURE 7-7.—Example of G-15 printout format-second computer program.

the computation time of each temperature calculation, the format was also changed to shorten typeout time. The various heat input rates were printed only for the first time around the orbit. During succeeding orbits, only $T_{\oplus}$ and heat output rate for each η_i were printed.

CDC 1604 Program for Upper Structure—Since only one series of runs was made during this program, the printout format could be made much simpler than for the G-15. For each of the seven runs the format heading gave the fixed values of ϵ , η_{in} , and η_{out} . The variable data consisted of:

1. Number of orbits spacecraft completed before temperature of upper structure stabilized
2. Average temperature of upper structure for the last orbit
3. Temperature of upper structure at each of the 72 points around the last orbit.

Summary of Results of the Computer Programs*

Satellite Wheel Section

All Surfaces Painted With Aluminum Paint—

In the first series of computer runs, the measured values $\alpha_{\oplus}$, $\alpha_{\odot}$, and $\epsilon_{\oplus}$ for the aluminum paint used on the OSO I prototype model were substituted into the program. In addition, a typical average value for the earth's albedo was assumed. These values were:

$$\left. \begin{aligned} \alpha_{\odot_i} = \alpha_{\odot_b} = \alpha_{\odot_s} &= 0.32 \\ \alpha_{\oplus_i} = \alpha_{\oplus_b} = \alpha_{\oplus_s} &= 0.343 \\ \epsilon_{\oplus_i} = \epsilon_{\oplus_b} = \epsilon_{\oplus_s} &= 0.343 \\ r_{\oplus} &= 0.4 \end{aligned} \right\} \quad (7.31)$$

Table 7-1 summarizes the results of the various runs made for this configuration. Changes in attitude and orbital inclination made little difference in the mean temperature of the spacecraft. The temperature in all cases was unsatisfactorily low, however, averaging about -20°C . These runs showed conclusively that the ratio of $\alpha_{\odot}/\alpha_{\oplus}$ for at least part of the surfaces would have to be increased to obtain the specified minimum temperature of -10°C .

Top and Bottom Surfaces Partially Unpainted—The next series of solutions determined the effect of leaving part of the top and bottom surfaces unpainted. The $\alpha_{\odot}/\epsilon_{\oplus}$ ration for unpainted, shiny aluminum is about five times greater than for the aluminum paint used on the prototype. This modification in surface characteristics therefore appeared to be a solution to the problem of elevating the temperature above -10°C .

For this configuration, $\alpha_{\odot_s}$, $\alpha_{\oplus_s}$, and $\epsilon_{\oplus_s}$ were kept the same as listed above for the first series of computer solutions. The absorptivity and emissivity for the top and bottom surfaces were calculated from the following relations:

$$\begin{aligned} \alpha_{\odot_i} &= \lambda \alpha_{\odot_{i1}} + (1 - \lambda) \alpha_{\odot_{i2}} \\ \alpha_{\oplus_i} &= \lambda \alpha_{\oplus_{i1}} + (1 - \lambda) \alpha_{\oplus_{i2}} \\ \epsilon_{\oplus_i} &= \lambda \epsilon_{\oplus_{i1}} + (1 - \lambda) \epsilon_{\oplus_{i2}} \end{aligned}$$

where

λ = fractional part of $A_{\oplus_i}$ or $A_{\oplus_b}$ of surface condition 1

$1 - \lambda$ = fractional part of $A_{\oplus_i}$ or $A_{\oplus_b}$ of surface condition 2

Table 7-1

Summary of Results of Commuter Runs
All Surfaces Painted with Aluminum Paint

Constant Values: all $\alpha_{\odot}=0.32$; all $\alpha_{\oplus}=\epsilon_{\oplus\rightarrow}-0.343$; $r_{\oplus}=0.40$

Run No.	ξ (deg)	ζ (deg)	η in (deg)	η out (deg)	Maximum q_{alt} (BTU/min)	Maximum q_{bb} (BTU/min)	Minimum T $\oplus \rightarrow$		Maximum T $\oplus \rightarrow$	
							(°R)	(°K)	(°R)	(°K)
1 -----	0	0	203. 10	336. 91	8. 38	4. 38	453. 5	252	458. 3	255
2 -----	0	45	203. 10	336. 91	8. 38	3. 91	450. 3	250	455. 0	253
3 -----	56. 78	0	225. 72	314. 28	4. 59	4. 38	455. 9	253	258. 6	255
4 -----	56. 78	45	225. 72	314. 28	5. 13	4. 23	455. 0	253	457. 7	254
5 -----	56. 78	90	225. 72	314. 28	5. 48	4. 10	454. 9	253	457. 6	254
6 -----	30	0	206. 93	333. 07	7. 26	4. 38	449. 3	250	453. 9	252
7 -----	30	45	206. 93	333. 07	7. 55	4. 02	448. 0	249	452. 5	251
8 -----	30	30	206. 93	333. 07	7. 76	3. 67	446. 2	248	450. 9	250

Let surface condition 1 be the shiny, unpainted state. Then the values of the parameters with subscript 1 in the above operations assumed were:

$$\alpha_{\odot 1}=0.25$$

$$\alpha_{\oplus 1}=\epsilon_{\oplus 1}=0.05$$

The subscript 2 parameters kept the same values as those assumed in the previous configuration for the totally painted spacecraft.

The procedure followed in varying λ in Equations (7.31) was to determine the percentage of unpainted surface required to maintain spacecraft temperature above 0°C even when a conservatively low value of 0.3 was assumed for the earth's albedo. As shown in the first seven runs in Table 7-2, this value was 85 percent. Several runs were then made to observe the effect of varying ϵ and ζ over their full range of values. From runs 7-21 it is obvious that their effect on spacecraft temperature is very slight.

Runs 22 and 23 of Table 7-2 were made to estimate the "worst possible" hot temperature that might be anticipated for this configuration. The assumptions made for these runs were:

$$\begin{aligned} r_{\oplus} &= 0.5 \\ \cos \beta &= 1 \text{ for } \sin \eta \cos \zeta \geq 0 \\ \alpha_{\odot i} &= \alpha_{\oplus i} = \epsilon_{\oplus i} = 0 \end{aligned}$$

The first assumption represents the upper limit for the average value of the earth's albedo quoted in the literature. The second assumption eliminates the previous assumption that reflected sunlight from the earth is dependent upon the angle of incidence of sunlight in the earth's surface. The last assumption implies that the top and bottom surfaces of the wheel are thermally insulated. The earlier assumption that the entire wheel is a homogenous, isothermal mass probably yields conservatively low mean temperatures because in the actual case, rather poor conductive heat transfer takes place between the internal parts of the wheel and the outer surfaces. Therefore, the last assumption should yield conservatively high temperatures since it asserts that no transfer of heat takes place through the top and bottom surfaces.

The maximum temperature of 40°C obtained in Run 23 of Table 7-2 is above the specified maximum of 35°C. However, it is close enough to the maximum allowable to indicate that when

Table 7-2.—Summary of Results of Computer Runs Top and Bottom Surfaces Partially Unpainted*

Run No.	$r_{\oplus}$	$\alpha_{\odot}$	$\alpha_{\oplus}$ $\epsilon_{\oplus}$	Percent Unpainted Surface	ϵ (deg)	ζ (deg)	Percent Sunlight	Maximum q_{ab} (BTU/min)	Maximum q_{bb} (BTU/min)	Min. $T_{\oplus}$ (°K)	Max. $T_{\oplus}$ (°K)
1-----	0.4	0.285	0.174	50	0	0	62.83	7.853	2.779	268	271
2-----	0.4	0.25	0.05	100	0	0	62.83	7.326	1.743	289	291
3-----	0.4	0.25	0.05	100	0	45	62.83	7.326	1.743	289	291
4-----	0.4	0.25	0.05	100	0	90	62.83	7.326	1.743	288	291
5-----	0.3	0.25	0.05	100	0	0	62.83	5.494	1.743	284	286
6-----	0.3	0.257	0.0793	90	0	0	62.83	5.573	1.912	278	280
7-----	0.3	0.2605	0.09395	85	0	0	62.83	5.613	2.020	275	278
8-----	0.3	0.2605	0.09395	85	0	45	62.83	5.613	2.008	275	278
9-----	0.3	0.2605	0.09395	85	0	90	62.83	5.613	1.996	275	277
10-----	0.3	0.2605	0.09395	85	10	0	63.04	5.527	2.020	275	278
11-----	0.3	0.2605	0.09395	85	10	45	63.04	5.547	2.009	275	278
12-----	0.3	0.2605	0.09395	85	10	90	63.04	5.566	1.997	275	277
13-----	0.3	0.2605	0.09395	85	20	0	63.71	5.274	2.020	275	277
14-----	0.3	0.2605	0.09395	85	20	45	63.71	5.346	2.008	275	277
15-----	0.3	0.2605	0.09395	85	20	90	63.71	5.418	1.998	274	277
16-----	0.3	0.2605	0.09395	85	30	0	64.96	4.861	2.020	274	277
17-----	0.3	0.2605	0.09395	85	30	45	64.96	5.002	2.011	274	276
18-----	0.3	0.2605	0.09395	85	30	90	64.96	5.144	2.002	274	276
19-----	0.3	0.2605	0.09395	85	40	0	67.11	4.300	2.020	277	279
20-----	0.3	0.2605	0.09395	85	50	0	70.89	3.608	2.020	277	278
21-----	0.3	0.2605	0.09395	85	56.78	0	75.40	3.075	2.020	280	282
22*-----	0.5	0	0	0	0	0	62.83	4.446	1.445	300	302
23*-----	0.5	0	0	0	56.78	0	75.40	4.446	1.445	311	313

*See text for further explanation.

the three previous assumptions are revised to approximate more closely the actual situation, this configuration can be modified to stay well within the specified temperature range of -10°C to 35°C regardless of attitude or orbital inclination.

Alternate Approach to Raising Mean Temperature.—The proposed solution of leaving part of the top and bottom surfaces unpainted is not very desirable from the ground handling standpoint. Extreme care must be exercised during ground operations to assure that these surfaces are not scratched or exposed to grease and other substances that might leave a film on the surface. A slight modification in surface characteristics of highly polished metals can substantially alter their radiative properties.

As a possible alternate method of raising spacecraft temperature above 0°C , the scheme of painting top and bottom surfaces with aluminum paint and the sides with thermally black paint was analyzed on the computer. It was reasoned that the increased heat input

through the sides, particularly from direct sunlight, should more than offset the additional loss of heat through these surfaces. As can be seen from Run 1 of Table 7-3, this reasoning proved to be incorrect. The computer solution shows that the converse is true, and that the minimum temperature drops below -30°C .

Effect of Various Assumptions on Mean Temperature—Runs 2 through 6 of Table 7-3 were made to determine the individual effect on temperature of various assumptions made in previous runs. Albedo values of 0.3, 0.4, and 0.5 were assumed for different runs keeping other parameters constant. Results showed that a change of 0.1 in the assumed albedo value produced only about a 5°C change in spacecraft temperature.

The assumption $\cos \beta = 1$ (Run 5) produced a substantial temperature change of 14°C . This result points out the importance of further investigation into the nature of reflected sunlight from the earth. This heat source is significant for low-orbit spacecrafts, and until

THERMAL CONTROL

Table 7-3.—Summary of Results of Computer Runs Sides Painted Black and Various Other Assumptions

Constant Values: $\sigma = \epsilon = 0^\circ$

Run No.	$r_\oplus$	$\alpha_{\odot_i}$	$\alpha_{\oplus_i}$ $\epsilon_{\oplus_i}$	$\alpha_{\odot_e}$	$\alpha_{\oplus_e}$ $\epsilon_{\oplus_e}$	Maximum q_{ais} (BTU/min)	Maximum q_{ie} (BTU/min)	Maximum q_{in} (BTU/min)	Min. $T_\oplus$ ($^\circ\text{K}$)	Max. $T_\oplus$ ($^\circ\text{K}$)	Remarks
1..	0.4	0.32	0.343	0.95	0.85	15.384	6.051	30.160	241	245	cos $\beta = 1$. Top and bottom insulated. mc = 4.84. $\epsilon = 56.78__$.
2..	0.3	0.2605	0.09395	0.32	0.343	5.613	2.020	16.803	275	278	
3..	0.4	0.2605	0.09395	0.32	0.343	7.484	2.020	18.675	280	283	
4..	0.5	0.2605	0.09395	0.32	0.343	9.355	2.020	20.546	284	287	
5..	0.5	0.2605	0.09395	0.32	0.343	11.534	2.020	22.748	298	302	
6..	0.4	0	0	0.32	0.343	3.557	1.455	14.206	290	292	
7..	0.4	0.2605	0.09395	0.32	0.343	7.484	2.020	18.675	264	296	
8..	0.4	0.2605	0.09395	0.32	0.343	4.100	2.020	15.290	272	291	

data on its nature and intensity are made available from present spacecraft flights, accurate temperature predictions for future spacecrafts will be difficult to make.

Runs 7 and 8 of Table 7-3 show the temperature fluctuations in a homogenous shell during one orbit. Due to its low thermal capacity the shell temperature varied as much as 32°C from minimum to maximum. However, the mean temperature of the shell was about the same as that of the entire wheel for similar surface characteristics. Thus, any particular isolated mass inside the wheel will probably fluctuate more significantly than the $\pm 2^\circ\text{C}$ indicated for the total mass, but its mean temperature can be expected to be about the same as that of the total mass.

Solar Cell Array

The silicon solar cells used on the OSO 1 covered practically the entire surface of the upper structure facing the sun. Because of their high solar absorptivity and low thermal inertia, the cells would heat up rapidly when exposed to direct sunlight. Since the efficiency of these cells decreased with increasing temperature, it was desirable to keep the solar-cell array as cool as possible when the spacecraft was in sunlight. This objective was achieved by painting the back surface of the solar cell panels and all support structure with a special metallic oxide paint. Therefore, the surface radiative parameters for this section of the spacecraft

were fixed, and the heat transfer computer study was concerned only with determining the expected operating temperature of the cells for various orbital conditions.

Computer runs were made for the seven discrete values of the orbital inclination angle, ϵ , listed previously in this report. Variations in spacecraft attitude angle, ζ , were unnecessary because, if shadowing by the wheel is neglected, the upper structure is exposed to the same heat input rates from the sun and earth regardless of attitude.

The following fixed values of parameters were assumed for this study.

$$\begin{aligned}
 A_{FR} &= A_{BK} = 4.52 \text{ ft}^2 \\
 \alpha_{\oplus FR} &= \epsilon_{\oplus \rightarrow FR} = 0.70 \\
 \alpha_{\oplus BK} &= \epsilon_{\oplus \rightarrow BK} = 0.85 \\
 \alpha_{\odot FR} &= 0.80 \\
 \alpha_{\odot BK} &= 0.20 \\
 mc &= 2.655 \text{ BTU}/^\circ\text{R} \\
 r_\oplus &= 0.40
 \end{aligned}$$

In the above list, subscripts FR and BK refer to the front and back sides of the structure, respectively.

Output data for each of the computer runs consisted of upper temperature and heat input rates from the earth into the front and back surfaces, at each of the 72 points around the orbit.

The temperature data for the minimum sunlight orbit ($\xi = 0^\circ$) and maximum sunlight orbit ($\xi = 56.78^\circ$) are plotted in Figure 7-8. Tem-

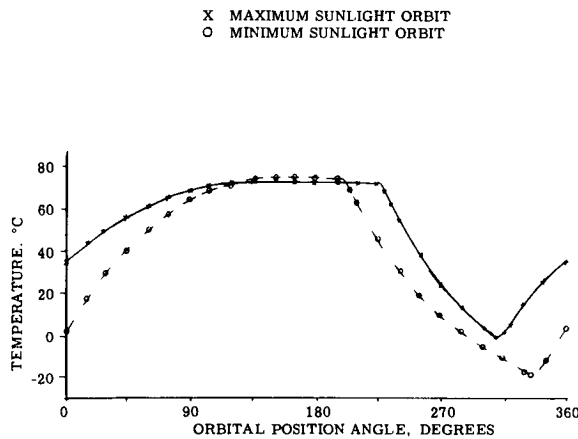


FIGURE 7-8.—Temperature variation of solar-cell array around orbit.

perature extremes for the five other orbits were bracketed by the two cases plotted. These curves indicated that solar-cell temperature remained in the range from -20°C to 75°C for all orbits, and averaged about 55°C in sunlight.

ANALYSIS OF SHIELDED MODEL OF SPACECRAFT WHEEL

A second solution of the heat transfer problem for the wheel section was obtained based on the assumption that the cover panels of this section act as radiation shields for the enclosed experiments, equipment and structure. On the actual flight model, relatively poor thermal contact existed between these panels and the rest of the wheel, resulting in little or no conductive heat transfer in the vacuum environment of orbital flight. Therefore, it was felt that this second, or shielded, model of the wheel was a more accurate representation of the physical problem than the simplified model.

The geometry assumed for the shielded model is shown in Figure 7-9. The top and bottom cover panels were circular flat plates and the side panel was a thin cylindrical shell. These panels completely enclosed a cylindrical isothermal mass representing all interior parts and components of the wheel. Conductive heat transfer between the four bodies was neglected, but ideal radiative heat transfer existed between the wheel interior and each of the panels; that

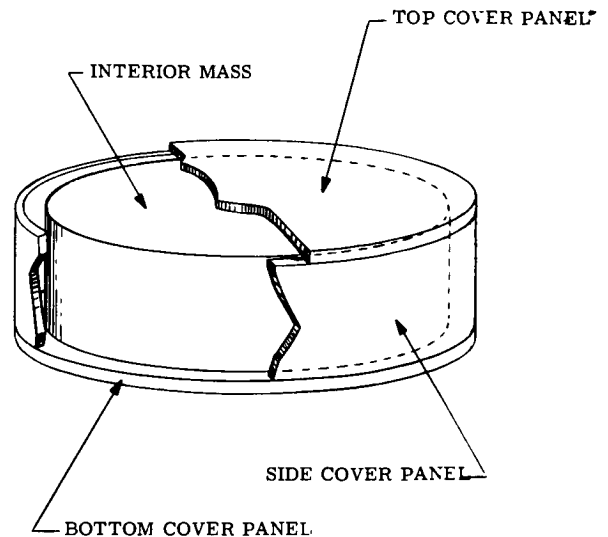


FIGURE 7-9.—Shielded model of the spacecraft wheel section.

is, the shape factor between the panels and the interior mass and the emissivity of all interior surfaces were unity.

Derivation of Heat Transfer Equations

The rate of change in temperature of each of the four bodies is given in general form by Equation (7.1):

$$\frac{dT}{dt} = \frac{q_i - q_o}{mc} \quad (7.1)$$

The radiative power received by the spacecraft through the outer surfaces of the cover panels came from either the sun or the earth. The rate at which the side panel received direct solar radiation is given by Equation (7.2). Reflected sunlight and thermal radiation from the earth were received by all outer surfaces at the rates given by Equations (7.3) and (7.4).

The thermal radiation rate from the outer surface of each of the cover panels is of the form of Equation (7.5):

$$q_{0i} = A_i \epsilon_{\oplus i} \sigma T_i^4 \quad (7.32)$$

where the radiation rate, area of radiating surface, emissivity, and temperature pertain to the i th panel.

Heat received by the interior mass was due to internal heat dissipation of electrical energy and

the net heat exchange between this mass and the three cover panels. These net heat exchange rates are given by the general expression:

$$q_{\text{net}} = \epsilon_1 \epsilon_2 F_{12} A \sigma (T_1^4 - T_2^4) \quad (7.33)$$

where

ϵ_1, ϵ_2 = emissivities of the interior surfaces
 A = surface area of the cover panel
 (assumed to be equal to the

adjacent surface of the interior mass in this model)

F_{12} = shape factor between the two surfaces

T_1, T_2 = temperatures of the two bodies

Equations (7.3), (7.4), (7.32), and (7.33) are substituted into Equation (7.1) to obtain the following heat transfer equations for the three cover panels and the interior mass:

$$\left. \begin{aligned} \frac{dT_t}{dt} &= \frac{1}{(mc)_1} [F_{1E} A_1 (\cos \beta \alpha_{\odot t} r_{\oplus} E_{\odot} + \alpha_{\oplus t} \sigma T_{\oplus}^4) - \epsilon_{\oplus t} A_1 \sigma T_t^4 - \epsilon_1 \epsilon_4 F_{14} A_1 \sigma (T_t^4 - T_w^4)] \\ \frac{dT_b}{dt} &= \frac{1}{(mc)_2} [F_{2E} A_2 (\cos \beta \alpha_{\odot b} r_{\oplus} E_{\odot} + \alpha_{\oplus b} \sigma T_{\oplus}^4) - \epsilon_{\oplus b} A_2 \sigma T_b^4 - \epsilon_2 \epsilon_4 F_{24} A_2 \sigma (T_b^4 - T_w^4)] \\ \frac{dT_s}{dt} &= \frac{1}{(mc)_3} [q_{\odot} + F_{3E} A_3 (\cos \beta \alpha_{\odot s} r_{\oplus} E_{\odot} + \alpha_{\oplus s} \sigma T_{\oplus}^4) - \epsilon_{\oplus s} A_3 \sigma T_s^4 - \epsilon_3 \epsilon_4 F_{34} A_3 \sigma (T_s^4 - T_w^4)] \\ \frac{dT_w}{dt} &= \frac{1}{(mc)_4} [q_{\oplus} + \epsilon_1 \epsilon_4 F_{14} A_1 \sigma (T_t^4 - T_w^4) + \epsilon_2 \epsilon_4 F_{24} A_2 \sigma (T_b^4 - T_w^4) + \epsilon_3 \epsilon_4 F_{34} A_3 \sigma (T_s^4 - T_w^4)] \end{aligned} \right\} \quad (7.34)$$

where

T_t, T_b, T_s, T_w = temperature of top, bottom, sides, and interior, respectively

F_{14}, F_{24}, F_{34} = shape factors between the interior body and the top, bottom, and sides, respectively

F_{1E}, F_{2E}, F_{3E} = shape factors for reception of thermal radiation from the earth by the top, bottom, and sides, respectively

Subscripts 1 or t —refers to top cover panel
 2 or b —refers to bottom cover panel
 3 or s —refers to side cover panel
 4 or w —refers to interior body

and the other symbols are the same as previously defined.

By assuming that top and bottom surfaces are of the same size and surface characteristics, and that ideal radiative heat transfer exists between adjacent interior surfaces, the following relations may be established:

$$\left. \begin{aligned} A_1 &= A_2; & F_{14} &= F_{24} = F_{34} = 1 \\ \alpha_{\odot t} &= \alpha_{\odot b}; & \epsilon_1 &= \epsilon_2 = \epsilon_3 = \epsilon_4 = 1 \\ \alpha_{\oplus t} &= \alpha_{\oplus b}; & \epsilon_{\oplus t} &= \epsilon_{\oplus b} \end{aligned} \right\} \quad (7.35)$$

To take into account the use of two different surface conditions on the outer top and bottom surfaces, relations of the form of Equations (7.31) are utilized. In addition, the change of independent variable from t to η is made. These changes and assumptions are incorporated into Equations (7.34), which are then rewritten as follows:

$$\left. \begin{aligned} \frac{dT_t}{d\eta} &= \frac{A_1}{\dot{\eta}(mc)_1} [F_{1E} (\cos \beta \alpha_{\odot t} r_{\oplus} E_{\odot} + \alpha_{\oplus t} \sigma T_{\oplus}^4) - (1 + \epsilon_{\oplus t}) \sigma T_t^4 + \sigma T_w^4] \\ \frac{dT_b}{d\eta} &= \frac{A_1}{\dot{\eta}(mc)_2} [F_{2E} (\cos \beta \alpha_{\odot b} r_{\oplus} E_{\odot} + \alpha_{\oplus b} \sigma T_{\oplus}^4) - (1 + \epsilon_{\oplus b}) \sigma T_b^4 + \sigma T_w^4] \\ \frac{dT_s}{d\eta} &= \frac{A_3}{\dot{\eta}(mc)_3} \left[\frac{A_s}{A_3} \alpha_{\odot s} E_{\odot} + F_{3E} (\cos \beta \alpha_{\odot s} r_{\oplus} E_{\odot} + \alpha_{\oplus s} \sigma T_{\oplus}^4) - (1 - \epsilon_{\oplus s}) \sigma T_s^4 + \sigma T_w^4 \right] \\ \frac{dT_w}{d\eta} &= \frac{1}{\dot{\eta}(mc)_4} [q_{\oplus} + A_1 \sigma (T_t^4 - T_w^4) + A_1 \sigma (T_b^4 - T_w^4) + A_3 \sigma (T_s^4 - T_w^4)] \end{aligned} \right\} \quad (7.36)$$

where

A_s = projected area of the side panel in the direction of the solar vector

$$\alpha_{\odot i} = \lambda \alpha_{\odot i1} + (1 - \lambda) \alpha_{\odot i2}$$

$$\alpha_{\oplus i} = \lambda \alpha_{\oplus i1} + (1 - \lambda) \alpha_{\oplus i2}$$

$$\epsilon_{\oplus i} = \lambda \epsilon_{\oplus i1} + (1 - \lambda) \epsilon_{\oplus i2}$$

λ = fractional part of top and bottom surface areas of condition 1 (shiny)

$1 - \lambda$ = fractional part of the same areas of condition 2 (painted) and the other symbols the same as previously defined.

The external shape factors, F_{1E} , F_{2E} , and F_{3E} , in Equations (7.36) are derived in the same manner as was used for the simplified model. The analytical expression for F_{3E} , for the sides, is the same as Equation (7.19); however, since in this case the top and bottom panels are not generally at the same temperature, their shape factors cannot be combined into a single expression of the form of Equation (7.18). The individual relations, Equations (7.15), (7.16), and (7.17), must be used. These complex expressions are closely approximated by the following simplified equations:

$$F_{1E} = 0.846 \cos^2 [0.6345 \cos^{-1} (-\sin \eta \sin \xi \sin \zeta - \cos \eta \cos \zeta)], \text{ if } \cos [] \geq 0$$

or

$$F_{1E} = 0, \text{ if } \cos [] < 0$$

$$F_{2E} = 0.846 \cos^2 [0.6345 \cos^{-1} (\sin \eta \sin \epsilon \sin \zeta + \cos \eta \cos \zeta)], \text{ if } \cos [] \geq 0$$

or

$$F_{2E} = 0, \text{ if } \cos [] < 0$$

CDC 1604 Computer Program and Summary of Results

Equations (7.36) were programmed for solution on the CDC 1604 digital computer utilizing the finite difference form of the equations previously employed in the G-15 program.

The values listed for the following parameters were assumed constant for this program:

$$(mc)_1 = (mc)_2 = 1.21 \text{ BTU/}^\circ\text{R}$$

$$(mc)_3 = 2.82 \text{ BTU/}^\circ\text{R}$$

$$(mc)_4 = 55.96 \text{ BTU/}^\circ\text{R}$$

$$\alpha_{\odot s} = 0.32$$

$$\alpha_{\oplus s} = \epsilon_{\oplus \rightarrow s} = 0.343$$

$$\alpha_{\odot i1} = 0.25$$

$$\alpha_{\odot i2} = 0.32$$

$$\alpha_{\oplus i1} = \epsilon_{\oplus \rightarrow i1} = 0.05$$

$$\alpha_{\oplus i2} = \epsilon_{\oplus \rightarrow i2} = 0.343$$

$$\Delta \eta = 5^\circ$$

In addition, the constant values for A_i , A_b , A_s , $E_{\odot}$, σ , $q_{\oplus}$, η , $T_{\oplus}$, R , and h listed on page 201 were used.

The variable parameters in this study were $r_{\oplus}$, λ , ξ , and ζ . The orbit and spacecraft orientation angles, ξ and ζ , were varied in the same manner as before. The earth's albedo, $r_{\oplus}$, was assumed to have the values 0.3, 0.4, and 0.5 in various computer runs, and the fraction, λ , was varied from 0.00 to 1.00. The results of this study are summarized in Figure 7-10 and Table 7-4.

Figure 7-10 shows the temperature variation of the wheel interior and the three cover panels around the minimum sunlight orbit ($\xi = 0^\circ$) for $\lambda = 1.00$ and an assumed value of 0.4 for the earth's albedo. These curves indicate that for this configuration all temperatures will remain in the range of $0^\circ - 30^\circ\text{C}$.

From the data given in Table 7-4, the following conclusions can be drawn:

1. Spacecraft attitude and orbital inclination have little effect on average temperatures, at least for the range of values of the orbital parameters used in this study.
2. For values of λ near unity (top and bottom panels shiny) average temperature gradient between wheel interior and cover panels is negligible.
3. Average temperatures near room temperature can be obtained with all of the outside surface area of the top and bottom cover panels left unpainted.

THERMAL CONTROL

Table 7-4

*Average Spacecraft Temperatures*Run No. 1; $r_{\oplus}=0.30$, $\lambda=0.00$

ξ (deg)	ζ (deg)	Average temperatures ($^{\circ}\text{K}$)			
		Top	Bottom	Sides	Interior
0	0	241	241	260	250
0	45	239	239	260	248
0	90	237	237	258	246
30	0	241	241	260	250
30	45	238	240	260	248
30	90	236	239	259	247
56.78	0	241	241	263	251

Run No. 2; $r_{\oplus}=0.40$, $\lambda=0.00$

0	0	245	245	263	254
0	45	243	243	262	252
0	90	240	240	261	250
30	0	244	244	263	253
30	45	241	244	262	252
30	90	239	243	262	250
56.78	0	243	243	265	253
56.78	45	241	244	265	253
56.78	90	240	245	265	253

Run No. 3; $r_{\oplus}=0.40$, $\lambda=0.50$

0	0	260	260	271	265
0	45	258	258	270	264
0	90	256	256	270	263
56.78	0	258	258	273	265
56.78	45	257	259	273	265
56.78	90	256	260	273	265

Run No. 4; $r_{\oplus}=0.40$, $\lambda=0.60$

0	0	264	264	273	269
0	45	262	262	273	267
0	90	261	261	272	266
56.78	0	262	262	275	268
56.78	45	261	264	275	268
56.78	90	261	264	275	268

Run No. 5; $r_{\oplus}=0.40$, $\lambda=0.70$

0	0	269	269	276	273
0	45	268	268	276	272
0	90	266	266	275	271
56.78	0	267	267	278	272
56.78	45	266	268	278	272
56.78	90	266	269	278	272

Table 7-4

*Average Spacecraft Temperatures*Run No. 6; $r_{\oplus}=0.40$, $\lambda=0.80$

ξ (deg)	ζ (deg)	Average temperatures ($^{\circ}\text{K}$)			
		Top	Bottom	Sides	Interior
0	0	275	275	280	278
0	45	273	273	279	277
0	90	272	272	279	276
56.78	0	273	273	281	277
56.78	45	272	274	282	277
56.78	90	272	275	282	277

Run No. 7; $r_{\oplus}=0.40$, $\lambda=0.90$

0	0	282	282	284	283
0	45	281	281	284	282
0	90	280	280	284	282
56.78	0	280	280	286	283
56.78	45	279	281	286	283
56.78	90	279	282	286	283

Run No. 8; $r_{\oplus}=0.40$, $\lambda=1.00$

0	0	290	290	289	291
0	45	289	289	289	290
0	90	288	288	289	289
56.78	0	288	288	291	290
56.78	45	288	290	291	290
56.78	90	288	290	291	290

Run No. 9; $r_{\oplus}=0.50$, $\lambda=1.00$

0	0	295	295	294	295
0	45	294	294	293	294
0	90	293	293	293	294
56.78	0	291	291	293	293
56.78	45	290	293	293	293
56.78	90	290	293	294	293

Figure 7-11 summarizes the results of the two solutions of the heat transfer problem for the wheel for the case, $\epsilon=0^{\circ}$, $\zeta=0^{\circ}$. The agreement of results for the two models for $r_{\oplus}=0.40$ is quite good, verifying the result of the shielded model solution that very small average temperature gradients exist between the wheel interior and the outside cover panels.

Performance

In general, the temperatures measured during the spacecraft orbit were in agreement with the computed temperatures. The major reason for variation from predicted temperatures was that the top and bottom surfaces of the exposed wheel were only 90 percent polished aluminum rather than the 100 percent assumed in the

$$r_{\oplus} = 0.40; \quad \lambda = 1.00; \quad \text{MINIMUM SUNLIGHT ORBIT}$$

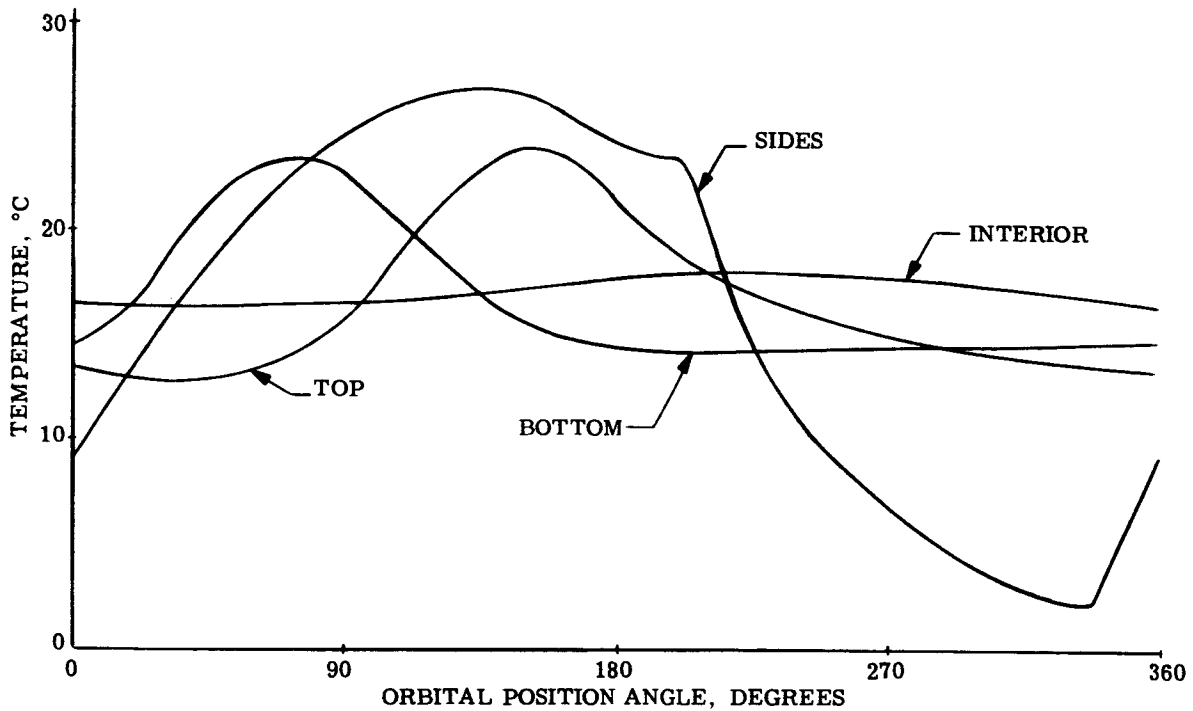


FIGURE 7-10.—Orbital position angle.

computations. As shown in Figure 7-12, this difference would theoretically be expected to reduce temperatures of the interior of the spacecraft to a mean of 9°C, 8°C lower than the value computed for the 100 percent polished condition. The temperature curves of Appendix B indicate the results accomplished by the passive temperature control work. The azimuth casting temperature was about 2°C and the electronics package on the back of the sail structure maintained a temperature between 0° and 10°C. The solar panels were about 15° C cooler than expected with the high day temperatures being about 60°C and the low night temperatures being about -35°C.

EVALUATION OF THE SHAPE FACTOR INTEGRALS

Flat Surfaces of Spacecraft

Figure 7-2 shows the geometry of the plate and the earth disk. Note that the normal to the

plate is always in the yz -plane. The polar coordinate angle is ϕ while θ is the angle between the z -axis and the line between the plate and dS . The angle between the z -axis and the plate normal is γ , while r is the polar coordinate distance from the origin to dS , the area element on the earth disk. The distance from dS to the plate is ρ . Thus Equation (7.12) becomes

$$F = \frac{1}{\pi} \iint (\sin^2 \theta \sin \phi \sin \gamma + \sin \theta \cos \theta \cos \gamma) d\phi d\theta \quad (7.37)$$

Where the integration is to be performed over the region of the disk "seen" by the plate.

$$\text{Case 1: } (0 \leq |\gamma| < \frac{\pi}{2} - \theta_0):$$

Figure 7-13 shows the limits of plate positions for Case 1. Since the entire earth disk is

ORBITING SOLAR OBSERVATORY

- ① SHIELDED MODEL OF WHEEL, RUN ON CDC 1604 ($r_{\oplus} = 0.40$)
POINT A: $r_{\oplus} = 0.30$; POINT B: $r_{\oplus} = 0.50$
- ② SIMPLIFIED MODEL, RUN ON G-15 ($r_{\oplus} = 0.40$)
- ③ SAME AS 2; WITH $r_{\oplus} = 0.30$
POINT C: $r_{\oplus} = 0.50$; TOP AND BOTTOM ASSUMED INSULATED

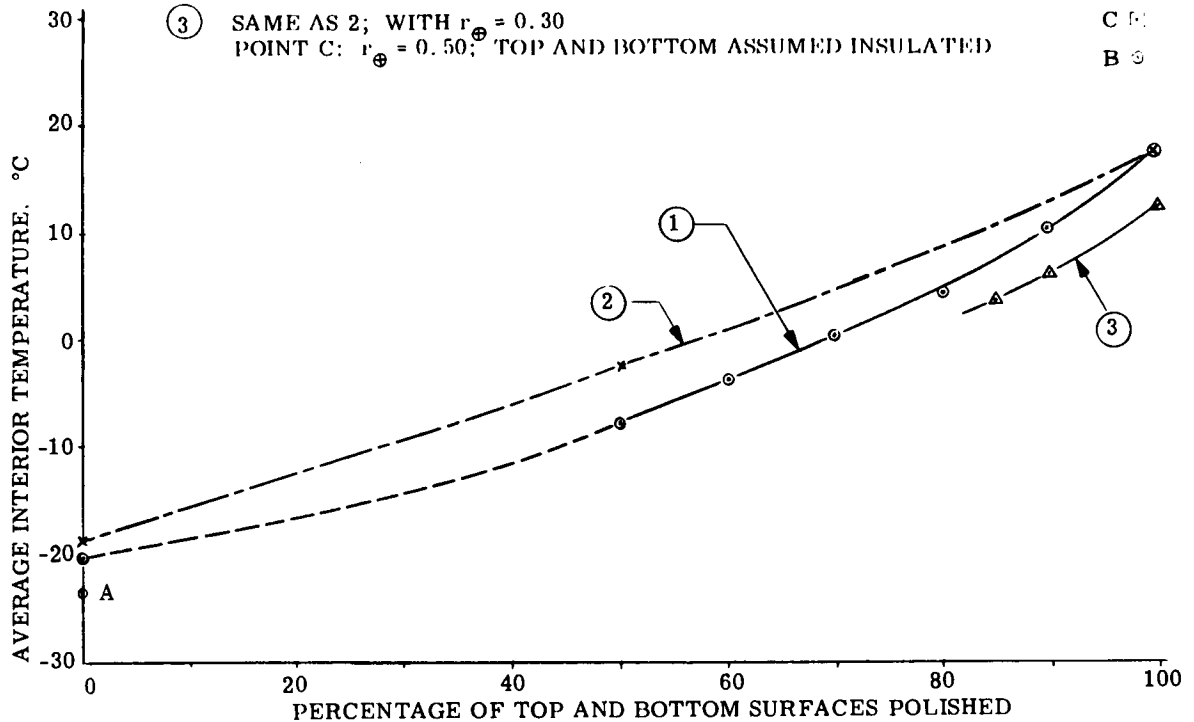


FIGURE 7-11.—Percentage of top and bottom surfaces polished.

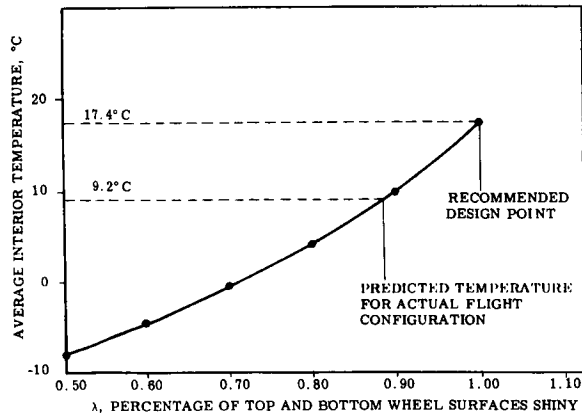


FIGURE 7-12.—Average interior wheel temperature as function of top and bottom surface conditions.

“seen” by the plate, the limits on ϕ are 0 to 2π and on θ are 0 to θ_0 . It is possible to take

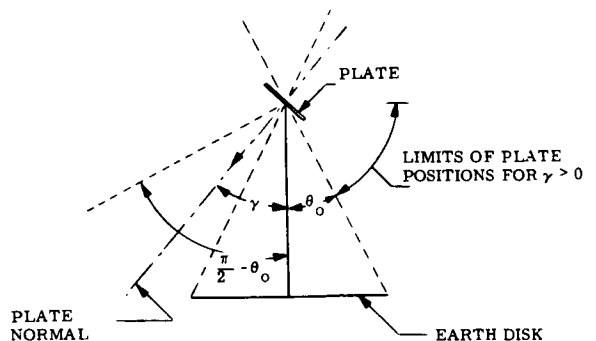


FIGURE 7-13.—Limits of plate position (case 1).

advantage of symmetry, however, and write

$$F = \frac{2}{\pi} \int_0^{\theta_0} \int_{-\pi/2}^{\pi/2} (\sin^2 \theta \sin \phi \sin \gamma + \sin \theta \cos \theta \cos \gamma) d\phi d\theta$$

Evaluation of this integral proceeds as follows:

$$\begin{aligned}
 F &= \frac{2}{\pi} \int_0^{\theta_0} \sin^2 \theta \sin \gamma (-\cos \theta) \Big|_{-\pi/2}^{\pi/2} d\theta + \frac{2}{\pi} \int_0^{\theta_0} \sin \theta \cos \theta \cos \gamma [\phi] \Big|_{-\pi/2}^{\pi/2} d\theta \\
 &= 0 + \frac{2}{\pi} \cos \gamma [\sin^2 \theta]_0^{\theta_0} = \cos \gamma \sin^2 \theta_0
 \end{aligned} \tag{7.38}$$

Case 2: $(\frac{\pi}{2} - \theta_0 \leq |\gamma| < \frac{\pi}{2})$:

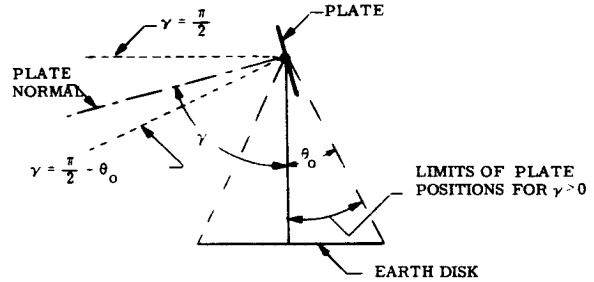


FIGURE 7-14.—Limits of plate position and region-integration (case 2).

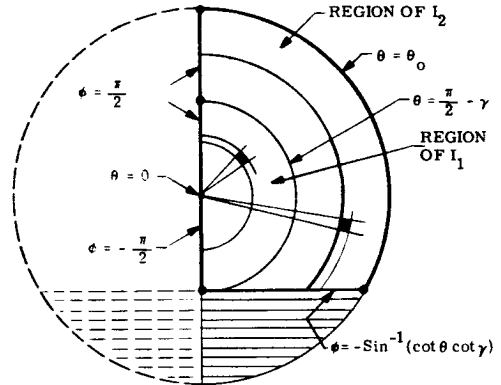


Figure 7-14 shows the limits of plate positions when the plate "sees" more than half, but not all, of the earth disk. Also shown is a diagram of the two regions of the earth disk to be covered in the evaluation of the integral. Again symmetry properties may be used to write

$$\begin{aligned}
 F &= \frac{2}{\pi} \int_0^{\pi/2-\gamma} \int_{\pi/2}^{\pi/2} (\sin^2 \theta \sin \phi \sin \gamma + \sin \theta \cos \phi \cos \gamma) d\phi d\theta \\
 &\quad + \frac{2}{\pi} \int_{\pi/2-\gamma}^{\theta_0} \int_{-\sin^{-1} \cot \theta \cot \gamma}^{\pi/2} (\sin^2 \theta \sin \phi \sin \gamma + \sin \theta \cos \phi \cos \gamma) d\phi d\theta \\
 &= I_1 + I_2
 \end{aligned}$$

$$I_1 = 0 + \frac{2}{\pi} [\sin^2 \theta]_0^{\pi/2-\gamma} \cos \gamma = \cos^3 \gamma \tag{7.39}$$

$$\begin{aligned}
 I_2 &= -\frac{2}{\pi} \int_{\pi/2-\gamma}^{\theta_0} \sin^2 \theta \sin \gamma [\cos \phi]_{-\sin^{-1}(\cot \theta \cot \gamma)}^{\pi/2} d\theta + \frac{2}{\pi} \int_{\pi/2-\gamma}^{\theta_0} \sin \theta \cos \theta \cos \gamma [\phi]_{-\sin^{-1}(\cot \theta \cot \gamma)}^{\pi/2} d\theta \\
 &= \frac{2}{\pi} \int_{\pi/2-\gamma}^{\theta_0} \sin \gamma \sin^2 \theta \sqrt{1 - \cot^2 \theta \cot^2 \gamma} d\theta + \int_{\pi/2-\gamma}^{\theta_0} \sin \theta \cos \theta \cos \gamma d\theta \\
 &\quad + \frac{2}{\pi} \int_{\pi/2-\gamma}^{\theta_0} \sin \theta \cos \theta \cos \gamma \sin^{-1}(\cot \theta \cot \gamma) d\theta \\
 &= I_{2.1} + I_{2.2} + I_{2.3}
 \end{aligned}$$

In $I_{2.1}$ let $x = \sin^2 \theta$. Then $dx = 2 \sin \theta \cos \theta d\theta$, and

$$d\theta = \frac{dx}{2 \sin \theta \cos \theta} = \frac{dx}{2\sqrt{x}\sqrt{1-x}}$$

Thus

$$\begin{aligned} I_{2.1} &= \frac{2}{\pi} \sin \gamma \int_{\cos^2 \gamma}^{\sin^2 \theta_0} x \sqrt{1 - \left(\frac{1}{x} - 1\right) \cot^2 \gamma} \frac{dx}{2\sqrt{x}\sqrt{1-x}} \\ &= \frac{\sin \gamma}{\pi} \int_{\cos^2 \gamma}^{\sin^2 \theta_0} \left(\frac{x \csc^2 \gamma - \cot^2 \gamma}{1-x} \right)^{1/2} dx \\ &= -\frac{1}{\pi} \sin \gamma \left\{ \left[\sqrt{(1-x)(x \csc^2 \gamma - \cot^2 \gamma)} \right]_{\cos^2 \gamma}^{\sin^2 \theta_0} - \frac{1}{2} \int_{\cos^2 \gamma}^{\sin^2 \theta_0} [(1-x)(x \csc^2 \gamma - \cot^2 \gamma)]^{-1/2} dx \right\} \\ &= -\frac{1}{\pi} \cos \theta_0 \sqrt{\sin^2 \theta_0 - \cos^2 \gamma} - \frac{1}{\pi} \sin^2 \gamma \tan^{-1} \left(\frac{\cos \theta_0}{\sqrt{\sin^2 \theta_0 - \cos^2 \gamma}} \right) + \frac{1}{2} \sin^2 \gamma \end{aligned} \quad (7.40)$$

$$I_{2.2} = \frac{1}{2} \cos \gamma [\sin^2 \theta]_{\pi/2-\gamma}^{\theta_0} = \frac{1}{2} \sin^2 \theta_0 \cos \gamma - \frac{1}{2} \cos^3 \gamma \quad (7.41)$$

$I_{2.3}$ may be integrated by parts, letting $u = \sin^{-1}(\cot \theta \cot \gamma)$ and $dv = \sin \theta \cos \theta d\theta$. Then

$$I_{2.3} = \frac{1}{\pi} \cos \gamma \sin^2 \theta_0 \sin^{-1}(\cot \gamma \cot \theta_0) - \frac{1}{2} \cos^3 \gamma + \frac{1}{\pi} \cos \gamma \frac{\cot \gamma}{|\cot \gamma|} \int_{\pi/2-\gamma}^{\theta_0} \frac{\tan \theta d\theta}{\sqrt{\tan^2 \gamma \tan^2 \theta - 1}} \quad (7.42)$$

Let this remaining integral be called $I_{2.3.1}$. In this integral, let $x = \tan \gamma \tan \theta$, so that $dx = \tan \gamma \sec^2 \theta d\theta$, or

$$d\theta = \frac{dx \cot \gamma}{1+x^2 \cot^2 \gamma}$$

Then

$$\begin{aligned} I_{2.3.1} &= \frac{1}{\pi} \cos \gamma \frac{\cot^3 \gamma}{|\cot \gamma|} \int_1^{\tan \gamma \tan \theta_0} \frac{x dx}{(1-x^2 \cot^2 \gamma) \sqrt{x^2 - 1}} \\ &= \frac{1}{\pi} \cos \gamma \frac{\cot^3 \gamma}{|\cot \gamma|} \frac{1}{\cot^2 \gamma} \sqrt{\frac{\cot^2 \gamma}{1+\cot^2 \gamma}} \left[\tan^{-1} \sqrt{\frac{\cot^2 \gamma (x^2 - 1)}{1+\cot^2 \gamma}} \right]_1^{\tan \gamma \tan \theta_0} \\ &= \frac{1}{\pi} \frac{\cos \gamma \cot \gamma}{|\csc \gamma|} \tan^{-1} \sqrt{\cos^2 \gamma (\tan^2 \gamma \tan^2 \theta_0 - 1)} \end{aligned}$$

or

$$I_{2.3.1} = \frac{1}{\pi} \cos^2 \gamma \operatorname{sgn}(\sin \gamma) \tan^{-1}(|\cos \gamma| \sqrt{\tan^2 \gamma \tan^2 \theta_0 - 1}) \quad (7.43)$$

If $0 \leq \gamma \leq \pi$ it is not necessary to include the factor $\operatorname{sgn}(\sin \gamma)$ in this expression. The shape factor for Case 2 is, therefore,

$$\begin{aligned} F &= \cos^3 \gamma - \frac{1}{\pi} \cos \theta_0 \sqrt{\sin^2 \theta_0 - \cos^2 \gamma} - \frac{1}{\pi} \sin^2 \gamma \tan^{-1} \left(\frac{\cos \theta_0}{\sqrt{\sin^2 \theta_0 - \cos^2 \gamma}} \right) \\ &\quad + \frac{1}{2} \sin^2 \gamma + \frac{1}{2} \sin^2 \theta_0 \cos \gamma - \frac{1}{2} \cos^3 \gamma + \frac{1}{\pi} \cos \gamma \sin^2 \theta_0 \sin^{-1}(\cot \gamma \cot \theta_0) \\ &\quad - \frac{1}{2} \cos^3 \gamma + \frac{1}{\pi} \cos^2 \gamma \operatorname{sgn}(\sin \gamma) \tan^{-1}(|\cos \gamma| \sqrt{\tan^2 \gamma \tan^2 \theta_0 - 1}) \end{aligned}$$

or,

$$F = \frac{1}{2} \sin^2 \theta_0 \cos \gamma + \frac{1}{2} \sin^2 \gamma - \frac{1}{\pi} \cos \theta_0 \sqrt{\sin^2 \theta_0 - \cos^2 \gamma} - \frac{1}{\pi} \sin^2 \gamma \tan^{-1} \left(\frac{\cos \theta_0}{\sqrt{\sin^2 \theta_0 - \cos^2 \gamma}} \right) + \frac{1}{\pi} \cos \lambda$$

$$\sin^2 \theta_0 \sin (\cot \gamma \cot \theta_0) + \frac{1}{\pi} \cos^2 \gamma \operatorname{sgn} (\sin \gamma) \tan^{-1} (|\cos \gamma| \sqrt{\tan^2 \gamma \tan^2 \theta_0 - 1}) \quad (7.44)$$

Note that as $\gamma \rightarrow \frac{\pi}{2} - \theta_0$, F approaches

$$F = \frac{1}{2} \sin^3 \theta_0 + \frac{1}{2} \cos^2 \theta_0 - \frac{1}{\pi} (\cos^2 \theta_0) \left(\frac{\pi}{2} \right) + \frac{1}{\pi} (\sin^3 \theta_0) \left(\frac{\pi}{2} \right)$$

$$= \sin^3 \theta_0$$

which is the same as F in Case 1 for $\gamma = \frac{\pi}{2} - \theta_0$.

Also, for $\gamma = \frac{\pi}{2}$, $F = \frac{1}{2} - \frac{1}{\pi} \sin \theta_0 \cos \theta_0 - \frac{1}{\pi} (1) \left(\frac{\pi}{2} - \theta_0 \right) = \frac{1}{\pi} (\theta_0 - \sin \theta_0 \cos \theta_0)$.

This must agree with the value of F in Case 3 for the same angle.

Case 3: $\left(\frac{\pi}{2} \leq \gamma < \frac{\pi}{2} + \theta_0 \right)$ (see Figure 7-15):

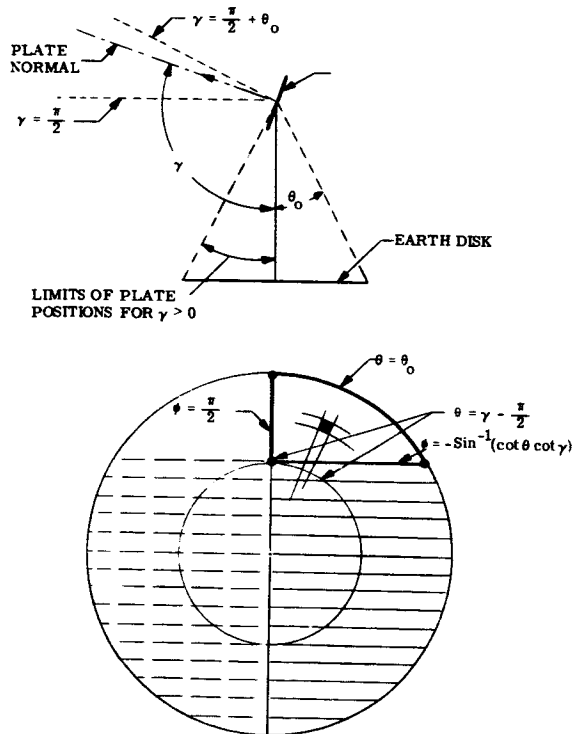


FIGURE 7-15.—Limits of plate position and region of integration (case 3).

In this case the integral for F becomes

$$\begin{aligned}
 F &= \frac{2}{\pi} \int_{\gamma-\pi/2}^{\theta_0} \int_{-\sin^{-1}(\cot \theta \cot \gamma)}^{\pi/2} (\sin^2 \theta \sin \phi \sin \gamma + \sin \theta \cos \theta \cos \gamma) d\phi d\theta \\
 &= -\frac{2}{\pi} \int_{\gamma-\pi/2}^{\theta_0} \sin^2 \theta \sin \gamma [\cos \phi]_{-\sin^{-1}(\cot \theta \cot \gamma)}^{\pi/2} d\theta + \frac{2}{\pi} \int_{\gamma-\pi/2}^{\theta_0} \cos \gamma \sin \theta \cos \theta [\phi]_{-\sin^{-1}(\cot \theta \cot \gamma)}^{\pi/2} d\theta \\
 &= \frac{2}{\pi} \sin \gamma \int_{\gamma-\pi/2}^{\theta_0} \sin^2 \theta \sqrt{1 - \cot^2 \theta \cot^2 \gamma} d\theta + \cos \gamma \int_{\gamma-\pi/2}^{\theta_0} \sin \theta \cos \theta d\theta \\
 &\quad + \frac{2}{\pi} \cos \gamma \int_{\gamma-\pi/2}^{\theta_0} \sin \theta \cos \theta \sin^{-1}(\cot \theta \cot \gamma) d\theta \\
 &= J_{2.1} + J_{2.2} + J_{2.3}
 \end{aligned}$$

Since the integrands are the same as those in Case 2, we have for $J_{2.1}$:

$$\begin{aligned}
 J_{2.1} &= \frac{\sin \gamma}{\pi} \int_{\cos^2 \theta_0}^{\sin^2 \theta_0} \left(\frac{x \csc^2 \gamma - \cot^2 \gamma}{1-x} \right)^{1/2} dx \\
 &= \frac{1}{\pi} \cos \theta_0 \sqrt{\sin^2 \theta_0 - \cos^2 \gamma} - \frac{1}{\pi} \sin^2 \gamma \tan^{-1} \left(\frac{\cos \theta_0}{\sqrt{\sin^2 \theta_0 - \cos^2 \gamma}} \right) + \frac{1}{2} \sin^2 \gamma
 \end{aligned} \tag{7.45}$$

Similarly, we determine that

$$\begin{aligned}
 J_{2.2} &= \frac{1}{2} \cos \gamma [\sin^2 \theta]_{\gamma-\pi/2}^{\theta_0} = \frac{1}{2} \sin^2 \theta_0 \cos \gamma - \frac{1}{2} \cos^3 \gamma \\
 J_{2.3} &= \frac{2}{\pi} \cos \gamma \left\{ \left[\frac{1}{2} \sin^2 \theta \sin^{-1}(\cot \theta \cot \gamma) \right]_{\gamma-\pi/2}^{\theta_0} + \frac{1}{\pi} \frac{\cot \gamma}{|\cot \gamma|} \int_{\gamma-\pi/2}^{\theta_0} \frac{\tan \theta d\theta}{\sqrt{\tan^2 \gamma \tan^2 \theta - 1}} \right\} \\
 &= \frac{1}{\pi} \cos \gamma \sin^2 \theta_0 \sin^{-1}(\cot \theta_0 \cot \gamma) - \frac{1}{\pi} \cos^3 \gamma \sin^{-1}(-1) \\
 &\quad + \frac{1}{\pi} (\cos \gamma) \frac{\cot^3 \gamma}{|\cot \gamma|} \frac{1}{\cot^2 \gamma} \sqrt{\frac{\cot^2 \gamma}{1 + \cot^2 \gamma}} \left[\tan^{-1} \sqrt{\frac{\cot^2 \gamma (\chi^2 - 1)}{1 + \cot^2 \gamma}} \right]_{-1}^{\tan \gamma \tan \theta_0}
 \end{aligned} \tag{7.46}$$

or

$$J_{2.3} = \frac{1}{\pi} \cos \gamma \sin^2 \theta_0 \sin^{-1}(\cot \theta_0 \cot \gamma) + \frac{1}{2} \cos^3 \gamma + \frac{1}{\pi} \cos^2 \gamma \operatorname{sgn}(\sin \gamma) \tan^{-1}(|\cos \gamma| \sqrt{\tan^2 \theta_0 \tan^2 \gamma - 1}) \tag{7.47}$$

The total results for F are:

$$\begin{aligned}
 F &= \frac{1}{\pi} \cos \theta_0 \sin^2 \theta_0 \cos^2 \gamma - \frac{1}{\pi} \sin^2 \gamma \tan^{-1} \left(\frac{\cos \theta_0}{\sqrt{\sin^2 \theta_0 \cos^2 \gamma}} \right) \\
 &\quad + \frac{1}{2} \sin^2 \gamma + \frac{1}{2} \sin^2 \theta_0 \cos \gamma + \frac{1}{2} \cos^3 \gamma + \frac{1}{\pi} \cos \gamma \sin^2 \theta_0 \sin^{-1}(\cot \theta_0 \cot \gamma) \\
 &\quad + \frac{1}{2} \cos^3 \gamma + \frac{1}{\pi} \cos^2 \gamma \operatorname{sgn}(\sin \gamma) \tan^{-1}(|\cos \gamma| \sqrt{\tan^2 \gamma \tan^2 \theta_0 - 1})
 \end{aligned}$$

or

$$F = \frac{1}{2} \sin^2 \theta_0 \cos \gamma + \frac{1}{2} \sin^2 \gamma - \frac{1}{\pi} \cos \theta_0 \sqrt{\sin^2 \theta_0 - \cos^2 \gamma} = \frac{1}{\pi} \sin^2 \gamma \tan^{-1} \left(\frac{\cos \theta_0}{\sqrt{\sin^2 \theta_0 - \cos^2 \gamma}} \right) + \frac{1}{\pi} \cos \gamma$$

$$\sin^2 \theta_0 \sin^{-1} (\cot \theta_0 \cot \gamma) + \frac{1}{\pi} \cos^2 \gamma \operatorname{sgn} (\sin \gamma) \tan^{-1} (|\cos \gamma| \sqrt{\tan^2 \gamma \tan^2 \theta_0 - 1}) \quad (7.48)$$

This expression is identical with that for Case 2 and therefore

$$F\left(\frac{\pi}{2}\right) \text{ is } 1 (\theta_0 - \sin \theta_0 \cos \theta_0).$$

As $\gamma \rightarrow \frac{\pi}{2} + \theta_0$, F should reduce to zero:

$$F = -\frac{1}{2} \sin^3 \theta_0 + \frac{1}{2} \cos^2 \theta_0 - \frac{1}{\pi} \cos \theta_0 \sin^2 \theta_0 - \sin^2 \theta_0 - \frac{1}{\pi} \cos^2 \theta_0 \left(\frac{\pi}{2}\right) - \frac{1}{\pi} \sin^3 \theta_0 \sin^{-1} (-1) + \frac{1}{\pi} \sin^2 \theta_0 \tan^{-1} (0)$$

$$= -\frac{1}{2} \sin^3 \theta_0 + \frac{1}{2} \cos^2 \theta_0 - \frac{1}{2} \sin^3 \theta_0 = 0$$

Case 4: $\left(\frac{\pi}{2} + \theta_0 \leq |\gamma| \leq \pi\right)$

(see Figure 7-16)

By inspection, $F=0$.

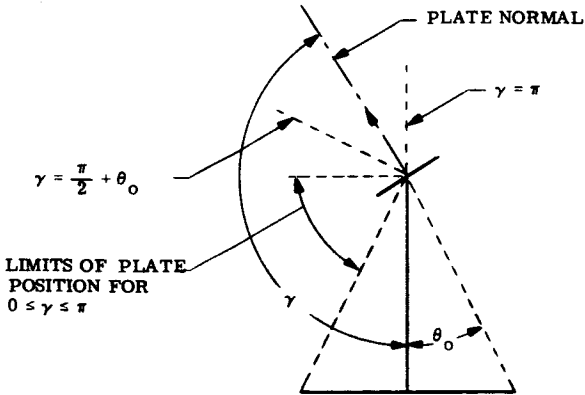


FIGURE 7-16.—Limits of plate position (case 4)

Lateral Surface of Cylinder Figure 7-17 shows a cylindrical surface receiving incident radiation from the direction of $\vec{u}$ making an angle α_s with the axis of the cylinder. Let

$$dS_2 = r d\lambda dh$$

be an element of surface area of the cylinder, where r is the radius of the cylinder, λ is the coordinate angle as shown, and h is the distance above the base plane of the cylinder. Let axes be chosen with the z -axis as the axis of the cylinder and with the radiation vector always in the xz -plane.

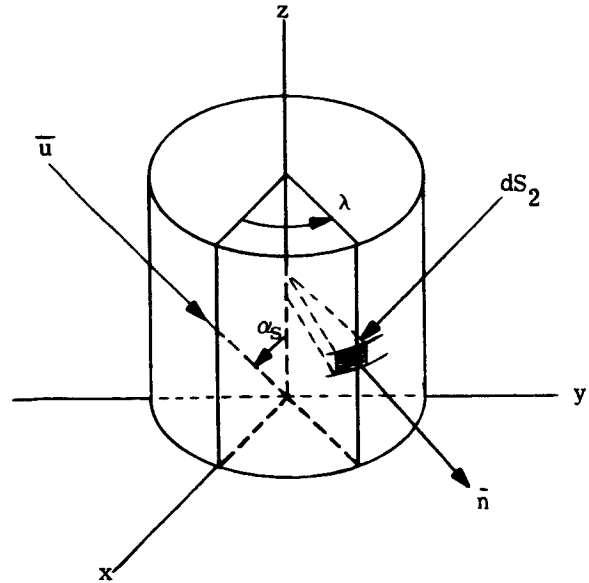


FIGURE 7-17.—Geometry of radiation on cylindrical surface.

Then

$$\mathbf{u} = -\mathbf{i} \sin \alpha_s - k \cos \alpha_s$$

and

$$\mathbf{n} = \mathbf{i} \cos \lambda + \mathbf{j} \sin \lambda$$

Since radiation is received only over the portion of the cylinder for which

$$-\frac{\pi}{2} \leq \lambda \leq \frac{\pi}{2},$$

the effective area in the direction of $\bar{\mathbf{u}}$ is

$$-\int (\bar{\mathbf{n}} \cdot \bar{\mathbf{u}}) dS_2 = \int_0^h \int_{-\pi/2}^{\pi/2} \cos \lambda \sin \alpha_s r d\lambda dh = 2rh \sin \alpha_s$$

Since the lateral area of the cylinder is $2\pi rh$ the fraction of the lateral area capturing radiation is

$$\frac{1}{\pi} \sin \alpha_s$$

If we now place the cylinder over the earth disk we can write the integral for the shape factor as

$$F = \frac{1}{\pi} \int \frac{\cos \phi_1}{\rho^2} \left(\frac{1}{\pi} \sin \alpha_s \right) dS$$

or

$$F = \frac{1}{\pi^2} \int \int \frac{\cos \theta}{h^2 \sec^2 \theta} \sin \alpha_s h^2 \sec^2 \theta \tan \theta d\phi d\theta \quad (7.49)$$

where the integration is over the entire earth disk (except for a small *hole* formed by the projection of the cylinder along its axis to intersect with the disk. This *unseen* region can be neglected because of the small size of the cylinder in comparison with the earth disk). The counterpart of ϕ_2 in the case of the flat plate is α_s .

Hence,

$$\sin \alpha_s = \sqrt{1 - (\sin \theta \sin \phi \sin \gamma + \cos \theta \cos \gamma)^2}$$

and

$$F = \frac{2}{\pi^2} \int_{-\pi/2}^{\pi/2} \int_{\theta}^{\theta_0} \sin \theta \sqrt{1 - (\sin \theta \sin \phi \sin \gamma + \cos \theta \cos \gamma)^2} d\theta d\phi \quad (7.50)$$

This integral is virtually impossible to evaluate in closed form for arbitrary γ , so it was decided to evaluate for $\gamma=0$ and $\gamma=\frac{\pi}{2}$ and investigate

the variation of F .

For $\gamma=0$, we have

$$F = \frac{2}{\pi^2} \int_{-\pi/2}^{\pi/2} \int_0^{\theta_0} \sin \theta \sqrt{1 - \cos^2 \theta} d\theta d\phi = \frac{2}{\pi^2} \int_{-\pi/2}^{\pi/2} \int_0^{\theta_0} \sin^2 \theta d\theta d\phi = \frac{1}{\pi} (\theta_0 - \sin \theta_0 \cos \theta_0) \quad (7.51)$$

This result agrees with that for one side of a flat plate with $\gamma=\frac{\pi}{2}$, which is to be expected.

For $\gamma=\frac{\pi}{2}$, we determine that

$$F = \frac{2}{\pi^2} \int_0^{\theta_0} \int_{-\pi/2}^{\pi/2} \sin \theta \sqrt{1 - \sin^2 \theta \sin^2 \theta} d\phi d\theta = \frac{4}{\pi^2} \int_0^{\theta_0} \sin \theta \int_0^{\pi/2} \sqrt{1 - \sin^2 \theta \sin^2 \theta} d\phi d\theta = \frac{4}{\pi^2} \int_0^{\theta_0} \sin \theta E\left(\frac{\pi}{2}, \sin \theta\right) d\theta \quad (7.52)$$

where E is an elliptic integral. Numerical integration of this result gave

$$F = \frac{4}{\pi^2} (0.816) = 0.331$$

The extreme values of $F(\gamma)$ were thus

$$F(0) = \frac{1}{\pi} (\theta_0 - \sin \theta_0 \cos \theta_0) = 0.254$$

$$F\left(\frac{\pi}{2}\right) = 0.331$$

Using the same type of approximate representation for the shape factor of the lateral surface of the cylinder as for the two-sided flat plate, we have

$$F(\gamma) = 0.254 + 0.077 \sin^2 \gamma \quad (7.53)$$

Chapter 8

ORBITING SOLAR OBSERVATORY SCIENTIFIC EXPERIMENTS

GENERAL DESCRIPTION

The study of the sun is important, not only because of immediate scientific gains which will support our man-in-space program (we know, for example, that radiation from solar eruptions or flares occurring during a manned spaceflight would penetrate the spacecraft and harm the occupants), but also because long-range scientific gains will help us more fully understand the effect of the sun on the earth. The purpose of most of the OSO experiments was to measure broad ranges of electromagnetic radiation from the sun in the ultraviolet, x-ray, and gamma-ray regions. In the past, it was difficult to obtain accurate information about solar radiation because of absorption and attenuation by the atmosphere. The OSO I spacecraft permitted precise and extensive measurements to be made without the interference of the atmosphere.

Thirteen experiments aboard the spacecraft measured wavelengths ranging from ultraviolet light through the very short high-energy gamma wavelengths. One of the several Goddard Space Flight Center experiments aboard was an x-ray spectrometer. Information obtained from this instrument may lead to a theory for predicting solar flares. The University of Minnesota instrument will tell us more about the thermal-nuclear processes that take place on the sun. The University of Rochester instrument will measure gamma rays with energies greater than 10^8 electron volts. Other experiments included the Ames Research Center instrument for studying emissivity stability of material, Goddard instruments to monitor solar x-ray emission and to measure the number and energies of interplanetary dust particles, and the University of California neutron, proton,

and electron detectors. Each of the thirteen scientific instruments in the spacecraft functioned properly for the first 1000 orbits.

The scientific instruments were carried both in the rotating wheel and the solar oriented upper structure. The wheel experiments were, in general, sky-mapping experiments comparing radiation from the sun to that in other portions of space. These experiments were exposed to the sun once every two seconds due to the rotation of the wheel and always passed within a line-of-sight angle of three degrees of the sun in elevation due to the controlled attitude of the spin axis. The oriented experiments were continuously pointed to the center of the sun to an accuracy of about one minute of arc.

SOLAR ORIENTED EXPERIMENTS

The solar oriented instrument gimbal mounting was designed to carry 75 pounds of instruments. These instruments were confined to a volume of space 38" x 8" x 8". The OSO I spacecraft carried two solar-oriented instruments. One was an x-ray spectrometer measuring the spectral range from 10 to 400 Å. The spectrometer weighed 20.5 pounds. The second instrument was an assembly of several different experiments. These were: two solar x-ray (20 to 100 keV and 1 to 8 Å) monitoring experiments, a gamma-ray (0.510 MeV) monitoring experiment, an interplanetary dust particle experiment, and an experiment designed to monitor the orbital environment effect of degradation on photoelectric solar error sensors that were used in the spacecraft servo control positioning systems. The composite instrument weighed 45 pounds. Figure 8-1 shows the location of these instruments in the spacecraft.

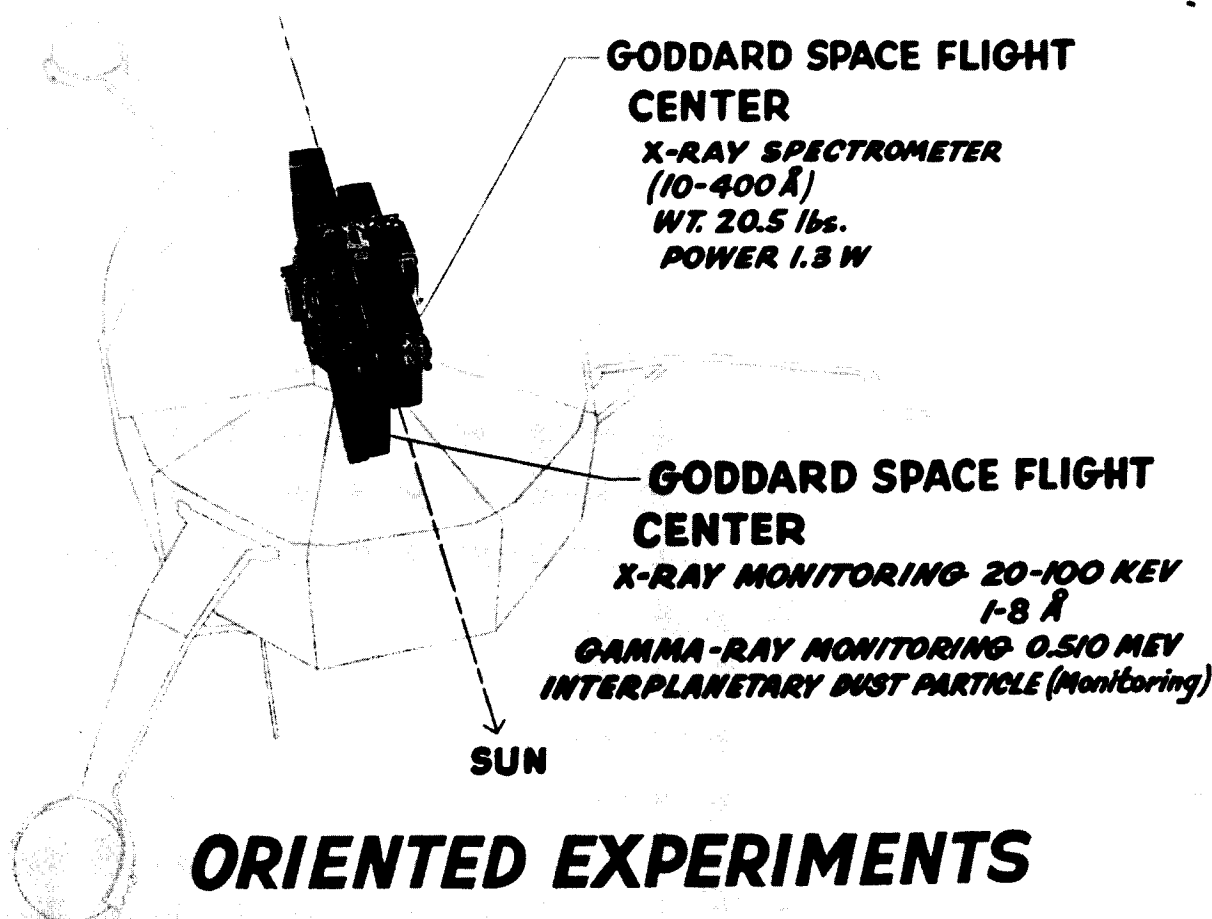


FIGURE 8-1.—Solar-oriented experiment.

Goddard X-Ray Spectrometer Experiment

One of the primary tasks set for the OSO I was the measurement of the solar spectrum over the wavelength range of 10 to 400 Å. The instrument designed to accomplish this was an x-ray spectrometer which had a sensitivity high enough to yield a satisfactory spectral distribution over the given wavelengths.

The angular aperture of the spectrometer was approximately 1.2 by 2.2 degrees. Hence, pointed with moderate accuracy (within plus or minus a few minutes of arc to the center of the solar disk), the spectrometer input encompassed the entire solar disk. The spectrometer was designed so that sunlight falling on the front face of the instrument (Figure 8-2) would pass through the entrance slit and strike a concave grating mounted for grazing incidence.

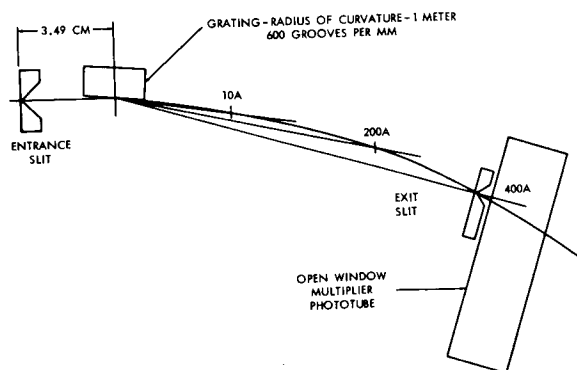


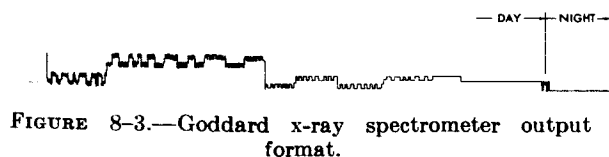
FIGURE 8-2.—Goddard x-ray spectrometer optical schematic.

The angle of incidence was 88 degrees. The grating was lightly ruled in a special glass by the Nobel Institute in Stockholm at 576 lines

per mm on a one-meter radius of curvature blank. No reflecting coatings of any kind were used on this grating. The diffracted rays then continue to the exit slit in front of the detector. The detector was mounted on a carriage moving on a rail so that the detector input slit would follow a *Rowland* circle where the spectrum was in focus. The plane of the detector slit remained approximately perpendicular to the diffracted ray at any position along the rail, thereby keeping the spectral passband nearly constant for all angles of diffraction. The 50-micron entrance and exit slits provided a spectral passband of 1.7 Å and permitted resolution of lines 0.85 Å apart.

The detector used was an open-window photomultiplier developed by the Bendix Corporation specifically for use in this spectrometer and designated the M-306. Its operation was as follows: Photoelectrons from the exposed tungsten cathode move along cycloidal paths in crossed electric and magnetic fields between two glass strips coated with a semiconducting secondary-emitting oxide layer. One of the glass strips serves as a continuous dynode. Each photoelectron is multiplied into a pulse of approximately 10^6 electrons at the anode.

The data format of the x-ray spectrometer experiment used a pulse-counting technique whereby narrow, randomly occurring pulses from the detection system were converted to bi-value waveforms by means of binary scalars. To extend the dynamic range over a wider range of input pulse rates, three different scaling factors were used. The three bi-valued waveforms thus generated were then coded and combined in such a way as to permit later unambiguous identification of the particular contributions of the different scaling factors to the composite waveform. A typical data output format is shown in Figure 8-3. This format was the input of the 1200-cps subcarrier oscillator providing the frequency modulating for the data transmitted.



The spectrometer used about 1.3 watts supplied from the spacecraft power supply at a nominal 18 volts dc. About one watt was used by an oscillator which drove the three-phase synchronous detector scandrive motor. The motor required about 300 mw at 137 cps to yield 100 mw of mechanical output power. The remainder of the power was used in the multiplier and pulse-handling circuitry. Power to the experiment was switched off when the spacecraft entered the earth's shadow and switched on at the beginning of a new spacecraft day.

Since the diffraction grating was quite susceptible to contamination, all of the exposed materials inside the spectrometer were tested at a pressure of about 10^{-6} mm Hg to eliminate any material having a high vapor pressure. Because the electronic circuits were potted in a compound having a high vapor pressure, they were enclosed in sealed boxes which were vented to the outside through holes in the spectrometer base plate. During assembly, all the parts exposed to the interior were carefully cleaned to be free of oil, grease, and other contaminants.

The temperature of the instrument was controlled by controlling the radiation balance. Most of the outer case was polished. The central section was black in order to thermally "tie" the experiment to the spacecraft. A thermistor was mounted in the experiment to monitor temperature during orbital flight. A plot of this temperature for the first 1,000 orbits is shown in Figure 8-4. The output of

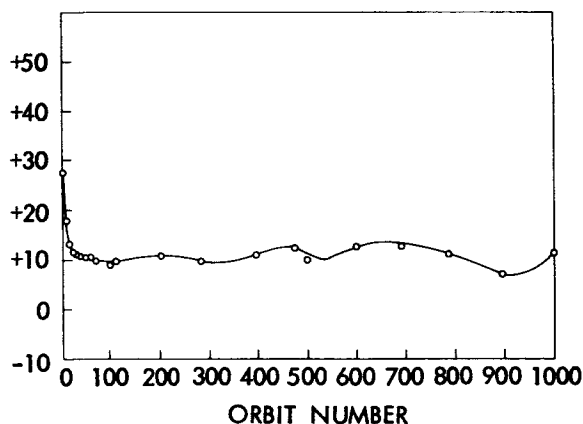


FIGURE 8-4.—X-ray spectrometer temperature.

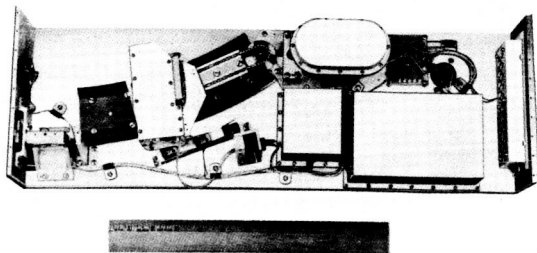


FIGURE 8-5.—Goddard solar-oriented x-ray spectrometer.

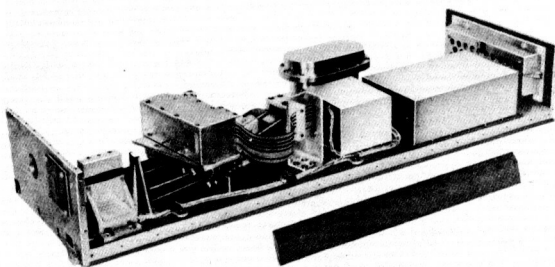


FIGURE 8-6.—Goddard solar-oriented x-ray spectrometer.

the thermistor detector amplifier was fed to the upper housekeeping commutator as one of 52 data points monitored by the spacecraft housekeeping commutators. Figures 8-5 and 8-6 show various views of the spectrometer.

Performance

The spectrometer aboard the spacecraft performed well, yielding information on the solar spectrum and variations in intensity of the solar lines as the solar activity changed. The region 10 to 60 Å was not useful due to scattered light. Solar radiation in the wavelengths covered by the spectrometer was primarily responsible for the formation of the F layers in the ionosphere. Thus, variations in the solar output should be correlated with changes in the F layer which may lead to a better understanding of the formulation and structure of the ionosphere.

Prior to the failure of the tape recorders, about 6000 spectra of the sun in the 10 to 400 Å region were collected. During this time over 30 flares of varying magnitudes occurred, the

highest magnitude being a Class 3 on March 22, 1962. It is believed that these data will

1. Aid in identifying the spectral lines.
2. Allow limits to be placed on the continua in this wavelength region for a quiet sun as well as during solar activity.
3. Allow quiet sun line intensity measurements to be made with some certainty. (From preliminary measurements it can be seen that a rocket-borne instrument would have encountered enhancement several hours after a visual flare was over.)
4. Determine line intensity enhancement, if any, before, during, and following visual flare activity.
5. Determine enhancement, if any, associated with other solar activity, i.e., plage areas and spot groups.
6. Determine short-time fluctuations in line intensities not associated with other easily observable phenomena.

Analysis of these results should throw light on energy transport mechanisms in the corona and the relaxation time of the corona, and form the basis for a more complete model of the chromosphere.

After the failure of the tape recorders, data were recorded only during those periods of time when the spacecraft was within range of a receiving station. The spectrometer continued to function during these periods. A preliminary report of solar flare observations has been prepared by W. E. Behring, W. N. Neupert and J. C. Lindsay of the Goddard Space Flight Center. This report, X-614-62-29 is entitled *Preliminary Solar Flare Observation with a Soft X-Ray Spectrometer on the Orbiting Solar Observatory*.

Alternate Pointed Instrument

The alternate pointed instrument package (Figure 8-7) contained the following listed experiments:

Goddard Gamma Ray Monitor

This experiment utilized a NaI (Tl) scintillator backed by a photomultiplier tube for the detection of the 0.510 Mev electron-positron annihilation line to study any possible temporal variations at this line.

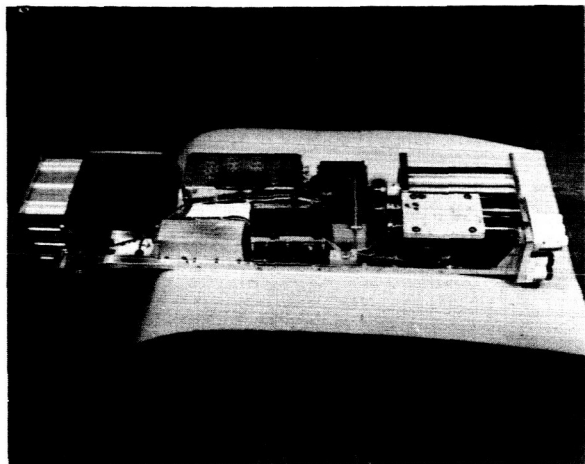


FIGURE 8-7.—Goddard alternate pointed-instrument package.

Pulse signals from the photomultiplier were fed into a 16-channel differential pulse-height analyzer which was time shared by this photomultiplier and two others in the GSFC wheel experiment. From the analyzer, the signals were coded and fed into a telemetry subcarrier oscillator.

The only difference between this experiment and the gamma-ray experiment in the GSFC wheel experiment package was that it was pointed at the sun continuously so that the energy analysis would be improved. This was accomplished by adjusting the gain of the system so that the 0.1 to 0.7 Mev region would be analyzed by the pulse-height analyzer giving approximately 50 keV resolution.

Goddard X-Ray Monitor (20–100 Kev)

A detector exactly like the scintillation detector used for the gamma-ray experiments, except for the use of a thinner crystal, was used to monitor the 20–100 keV x-ray emissions believed to be associated with Type III radio events.

It will be of interest to compare the intensity variation of the transient x-ray phenomenon with that obtained from radio observations.

Goddard X-Ray Monitor (1–8 Å)

Monitoring the solar flux in the 1 to 8 Å x-ray region was accomplished with two Be

(Xe) ion chambers, collimated to reduce the effect of scattered light.

The monitoring of the x-ray region substantially supplemented the information obtained from the x-ray spectrometer, which covered the 10 to 400 Å region, particularly during a solar flare when the x-ray emission was enhanced resulting in a hardening of the spectrum as well as an increase in intensity.

Goddard Dust Particle Experiment

The interplanetary dust particle experiment was designed to measure the influx rate, momentum, and kinetic energies of impacting dust particles. A photomultiplier tube, coated with aluminum approximately 2 microns thick, was used to measure the luminous energy emitted from the cloud of plasma formed when a dust particle impacts onto the aluminum-covered tube. A microphone attached to the side of the tube measured the mechanical impulse delivered by an impacting dust particle. The ranges of sensitivity of the two detectors were overlapped so that coincidence measurements were possible for a large fraction of the dust particles. A collimating tube was used to shield the photomultiplier tube from earth-reflected sunlight and to limit the view of the detector to a cone of approximately 30° half angle. The output signal from this experiment was fed to the 294 cps subcarrier oscillator channel.

Photoelectric Error Sensor Stability Experiment

Ball Brothers Research Corporation supplied an experiment designed to monitor the effects of the launch and the space environment on the photoelectric sensors used as error detectors in the spacecraft servo positioning control system. The experiment comprised six error sensors mounted in a block which was attached to the front surface of the alternate pointed instrument. A detector block temperature readout was also provided. The output data were sampled by a 20-data-point commutator operating at a one-sample-per-second rate. The commutator channels were shared by the Goddard alternate pointed instrument experiments and the BBRC experiment. Table 8-1 lists the channel assignments.

Table 8-1
Alternate Pointed Experiment Data Channel
Assignments

Channel No.	Function
1	Experimental target eye, no filter or lens
2 and 3	GSFC voltage readouts
4	5-volt calibration
5	2.5-volt calibration
6 and 7	0-volt calibration
8 and 9	20 kev to 100 kev x-ray monitoring experiment
10 and 11	1 to 4A x-ray monitoring experiment
12	15-volt regulated supply
13	Experiment temperature
14	18-volt B+ monitor
15	Experimental target eye No. 1
16	Experimental target eye No. 2
17 and 18	Dust particle experiment
19	Experimental fine eye pair (FE-3)
20	Experimental coarse eye (CE-3)

Of interest were engineering data concerning cell, lens, and filter-glass damage, mechanical shifts in position, and spectral response changes. The detectors selected and the intended uses for each are as follows:

Bare Cell Detector.—This detector was a fine error sensor with the lens and filter removed, exposing the unprotected silicon cell to direct solar irradiation. This detector was intended as a control so that differences in cell changes and filter and lens changes could be assessed.

Coarse Pointing Error Detector.—This detector provided a cell protected by a filter glass. The detector was identical to the servo coarse error detectors and was intended to provide data on the long-term stability of the coarse detectors.

Fine Pointing Error Detectors.—These two detectors provided cells protected by a lens and a filter glass. These detectors were of the same type used for fine servo error detectors. The detectors, which were not spectrally matched, gave information on the immediate differential output changes to be expected by being above the atmosphere, on the long-term differential changes between two detectors, and on the long-term absolute output degradation to be ex-

pected so that servo error detector gain changes could be assessed.

Fine Pointing Detector Pair.—This aligned and positioned detector pair provided data on mechanical shifts of position of the pointed instruments during launch by comparison with the readout eye pair. It also backed up the prime positional readout detector pair.

The amplifiers used to amplify the individual detector currents to the input voltage level required for the telemetry subcarrier oscillator were zero-suppression amplifiers. This was done to increase the accuracy of the telemetry channel readout. Full-scale output represented about plus or minus 12 percent of the nominal output. The amplifiers were calibrated with a $\frac{1}{2}$ percent current meter.

The amplifier (Figure 8-8) was adjusted to give 2.5v out with predicted eye current by adjusting R2 (zero-suppression resistor) and R4 (gain resistor).

The principle of circuit operation is as follows: Q1 presents the eye cell with a low input impedance to assure that the cell is being operated in the current mode. The collector of Q1 is a very good current source for R2. Thus, α_1 times the eye detector current flows out of the collector regardless of the collector circuit.

When the eye detector current is such that the voltage across R2 is high enough, the emitter of Q2 becomes a low impedance compared to R2 causing the majority of Q1 collector current will flow into Q2; α_2 times this current will flow through R4. The voltage across R4 is fed to telemetry by emitter follower Q4. Diode CR3 is the compensation for changes in the V_{BE} of Q4.

The voltage at which the current from the collector of Q1 is accepted by the emitter of Q2 is determined by the emitter voltage of Q3, which is in turn determined by the voltage divider R8, R6, R7. The V_{BE} of Q2 is nearly compensated for by Q3 V_{BE} and trimmed by R6.

The detector block temperature probe, using a thermistor detector, was similar to those used elsewhere on the spacecraft. The fine detector position reading pair required the use of a special low-level detector amplifier. The opera-

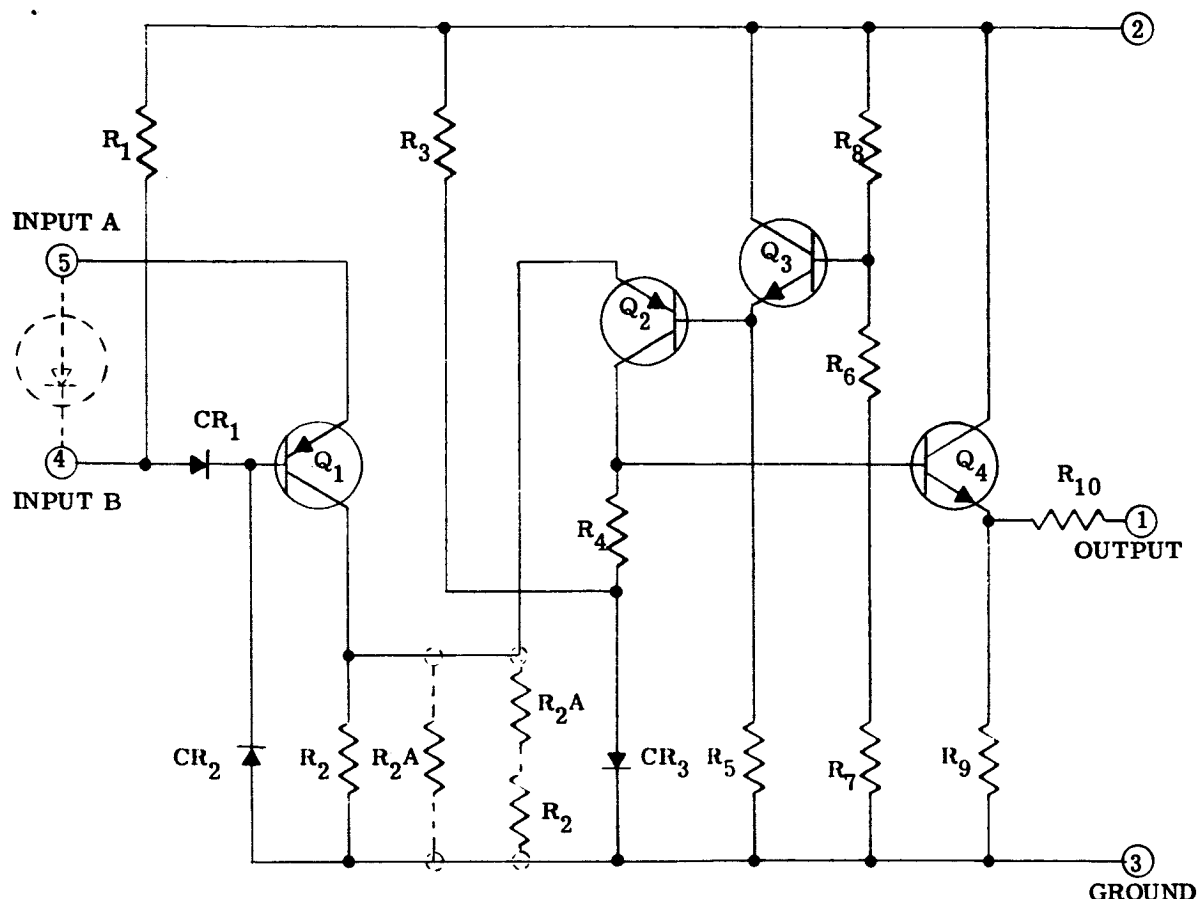


FIGURE 8-8.—Zero-suppression detection amplifier.

tion of this device (see Figure 8-9) is the same as the readout amplifiers previously described in Chapter 4.

The sensor monitoring experiment required a voltage shifter designed to accept an input signal ranging from 0 to 5 volts and to supply an output ranging from about 4 to 9 volts. The requirement for this unit arose from the fact that it was desirable to use an existing subcarrier oscillator which required a 4- to 9-volt input.

Figure 8-10 shows the schematic diagram of this circuit. The requirements on the voltage shifter were that it present the proper input and output impedance to the signal source and the subcarrier oscillator, and that it have good linearity and temperature stability. Voltage shifting is accomplished by means of the Zener diode D1. This diode adds constant voltage

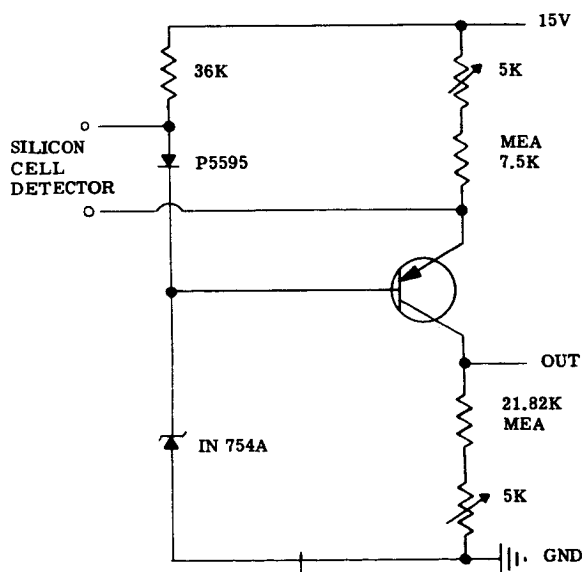


FIGURE 8-9.—Low-level detector amplifier.

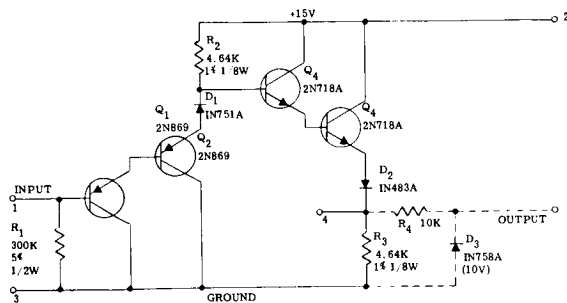


FIGURE 8-10.—Error detector experiment, voltage shifter.

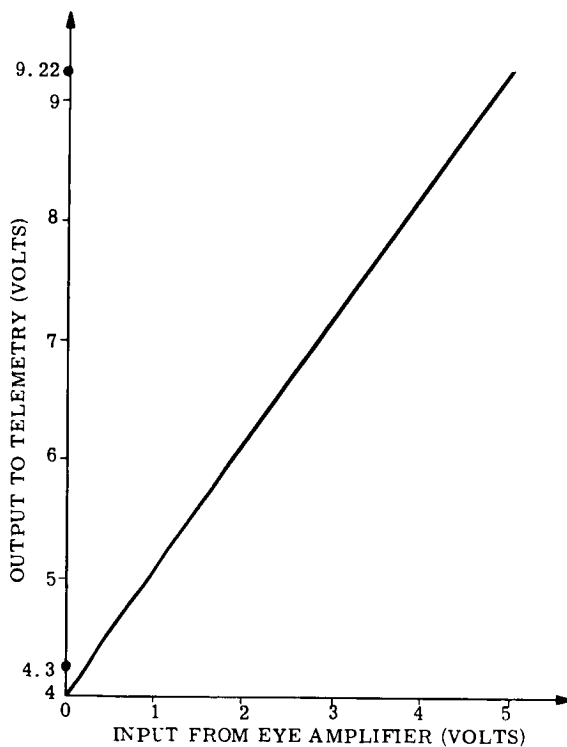


FIGURE 8-11.—Voltage shifter transfer curve.

to the input signal. Transistors Q1 and Q2 are used to present the proper impedance to the signal source. Transistors Q3 and Q4 present the proper impedance to the subcarrier oscillator. The output of the voltage shifter appears across R3. Figure 8-11 shows the transfer characteristics of a typical unit. Typically, the level shifter has a linearity of about plus or minus ½ percent and a temperature stability of about plus or minus 1 percent over the range of temperature, -20° to $+50^{\circ}\text{C}$. A limiting

network was added to the output of the level shifter to limit the maximum output to 10 volts. This was done to make it impossible to overdrive the subcarrier oscillator.

Performance

Information on the performance of the four GSFC solar-oriented experiments is not included in this chapter, but may be obtained by referring to the bibliography included in this publication.

The effects of the space environment upon the Ball Brothers photoelectric Error Sensor experiment were:

Detector Type	Degradation (percent per month)
Open cell	3.6
Fine pointing	1.8
Fine pointing	1.2
Coarse pointing	0

The degradation figures were derived over the first 1,000 orbits and were, for this period of time, linear functions of time. It was assumed that the data could be extrapolated to six months with some confidence if the electron flux problem were ignored.

The open-cell detector exposed the silicon cell directly to solar irradiation. The degradation was due to direct ultraviolet irradiation damage.

The fine pointing detectors with lenses and narrow transmission band filters were the type used for the servo positioning control error detectors. The two detectors were not spectrally matched. This is the reason for the slightly different degradation rates. The servo gain decrease in six months (nine percent average) due to eye output reduction caused a positioning error of the same percentage applied to the initial error; that is, with a 1.0 arc-minute initial control accuracy, the error after six months was increased by 0.1 minute of arc. The variation in positional accuracy caused by the different degradation rates was negligible.

The coarse pointing detector was identical to the coarse detectors used for the positioning control system. No degradation was measured. The detector output varied, in agreement with the theoretical, as the variation in the solar constant. This information indicated that the

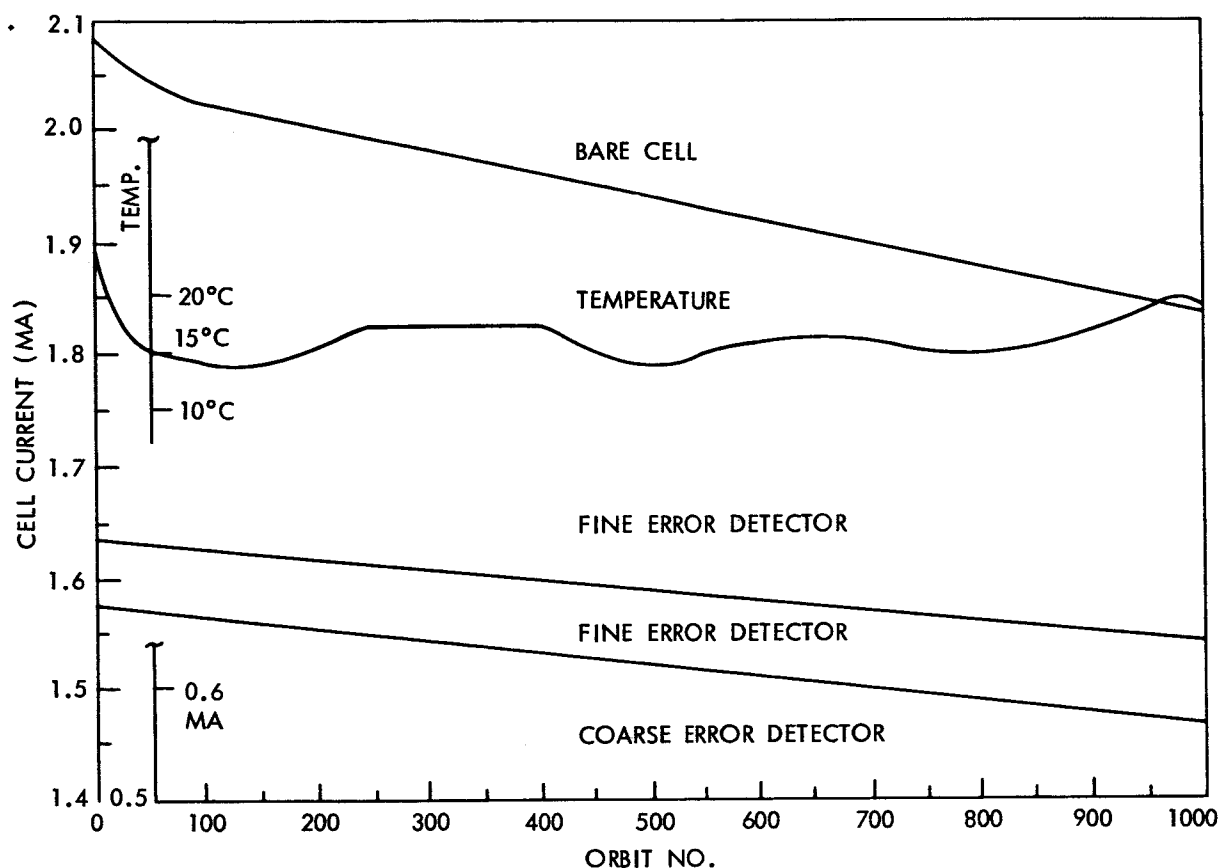


FIGURE 8-12.—Error detector stability experiment—detector output variations.

degradation measured for the fine pointing detectors was due entirely to effects on the lens glass (probably ultraviolet radiation-induced color center formation causing some attenuation) and that the degradation of the cells themselves was negligible when protected by the filter glass.

Data concerning the effects of high-energy electron flux on the detectors would be of considerable interest, but unfortunately, the detector outputs were reduced to the lowest level readable by about orbit 2000. This was about the time the 9 July 1962 high-altitude nuclear device was exploded. Since the coarse detector eye did not continue to give on-scale readings after that time, as was anticipated since the solar constant was on the increasing half-cycle, it was assumed that some radiation change occurred. This change needed to be only about two percent to produce the effects measured, thus no real limits could be estimated.

The azimuth position readout pair of detectors performed well. After the upper commutator failure about orbit 1900, this detector pair became the only readout available to allow determination of azimuth pointing accuracy. A relative shift in azimuth pointing direction between the pointed experiments (about one arc-minute) occurred during launch. After 3500 orbits, the experimental pair was reading azimuth positioning error changes from the initial position of less than one minute of arc.

Two curves of interest are presented concerning the experiment. The curves of Figure 8-12 show the combined effects of degradation and solar constant variation over the first 1000 orbits. The solar constant change was a decrease of 3.2 percent over this period. The coarse detector, as noted previously, followed this change very closely.

The curve of Figure 8-13 shows the coarse detector output variation over one orbit. The

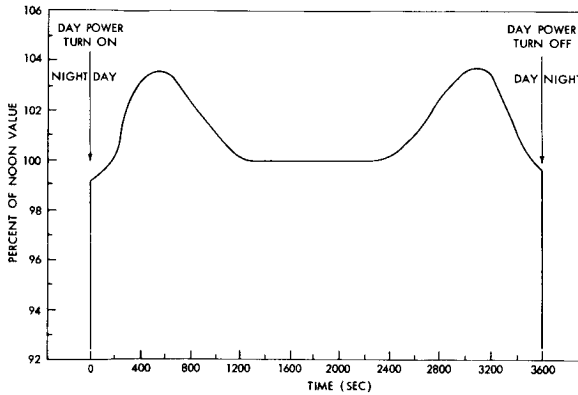


FIGURE 8-13.—Error detector stability experiment—coarse detector output.

peaks after dawn and before dusk were of variable amplitude and varied somewhat in shape and in time as the spacecraft-earth-sun orientation changed. The peaks are due to earth-light inputs to the coarse detector. The curve plotted is typical.

WHEEL EXPERIMENTS

There were nine wedge-shaped compartments in the wheel section of the spacecraft. Of these nine compartments, five were available for scientific instruments. The space allocated to the wheel experiments, except the Ames Research Center experiment, is shown in Figure 8-14. The OSO I carried six different experiments in the wheel section. The location of these experimental packages and a brief description of each are shown in Figure 8-15. The total weight of the wheel experiments in OSO I was 113 pounds.

Goddard Wheel Experiments

The Goddard Space Flight Center wheel experiment package for the OSO I is shown in Figure 8-16 and comprised the following experiments:

Solar Radiation Experiment

This experiment utilized a special photodiode with a filter which restricted its spectral response to the region between 3800 and 4800 Å at the half-intensity point, very similar to a band widely used in stellar photometry.

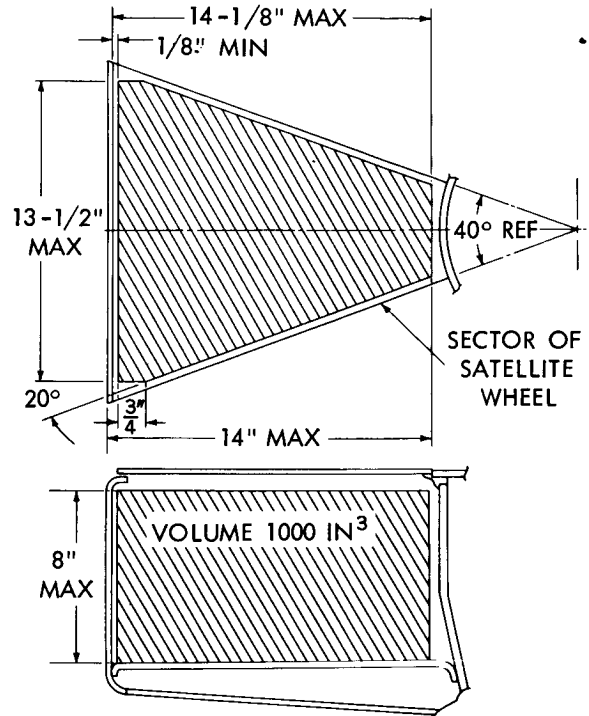


FIGURE 8-14.—Compartment space allocated to wheel experiments.

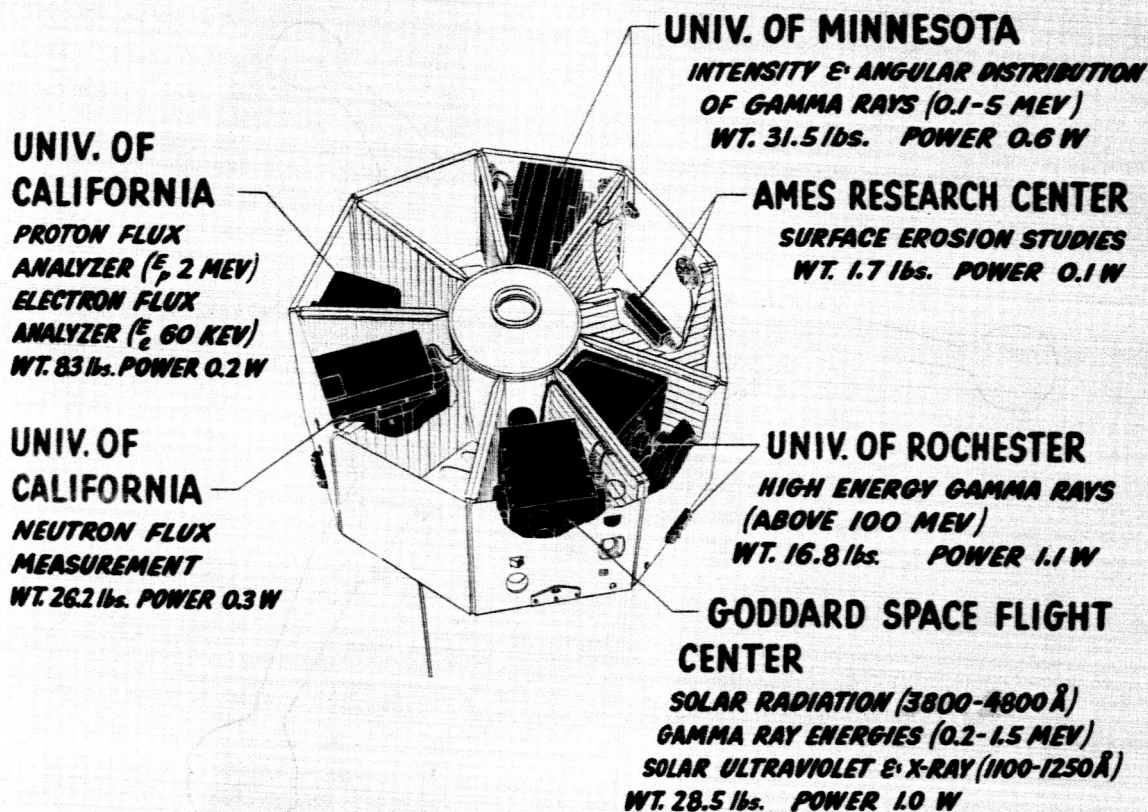
This wavelength region, encompassing the frequencies of maximum intensity of flux radiated from the sun, is important in determining the total energy balance of the sun. With this instrument, variations of the solar flux as small as 0.1 percent were detected. This allowed very rapid variations to be studied, as well as the anticipated variations within the solar cycle.

Solar Ultraviolet Experiment

Monitoring of the solar flux from 1100 to 1250 Å was accomplished with a LiF-CS₂ ion chamber suitably stopped down to retard deterioration. Considered of primary importance, this wavelength region included the Lyman-alpha chromospheric emission line which is strongly associated with solar activity. Intensity variations were measured to allow correlation to be made with active features on the solar surface.

Solar Gamma-Ray Experiment

This gamma-ray experiment was designed to detect gamma rays from the sun and analyze



WHEEL EXPERIMENTS

FIGURE 8-15.—Wheel experiments.

their energy spectra from 0.2 to 1.5 Mev. The experiment utilized for detection two NaI (Tl) scintillators, backed by photomultipliers. One detector was shielded to achieve a penumbral acceptance angle of 20 degrees. The second detector, which was unshielded, was used in anticipation of background events induced by the presence of spacecraft mass and, in particular, by the shielding of the other detector. The unshielded detector had nearly isotropic response.

Pulse signals from the photomultipliers were fed into a 16-channel differential pulse-height analyzer which was time-shared by these two detectors and by one in the composite pointed experiment. From the analyzer, the signals were coded and fed to the spacecraft telemetry system. Of a particular interest in this experiment was the ability to detect the

0.5 Mev electron-positron annihilation line and to study any possible temporal variations. The detectors looking outward from the wheel spent a good fraction of the time scanning

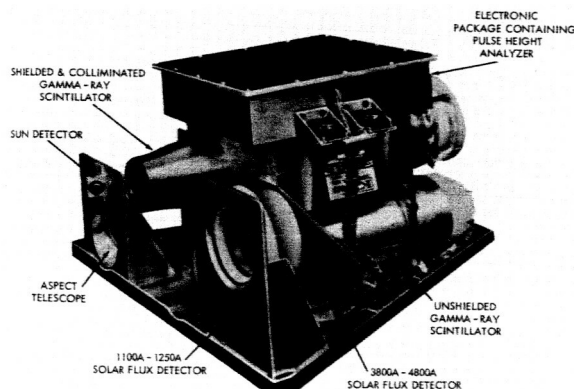


FIGURE 8-16.—Goddard Space Flight Center wheel experiment package.

celestial space and thus may be able to detect other moderately strong sources of low-energy gamma rays.

Earth Horizon Sensor

The Goddard wheel instrument contained an earth horizon sensor for use in obtaining data on the orientation of the spacecraft roll axis. The earth sensor was pointed in the same azimuth direction as the gamma-ray detectors, but its axis was positioned 15 degrees downward (away from the upper section) to prevent direct exposure to the sun. The field of view of the earth sensor was a cone with a four-degree included angle. When any portion of this field intercepted the sunlit portion of the earth, an output signal was generated. The earth sensor was not responsive to radiations from the dark portion of the earth, but intervening clouds on the sunlit side did not cause a change in the output signal amplitude.

These data sources were time-multiplexed into the 790 cps subcarrier oscillator telemetry channel during playback.

Synchronization pulses were inserted into the data format at frequent intervals to facilitate decommutation on the ground. The data was commutated in frame intervals consisting of 67 intervals per frame. Table 8-2 shows the interval assignment. Each interval was timed with a sun sensor output signal by the rotation period of the wheel portion of the spacecraft. For nominal spacecraft spin rates, the intervals were two seconds. The output format is shown in Figure 8-17.

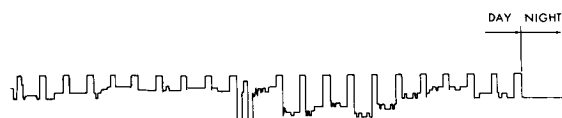


FIGURE 8-17.—Goddard wheel experiment data output format.

Performance

The earth sensor received signals from scattered light reflected from the cover during each rotation past the sun. This scattered light produced an extra pulse in the output signal but did not interfere with the normal function of the earth sensor. The solar radiation diode

Table 8-2

Goddard Space Flight Center Wheel Experiment Frame Intervals

Interval No.	Assignment
1 -----	Analog 1 (earth-horizon aspect)
2 -----	Analog 2 (ion chamber)
3 -----	Analog 3 (blue diode)
4 -----	Proportional Counter (PC)
5 -----	Collimated Photomultiplier (CPM)
6 -----	Non-collimated Photomultiplier (NCPM)
7 -----	Sync No. 1
8 -----	PC
9 -----	CPM
10 -----	NCPM
11 -----	Sync No. 2
12 -----	PC
13 -----	CPM
14 -----	NCPM
↓ -----	↓
↓ -----	↓
↓ -----	↓
63 -----	Sync No. 15
64 -----	PC
65 -----	CPM
66 -----	NCPM
67 -----	Sync No. 16

was initially saturated. The Lyman-alpha sensor was near midscale initially and later began to show a very slow degradation with time indicating that the lithium window was darkening from integrated radiation effects. The gamma-ray experiments functioned properly, but it will be some time before statistically plotted data becomes available due to the low flux levels obtained.

University of Minnesota Gamma-Ray Experiment

This experiment was designed to provide preliminary measurements of the intensity and directional properties of low-energy gamma rays in space. These photons have energies of about 1 Mev and are similar to light rays or x-rays in that they travel with the same speed and are not deflected by electric and magnetic fields. Thus, unlike true cosmic rays, but like starlight, the gamma rays arrive from the same direction in space from which they originated. They

cannot, however, penetrate the earth's atmosphere.

The gamma rays observed had their origin in the interaction of atomic nuclei in or near stars. Since the sun itself is a giant atomic furnace sustained by the energy of nuclear reactions, gamma rays leaking from the solar surface should have been detected. From the intensity of these rays, more could be learned about the thermonuclear process responsible for the characteristics of the radiative output of the sun.

At times, when large sunspots appear on the solar surface, an explosive eruption known as a flare occurs. These flares produce disturbances which disrupt radio communications on the earth, cause the northern lights or aurora borealis, greatly disturb the earth's magnetic field, and probably cause changes in the weather. Bursts of gamma rays from flares have been observed from balloons on rare occasions. The instruments on the spacecraft continuously looked for solar gamma-ray bursts in order to better understand the solar flare phenomena.

The Milky Way, which is our edge-on view of the galaxy containing our sun, may be an interesting source of gamma-ray activity. An explanation correlating an expanding universe, as observed by astronomers, and a steady-state universe is the *continuous creation of matter* theory proposed by the British Physicist, Fred Hoyle. The theory implies an annihilation of matter, and this process produces gamma rays. The intensity of gamma rays from the Milky Way may then tell us something about the creation of matter in our galaxy. Among other objects in the sky which may emit gamma rays is the Crab Nebula in the constellation Taurus. This is the remnant of a supernova or sudden explosion of a star seen by the Chinese in the year 1054. From the number and kind of gamma rays, some information about the cause of the explosion may be obtained. In the constellation Cassiopeia there is an object, or an area, discovered by radio astronomers, known as the *most powerful radio transmitter in the universe*. One explanation of the amount of radio waves given off is that two galaxies, one composed of ordinary matter

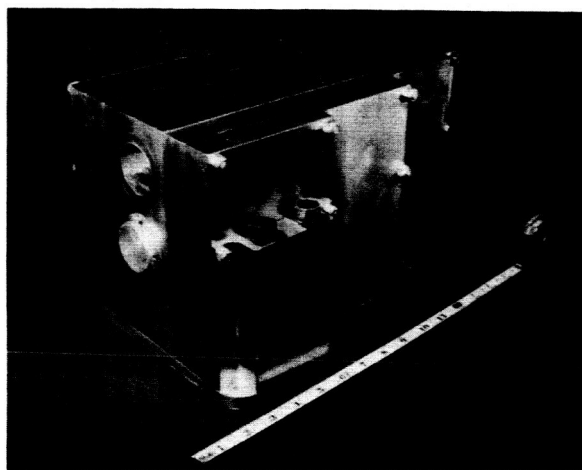


FIGURE 8-18.—University of Minnesota gamma ray experiment.

and one of *antimatter*, are in collision. If this were indeed true, a large number of gamma rays would be emitted in the process of the annihilation of matter.

These measurements are all complicated by the high *background* due to cosmic rays and the low flux density to be measured. The flux anticipated was less than one gamma ray crossing a one centimeter square area each second. A special detector arrangement, known as a Compton coincidence telescope, and some special circuits were developed at the University of Minnesota to perform the measurement. The telescope was mounted in the wheel of the spacecraft and scanned the sky as the wheel rotated, continuously recording and storing information about gamma rays. The instrument weighed 30 pounds and had a volume of less than one-half cubic foot. The instrument is shown in Figure 8-18. The experiment used nearly 400 transistors and operated continuously on a power of one-half watt. Data from the detectors were entered into a temporary binary storage device and read out, together with identification and spacecraft orientation information, to a spacecraft telemetry channel at a one bit-per-second rate. The output format for this experiment is shown in Figure 8-19. A day-night transition is illustrated in the format.

The University of Minnesota gamma-ray experiment monitored the intensity and direc-



FIGURE 8-19.—University of Minnesota experiment output format.

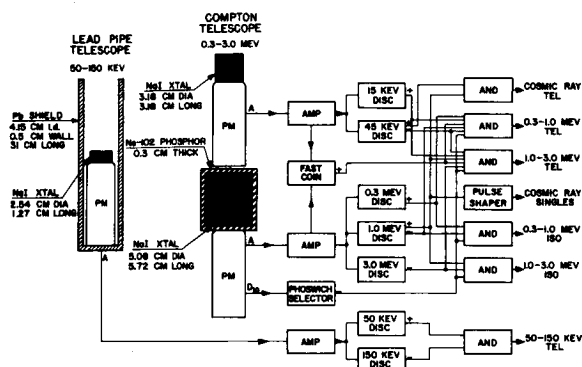


FIGURE 8-20.—University of Minnesota gamma ray experiment block diagram.

tion properties of gamma rays between 50 kev and 3 Mev energy and observed time variations of the solar flux in this region. Advantage was taken of the fact that the wheel spin axis was maintained normal, within a few degrees, to a plane containing the solar vector. The telescope, pointing radially outward from the wheel, then swept by the sun, across the sky, and probably across the earth with every two-second rotation period of the wheel.

The 50 to 150 kev telescope, shown in the block diagram of Figure 8-20, consisted of a NaI (Tl) crystal scintillation counter in a lead pipe shield which looked outward through the wall of the spacecraft. A directional detector, operating in the 0.3 to 1.0 Mev energy regions, was obtained by using two scintillation counters connected as a Compton coincidence telescope. Photons, after being scattered in the first crystal, were absorbed in the second scintillation detector. The directional properties were obtained by selecting only photons scattered forward in the first crystal. The absorbing NaI (Tl) crystal was surrounded by a plastic phosphor fence which rejected events having proper energy but caused by edge effects of particles in the crystal. Photon counters in the 0.3 to 1.0 and 1.0 to 3.0 Mev energy region

with isotropic sensitivity and high counting rate were obtained from the absorption detector. A cosmic-ray single counter was formed by counting all events in the absorption detector, and a cosmic-ray telescope was formed from coincidences between the absorbing and scattering detectors.

During the day, the direction of the telescopes with respect to the sun was obtained from 16 sun detectors placed every 22.5 degrees around the spacecraft wheel rim. A gate of 22.5 degrees duration (125 milliseconds at 30 rpm) was generated every time the telescopes were pointed at the sun and during one other pre-programmed 22.5-degree sector interval. These gates allowed counts registered in the telescope to pass into the storing and accumulating binary scalars. The sector gate was programmed so that after every night-day sequence, the sector was advanced one 22.5-degree interval. Hence, after every 16 orbital passes of the satellite, the telescopes would have been gated at each 22.5-degree interval around the wheel for one orbit. The sun interval was always gated ON during every orbital pass and every daytime wheel revolution.

The six 9-binary storage registers accepted events from the detectors in the following manner:

1. From the 50 to 150 kev lead pipe telescope during the sun gate interval.
2. From the 50 to 150 kev lead pipe telescope during the sector gate interval.
3. Scaled-down counts from the 0.3 to 1.0 Mev isotropic counter continuously.
4. From the 0.3 to 1.0 Mev Compton telescope during the sector gate interval.
5. Scaled-down counts from the 1.0 to 3.0 Mev isotropic counter continuously.
6. From the 1.0 to 3.0 Mev Compton telescope during the sector gate interval.

Events from the detectors were continuously accumulated in the storage register during the daytime portion of the orbit. The storage registers were read nondestructively in sequence into a shifting register which advanced at a two-bit-per-wheel-rotation rate (1 bit-per-sec at 30 rpm). Six words of 13 nonreturn-to-zero bits were supplied to the telemetry subcarrier oscillator. The first four, or "1D," bits of each

word were of five-volt amplitude and identified the beginning of the six-word sequence, the word number, the position of the sector gate, and the position of the earth horizon. The last nine bits of each word were of three-volt amplitude and corresponded to the reading of the storage register at the time of the start. Numbers read from the storage register continually changed due to the continued accumulation of counts from the detectors.

During the night, the readout of the storage and operation of the shifting register were inhibited, and the cosmic-ray single counter and telescope data were recorded. The outputs of detectors were suitably scaled and the resultant *square waves* were added together and fed into the telemetry subcarrier oscillator.

Performance

Inspection of the early telemetered passes at Cape Kennedy indicated that the instrument was performing properly. Later passes revealed that occasionally incorrect digital words were received. This effect, apparently due to an electrical transient resetting the shift-register logic, was particularly troublesome on passes 50 to 150 and had vanished by pass 400. No actual loss of information resulted, but automatic data reduction became more difficult. No detrimental effect was observed due to the lower than anticipated operating temperature of the satellite, and no evidence of a gain change in the scintillation counters was noted. The signal-to-noise ratio on most passes was quite adequate for automatic data reduction.

The background gamma-ray rates in orbit were about a factor of three higher than those predicted by a reasonable extrapolation of balloon results at Minneapolis. Most of these

gamma rays presumably were produced by cosmic rays. That most of the counting rate was not caused by local radioactivity in the satellite itself was inferred from a preflight calibration. The calibration, performed a few days before the launch with the entire payload and vehicle configuration in the gantry, gave lower background rates than in any previous situation. These were about a factor of ten below those observed in orbit.

Of the first 1000 passes, about 50 passes had been completely reduced and another 50 were in process at the time of this writing. Since the Compton telescopes had the possibility of indicating the presence of an extraterrestrial gamma-ray flux in the most unambiguous manner, the data from 48 passes, numbered generally in the 400's, are presented in Table 8-3.

Events occurring when the spacecraft was in the trapped radiation areas, as well as noisy records, were eliminated. Events were accumulated when the earth detector was illuminated, when it was off, and when the telescopes looked at the sun. Since the acceptance angle of the earth telescope was about the same as the Compton telescope at 1.0 Mev, this formed a simple means for distinguishing earth albedo gamma rays from those that were incoming. The rates of the isotropic counters, averaged over the same time intervals as the telescopes, are given. The near identity of these rates indicated the absence of any large systematic effect due to a nonrandom variation of the isotropic flux during the intervals in which counts were tallied: for example, from radiation belt background. The results would indicate an extraterrestrial flux greater than the earth's albedo. In order to actually determine the flux and to

Table 8-3
Instrument Rates—48 Passes

	0.3-1.0 Mev		1.0-3.0 Mev	
	Comp. Tel.	Isotropic	Comp. Tel.	Isotropic
Earth.....	$1.99 \pm 0.3 \times 10^{-2}$	21.0 c/sec.....	$1.40 \pm 0.24 \times 10^{-2}$	8.7 c/sec
No earth.....	$3.10 \pm 0.3 \times 10^{-2}$	20.3 c/sec.....	$2.36 \pm 0.26 \times 10^{-2}$	9.6 c/sec
Sun.....	$2.05 \pm 0.25 \times 10^{-2}$	18.6 c/sec.....	$3.03 \pm 0.03 \times 10^{-2}$	8.4 c/sec

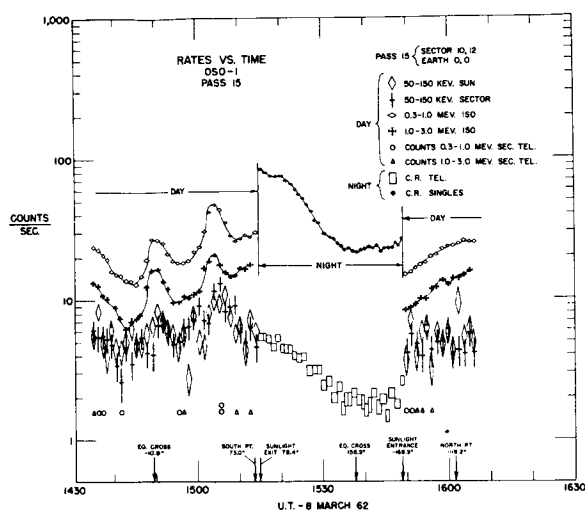


FIGURE 8-21.—University of Minnesota experiment data for a typical pass.

gain confidence in these results, an account was taken of possible background in the Compton telescopes, their angular response, and the incident photon spectrum. At the time of this writing the data were insufficient to determine with any certainty the anisotropies in the counting rate of the Compton telescope looking away from the earth.

The counting rates as a function of time for an early pass are shown in Figure 8-21. Day and night portions of the pass are clearly indicated as well as the longitude of certain significant orbital points. The recording of Pass 15 started after the readout of the previous pass by the Fort Myers station. The spacecraft crossed the equator over the Atlantic, reached its southernmost point over the southern tip of Africa, again crossed the equator going north over the mid-Pacific, and reached its northernmost point over the United States just before readout by the Fort Myers station.

The rates of the isotropic counter in the 0.3 to 1.0 and 1.0 to 3.0 Mev regions are indicated in the two uppermost curves of the figure. The second daytime portion of the pass clearly indicates the variation with latitude of the background counting rate due to cosmic rays. Since the omnidirectional cross-section of the counter is 33 cm², the rates correspond to a flux of about 0.45 and 0.24 counts per cm² sec in the 0.3 to 1.0 Mev and 1.0 to 3.0 Mev regions,

respectively, at the equator. A variation of about 2 to 1 in the rates from the equator to geomagnetic 40° is observed.

The first daytime portion, in addition to the general latitude variation, shows effects due to the trapped radiation which were observed on most passes over the South Atlantic and Africa. These presumably were due to the displacement of the earth's magnetic dipole or to the Cape-town anomaly that caused a detectable fraction of the trapped flux to have turning points less than 60 km above the earth's surface. Indeed, on many passes, the rates were so high in these areas that the accumulating registers overflowed between readouts and gave ambiguous results.

The rates of the lead-pipe telescope in the 50 to 150 keV energy region are also shown. The sun gate counted during the 22.5° portion of the satellite wheel rotation, and the sector gate was opened during sectors 10 and 12 during the first and second daytime portions of the pass, respectively. This corresponds to looking 225° and 270° with respect to the solar radius vector. Although the statistical accuracy of the rates was not very high, no significant difference was noted between looking at the sun and looking at wide angles to it.

The counts from the Compton telescope are indicated in an event-by-event manner. Apparently, the earth did not fall into view of the earth detector since the indicator read zero during the entire daytime portions of the pass.

The rates obtained during the night portion of the pass are also shown in the figure. The upper curve shows all events greater than 1.0 Mev in the isotropic counter, and the lower curve gives the total coincidence rate of the detectors forming the Compton telescope. Most events in the isotropic counter can be accounted for in terms of the charged cosmic-ray flux. The counter rate shows about a 4 to 1 variation with geomagnetic latitude to $\lambda = 40^\circ$. This should be compared with the ratio of gamma rays of about 2 to 1.

Conclusions

The following information will be forthcoming from the University of Minnesota experiment on the OSO I satellite:

1. A measurement of extraterrestrial gamma-

ray flux between 0.3 and 3.0 Mev, and an indication of its origin on the celestial sphere.

2. Upper limits on the solar flux in the 50 to 150 kev energy region during quiet time and certain disturbed times, and possibly the detection of solar x-ray bursts during flares.

3. Accurate determination of the cosmic-ray latitude effect and the cosmic-ray equator at 600 km.

4. The production of earth albedo gamma rays by cosmic rays, and hence an evaluation of the production function with primary energy.

5. A mapping of the effect of the displacement of the earth's dipole and the magnetic anomalies on the trapped radiation over a limited altitude, latitude, and intensity range.

6. Information relative to the background problem in gamma-ray astronomy.

University of Rochester High-Energy Gamma Ray Experiment

The primary purpose of the OSO I spacecraft system was to study the short wavelength portion of the electromagnetic spectrum emitted by the sun. This will result in a better understanding of the sun and its influence on the earth. Also, since the sun is a star, its study will greatly increase our knowledge about other stars.

The University of Rochester wheel experiment shown in Figure 8-22 detected and measured the intensity of the shortest wavelength portion of the spectrum—gamma rays with energy greater than 100 Mev. The sun was the primary source investigated. However, facilities were incorporated into the experiment to search most of the sky. The sensitive area of the detector was 16 cm², its angular resolution was about 20 degrees, and it responded only to the fluxes entering the sensitive cone in the inward radial direction of the spacecraft wheel section.

The importance of the energetic gamma rays can be realized if the nature of their origin is considered. Radiation in this energy region would result from the decay of π^0 meson, an elementary particle with a mass 270 times that of an electron, which decays into two gamma rays. The mean lifetime of the particle

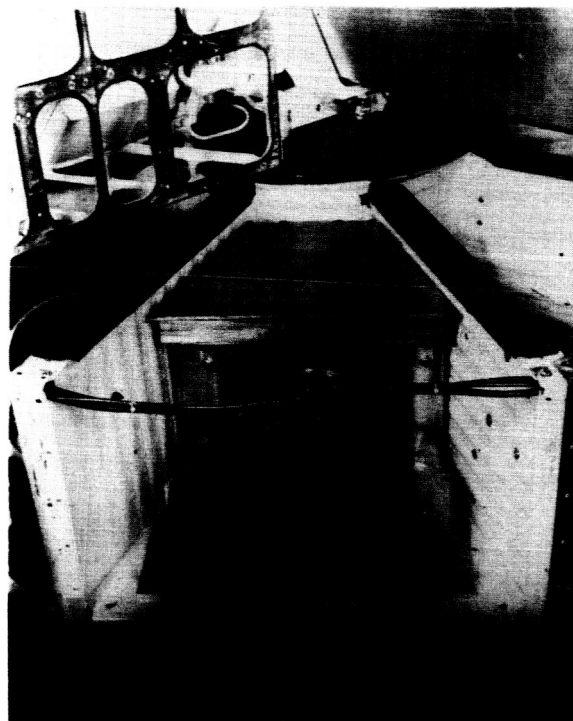


FIGURE 8-22.—University of Rochester wheel experiment.

is 2×10^{-16} seconds. These mesons can be produced either by high-energy proton-proton collisions or by the annihilation collision reaction of protons and antiprotons, i.e., negative-charged particles with the mass of the proton.

Sources which emit energetic gamma rays are usually anomalous stellar objects or occurrences, and the relatively simple nature of the production of energetic gamma rays leads to direct information about the source. Location of objects which contain high-energy protons, for example, could increase our knowledge of the origin of cosmic rays.

The most probable sources of gamma rays are those regions known to be strong sources of short-wave radio waves. Such radio emission is characteristic of synchrotron radiation from high-energy electrons. The mechanism which accelerated these electrons could also possibly accelerate protons to the high energies (several hundred Mev) necessary for gamma-ray production. Solar flares on the active sun, our galaxy, the supernova Crab nebula, anomalous

extragalactic sources, such as the galaxy M87 and the colliding galaxy system, and Cygnus α are all possible gamma-ray sources. Cygnus α may possibly be a system involving matter-antimatter collisions.

Enthusiastic desire to understand and explore these sources was almost lost when estimates of the fluxes arriving at the earth were considered. Information from the radio spectrum flux of the Crab nebula led to an expected flux of about 10^{-5} gamma rays per square centimeter per second. Proton-anti-proton annihilation resulting in production of π^0 mesons gave an estimated flux of gamma rays from Cygnus α of 10^{-7} per square centimeter per second. These fluxes can be compared to the limiting sensitivity of the 200-inch Palomar telescope to visible light, which is 6×10^{-3} quanta per square centimeter per second.

The study of this last frontier of the electromagnetic spectrum is important because of the unique and exciting knowledge it will provide about the structure of our sun and its effect on the earth. Knowledge of our galaxy, other galaxies in the universe, and insight into the origin of cosmic rays may be gathered. Some day a gamma-ray map of the sky may be plotted from data such as that derived from this experiment.

The principle of operation of the experiment is essentially as follows. The gamma ray is converted into an electron-positron pair in a lead sheet. The pair generates Cerenkov light in a plastic cylinder and is detected by a photomultiplier optically coupled to the plastic. As protection against primary relativistic charged particles, an anticoincidence scintillator is placed between the lead sheet and the Cerenkov cylinder. The rays so detected are counted for various sections of sky as the spacecraft wheel sweeps the circle.

The output data is presented to the telemetering system in a pulse duration-modulation form (PDM). The output pulse sequence contains the following kinds of data:

Solar gamma-ray counts—Channel 1

Nonsolar gamma-ray counts—Channels 2 through 16

Unique identification pulse—Channel 17 or calibration

Saturation count—Channel 18

Zero count—Channel 19

Five-pulse count—Channel 20

The output format for this experiment is shown in Figure 8-23.

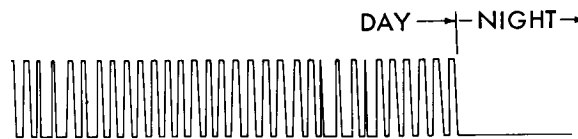


FIGURE 8-23.—University of Rochester gamma ray experiment output format.

Performance

From the data gathered, the detector system and logic functioned perfectly. Once every rotation of the spacecraft wheel, the detector was pointed toward the sun. In addition, 16 other sectors of the sky and earth were investigated for gamma-ray production. The system of photo diode sensors on the wheel perimeter used to identify the arrival direction of the gamma-ray flux also worked perfectly.

The data format was relatively simple so that reduction could proceed rapidly. The detector viewed the sun during several flares, and upper limits on the gamma-ray flux produced will be quoted. The galactic plane was located almost normal to the plane of the wheel, permitting the detector to view two sectors of the galaxy. The radio source Cygnus α was also in the acceptance cone of the detector.

University of California Experiments

Neutron Flux Experiment

The neutron experiment on the Orbiting Solar Observatory was the first substantial neutron experiment in space. Flying in a relatively low orbit, as OSO did, those neutrons counted were predominantly from the earth. These neutrons are thought to leak out of the atmosphere of the earth. They are thought to be produced in the atmosphere by cosmic rays striking oxygen and nitrogen nuclei. Calculations have been made as to how many of these leakage neutrons should be detected and what their energy spectrum should be.

• The previous experiments by the University of California on two Atlas rockets have given reasonable measurements of the neutron fluxes in space for short periods of time. The values in space for two short periods of time. The values obtained were different, but one flight was during a solar storm, so this difference was not surprising. The OSO spacecraft flight provided measurements of the neutron flux in space for an extended period of time. The experiment had to be able to measure the time variations of the flux, and hopefully, be able to study the neutron enhancement effects during a solar storm. It was intended to try to determine the latitude dependence of this enhancement.

Besides measuring the quiet day neutron flux and time variations of flux, the experiment looked for flux variations at sunset to see if an effect could be observed related to solar neutrons or x-rays.

Understanding the neutron flux was important in itself, but it was also important because the decay of these neutrons forms one of the important sources of the Van Allen radiation belt. Understanding the neutrons quantitatively will help understand the radiation belts quantitatively.

The experiment is shown in Figure 8-24. The unit weighed approximately 26 pounds. The power requirements of this experiment

were 20 milliamps at 18 volts, or about 360 milliwatts.

Proton-Electron Flux Experiment

The objective of this experiment was to determine the time and position variations of the fluxes of protons of 2.0 Mev energy and electrons of 60 kev energy in the region below the Van Allen belts. In particular, the experiment distinguished between electron ionization events and proton (and other heavy particles) ionization events. This experiment was chosen by GSFC because it was expected that the largest fraction of high-energy radiation in the plasma sphere and in outer space would be of these types. The proton-electron (P-E) detector, shown in Figure 8-25, was the first space application of a phenomenon that had been well known in nuclear physics for several years. It was based on the fact that in certain scintillators, protons and electrons produce fluorescent pulses of distinctly different decay times. This made it possible through use of time-comparing and discriminating circuits to employ a single scintillator on a single photo-

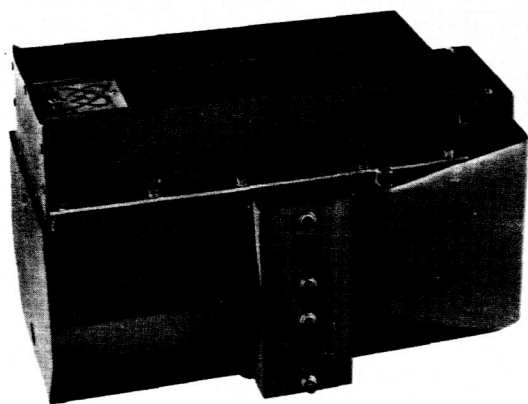


FIGURE 8-24.—University of California neutron flux experiment.

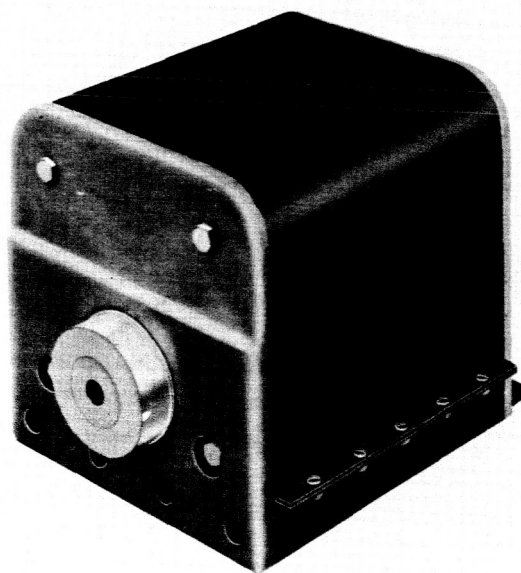


FIGURE 8-25.—University of California proton-electron detector.

multiplier to detect and separately count both protons and electrons.

The basic detector consisted of a 15 cm-diameter by 1.0 cm-high stilbene crystal on an RCA C7151 ruggedized photomultiplier tube. Side and front shielding provided collimation of the crystal and excluded protons of $E_p=100$ Mev. A quartz light pipe provided similar shielding from the rear. The electronic components were mounted on foam-covered circuit boards which compactly surrounded the photomultiplier housing. The unit weighed about 9 pounds. Power requirements were 14.5 ma at 18 volts, or about 265 mw.

To measure the neutron flux, two BF_3 proportional counters, one B^{10}F_3 and one B^{11}F_3 were used. The neutrons were counted by the reaction, $\text{B}^{10} + \text{N} \rightarrow \text{Li} + \alpha$. The B^{11}F_3 counter was used to measure the background counting rate. The background due to (p,n) events in the vehicle and detector was estimated to be low, but it also was measured by exposing the detector to a high-energy proton beam. The BF_3 counters were covered with about $1\frac{1}{2}$ inches of epon moderator to make the neutron count rate larger. This detector had a nearly constant efficiency from one kev to several Mev.

Two GM counters to study charged particles were also used. This was not a primary measurement. It was intended to use the GM counter count rates to determine whether increases in the neutron count rate were in time coincidence with large charged particle count rates. This determined whether protons producing neutrons in the detector were responsible for the neutron count rate, or if neutrons were themselves incident on the detector. One GM counter had a sweeping magnet so that it would count mostly protons. The second GM counter had no magnet so it would count protons and electrons.

The output signal from the two experiment signals was combined into one and was the input to the 572-cps telemetry subcarrier oscillator, which during playback became Channel

12 (the 10.5 kc channel). A sample format of the putput signal is shown in Figure 8-26. The output signal is made up of the following items:

G(W)—Geiger tube with magnet (electrons)
 G''— $\text{G}''\text{F}_3$ neutron detector
 G(o)—Geiger tube without magnet
 B^{10} — B^{10}F_3 neutron detector
 P+E—Proton plus electron scintillator detector
 PI—Proton identifier
 (P+E)/64
 B^{10} — B^{10}F_3 neutron detector

The output signal commutator required 51.2 seconds to complete a frame. Every sixteenth frame was an automatic calibration.

Performance

The neutron experiment worked well except for improper count presentation from the GM counters. These counters were to tell if some counts in the BF_3 counters were caused by neutrons produced in the spacecraft by a large charged-particle flux, rather than by incident neutrons. This function was also performed by the proton-electron package, making the GM counters less necessary. The calibration system in the neutron package indicated that the threshold sensitivity of all channels following the detectors had changed by less than 20 percent (which was insignificant with good plateaus).

The proton-electron experiment operated successfully. The calibration system on board indicated that the threshold sensitivity of the proton-plus-electron channel following the photomultiplier had changed by less than 4 percent. The generally high electron and proton flux region over the South Atlantic was mapped for several orbits, but a comprehensive data accumulation has yet to be achieved. Some indication of time dependence of the proton fluxes observed over North America was also found.

Ames Research Center Emissivity Stability Experiment

During their lifetime in space, earth satellites and space vehicles must rely entirely upon

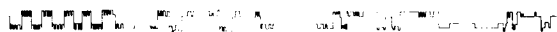


FIGURE 8-26.—University of California experiment output format.

thermal radiation to gain or lose heat to their surroundings. Therefore, an important consideration in the design of space vehicles is the type of surface used for temperature control. For example, although most polished metal surfaces reflect a large part of the sun's rays, they give up the heat they do absorb at a very low rate, and consequently reach high temperature equilibrium in the sun. White paints and enamels, on the other hand, while reflecting most of the incident sunlight, emit heat at a high rate, causing them to reach low-temperature equilibrium. Stability of the thermal characteristics of temperature control surfaces is very important in the design of spacecraft. The space environment can cause changes in the radiation properties of surfaces, leading to inadequate temperature control. The magnitude and rate of such changes is at present not known for most materials.

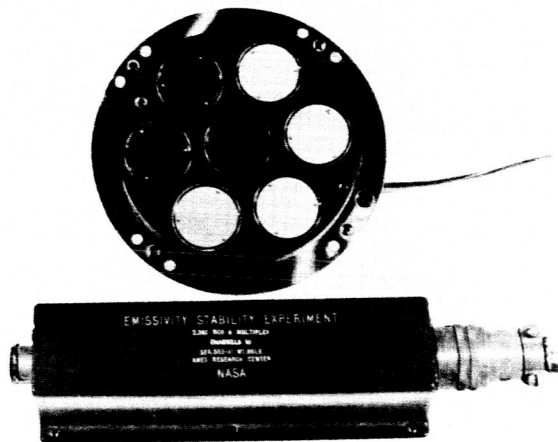


FIGURE 8-27.—Ames Research Center emissivity stability experiment.

A project in progress at the time of this writing at Ames Research Center involved measuring changes in the radiation characteristics of a number of surfaces intended for use in thermal control systems during spaceflight. The experimental package shown in Figure 8-27 was developed to perform these tests on the spacecraft. The experiment consisted of measuring the temperature of several test surfaces during exposure to conditions in space. From

changes in the temperature of the surfaces with time, it was possible to determine the amount and rate of change in the thermal radiation characteristics of the surfaces.

Four basic types of surfaces were studied:

1. A flat absorber which absorbed radiation at a uniform rate throughout the wavelength spectrum.
2. A flat reflector which reflected radiation at a uniform rate throughout the spectrum.
3. A solar absorber which absorbed radiation in the solar wavelength region and was useable to reject heat at its own temperature in the infrared region.
4. A solar reflector which reflected solar radiation and rejected heat in the infrared region.

Six test surfaces were placed in thermally isolated mountings and the equilibrium temperature of each surface was measured in space. Surface temperature provided a measure of the ratio of heat absorbed to heat emitted from the surface. The surface was selected to provide information on the emissivity stability of various coatings applicable to spacecraft temperature control. The surfaces were of three types:

1. Surfaces which were being used or appeared to be useful for temperature control.
2. Surfaces which were expected to change only under certain conditions of space environment.
3. A surface serving as a reference. Comparison of the temperatures of the test surfaces with that of the reference surface provided a basis for evaluating changes in the thermal characteristics of the test surfaces.

The temperature of the surfaces was detected by thermistors. The varying resistance of the thermistors changed the frequency of an oscillator. This, in turn, modulated the frequency of transmitter No. 1 together with the recorded output of either tape recorder 1 or 2 during playback. This was the only experiment signal normally transmitted in real time. Figure 8-28 shows a typical output signal format.

Test surfaces and their relative characteristics are as shown in Table 8-4.

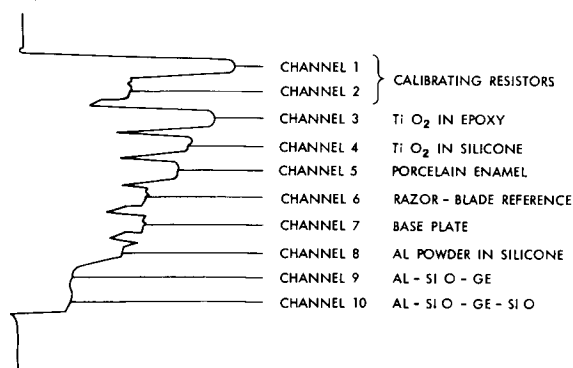


FIGURE 8-28.—Ames Research Center emissivity experiment output format.

Table 8-4

Relative Characteristics of Test Surfaces

Surface	Emissivity	Equilibrium temperatures (°F)
TiO ₂ (in epoxy) -----	0.30	-22
TiO ₂ (in silicone) -----	0.36	-6
White porcelain enamel -----	0.41	8
Aluminum powder and silicone -----	1.0	100
Al-SiO-Ge -----	2.6	200
Al-SiO-Ge-SiO -----	2.9	225
Reference Surface -----	1.0	100

Performance

The analysis of measurements from the sensors of the emissivity experiment showed that some of the surfaces had undergone substantial changes in their characteristics, even when exposed to the space environment for only a few days. As the exposure time was increased, the deterioration appeared to continue. As expected, those surfaces showing the greatest change were the two white paints; titanium dioxide in epoxy, and titanium dioxide in silicone.

Results

A summary of preliminary data reduced from records received at Ames is given in Table 8-5. This table lists the sensor temperatures for various times during a number of early orbits. The times were measured approximately from the start of spacecraft tape recorder playback, and are given here primarily for identification. Comparisons can be made in this table for different orbits in which the razor-blade reference surface has reached the same temperature. The portions of orbit being compared must both be in either daytime or nighttime passes. A comparison can be made between the temperature at 63 seconds in orbit 29 and the value

TABLE 8-5.—Measured Temperature of Sensor on Ames Emissivity Experiment on OSO I

Channel No.		Location and date of acquisition	Temperatures in degrees Fahrenheit							
			3	4	5	6	7	8	9	10
Orbit	Time of playback (seconds)		TiO ₂ epoxy	TiO ₂ silicone	Porcelain	Razor blade reference	Base plate	Al-SiO-Ge	Al-SiO-Ge-SiO	Al-SiO-Ge-SiO
1	39	Fort Myers, day -----	-3	12	7	76	69	85	177	190
	60	3/7/62 -----	-2	13	9	77	70	86	178	191
	124	-----	0	16	11	81	72	88	180	194
	180	-----	1	16	14	83	72	89	184	196
	240	-----	2	16	14	86	73	90	186	199
	300	-----	3	19	16	89	75	93	190	202
10	227	Canaveral, night -----	-55	-----	-39	5	39	9	67	-----
11	0	Canaveral, night -----	-55	-38	-40	4	36	8	68	62
	340	Fort Myers -----	-63	-51	-48	-4	33	-4	59	-----
	55	3/8/62, day -----	-54	-39	-43	1	32	9	67	62
	120	-----	-52	-35	-36	6	35	10	70	66
	192	-----	-46	-28	-32	8	36	14	77	73
	255	3/8/62, day -----	-40	-24	-28	11	36	21	84	79
	296	-----	-36	-21	-25	16	39	24	91	87
	328	-----	-34	-20	-22	17	39	25	96	89

ORBITING SOLAR OBSERVATORY SCIENTIFIC EXPERIMENTS

TABLE 8-5.—*Measured Temperature of Sensor on Ames Emissivity Experiment on OSO I—Continued*

Channel No.		Location and date of acquisition	Temperatures in degrees Fahrenheit							
			3	4	5	6	7	8	9	10
Orbit	Time of payback (seconds)		TiO ₂ epoxy	TiO ₂ silicone	Porcelain	Razor blade reference	Base plate	Al-Si-I	Al-SiO-Ge	Al-SiO-Ge-SiO
13	21	-----	-36	-21	-26	17	38	22	92	91
	70	-----	-35	-18	-22	20	39	27	98	96
	119	-----	-34	-16	-21	21	40	34	99	97
13	183	Fort Myers	-31	-13	-20	25	39	35	104	107
	240	3/8/62	-28	-11	-19	28	41	35	109	111
	320	-----	-27	-7	-14	32	42	40	119	124
14	22	-----	-26	-8	-13	30	41	42	116	123
	62	-----	-24	-8	-13	35	40	42	120	126
	120	-----	-19	-4	-10	39	43	45	124	133
	184	-----	-19	-3	-10	39	41	52	132	140
	240	-----	-18	0	-9	44	43	55	133	146
	320	-----	-18	0	-4	47	42	58	142	151
	29	-----	-11	6	-4	52	43	65	148	159
15	61	-----	-11	6	-4	54	44	65	151	161
	119	-----	-10	8	-2	56	45	69	153	166
	185	-----	-9	9	4	59	46	72	156	170
	240	-----	-6	11	6	63	46	73	160	172
	329	-----	-3	12	5	65	46	74	162	178
	14	-----	0	16	8	69	48	79	166	182
	55	-----	2	16	10	71	49	80	169	183
16	121	3/8/62	3	17	11	73	50	80	172	186
	178	-----	3	19	11	76	50	83	174	188
	242	-----	5	19	12	81	51	83	176	190
	315	Fort Myers, day	7	21	15	83	52	86	180	193
		3/8/62.								
27	24	Fort Myers, day	-68	-54	-58	-10	31	-5	48	42
	57	3/9/62	-65	-51	-50	-9	31	-4	49	46
	124	-----	-63	-46	-45	-2	31	3	59	57
	179	-----	-58	-43	-43	-2	31	3	59	57
	236	-----	-51	-36	-36	2	31	9	70	64
28	324	-----	-46	-29	-32	6	31	12	75	72
	33	-----	-42	-24	-30	11	35	20	83	79
	58	-----	-41	-23	-29	12	36	22	84	82
	122	-----	-35	-17	-23	15	36	23	96	87
	179	-----	-32	-15	-20	21	36	26	100	97
	243	-----	-31	-13	-20	22	36	32	101	101
	335	-----	-29	-10	-17	26	37	36	114	111
	30	-----	-22	-5	-14	30	38	41	116	120
29	63	-----	-21	-4	-13	35	40	45	118	125
	120	-----	-20	-3	-10	36	40	47	123	130
	178	-----	-17	-1	-8	37	40	49	129	138
	243	-----	-14	2	-7	42	40	54	133	143
	322	-----	-11	6	-3	47	42	59	140	151
	71	Fort Myers	-68	-54	-50	-3	35	0	56	51
		3/12/62, night.								
86	3/13/62, night	-62	-49	-45	6	36	1	62	59	
89	3/13/62, day	-14	1	-9	35	39	45	118	128	
103	3/14/62, day	-31	-20	-26	14	33	23	88	88	
118	3/15/62, day	-28	-15	-22	18	36	26	94	94	
177	3/19/62, day	-26	-18	-20	14	38	24	94	89	

given for orbit 89. The reference-surface temperature was 35°F in both cases. For this position in the orbit, the temperature of the epoxy-base paint rose from -21°F in orbit 29 to -14°F in orbit 89, an interval of 4 days. Similar comparisons can be made between other temperature listings, and for the other coatings. A study of the table will reveal a certain amount of normal scatter in the measurements. The changes in temperature noted above, however, were sufficiently marked to indicate definite degradation in the surface characteristics.

EXPERIMENT INTEGRATION

Assurance of the compatibility of the spacecraft electrical and mechanical systems with each experiment was a necessary undertaking from very early in the program until a checkout of the entire prototype spacecraft was completed. Problem areas in which much coordination was required were the outgassing properties of materials used in experiments, the magnetic balance requirements due to experiment detector magnets, the command requirements of certain experiments, the balance of the pointed experiments, the electrical interference between experiments and spacecraft systems or between the experiments themselves, and the experiment checkout procedures required during system and environmental testing.

The amount of outgassing had to be controlled in order to avoid contamination of the

optical surfaces of experiments and to avoid weight loss which could disturb the spacecraft balance. Outgassing problems were controlled by the proper choice of materials on board. If an experiment contained a magnet, additional magnets had to be added and oriented to effect a magnetic balance for the overall spacecraft system.

All experiments were electrically checked on the bench for proper operation prior to installation into the spacecraft. The final calibrations of the University of California and the University of Minnesota experiments had to be performed in the spacecraft. The calibrations required that the spacecraft be suspended in the air a minimum of 10 feet away from any surrounding object or the ground. Due to the hazards to the spacecraft involved in suspending it high in the air, the prototype was used. Calibration was accomplished using various radiation sources to obtain calibration curves with the actual spacecraft mass properly moderating the sources. Various background levels associated with these sources were measured. System and environmental testing required the use of various radiation sources to check out the instruments contained in the spacecraft. Due to the nature of radiation which some of the experiments were measuring, it was not practical to provide all the sources required to completely check out the experiment. Instruments of this type were checked electrically and by using known background count levels as sources.

Chapter 9

TEST PROGRAM

INTRODUCTION

Extensive testing was required throughout the development of the spacecraft to demonstrate feasibility of design and to evaluate the performance of the various systems. This test program proceeded through the developmental, prototype and flight model stages of design.

In many instances, successful testing required the use of special test equipment and fixtures. A description of each of these test devices is included herein as a preface to the actual test program.

TEST EQUIPMENT

Wobble Table

This device was designed to simulate the characteristic wobbling motions of the spacecraft in orbit.

It consisted of a large flat plate simulating the spacecraft wheel and an upper structure mounted on a shaft attached to the wheel. The upper structure contained a second shaft on which the pointed instrument experiments could be mounted for testing. The entire device was mounted on a fixed base.

This base also supported a motor and a drive mechanism for rotating the wheel with respect to the fixed base. Also mounted on the fixed base was a drive mechanism which rotated an eccentric through which the lower wheel shaft bearing was mounted. The eccentric drive speed was adjustable to give the wheel wobbling motions at any frequency from zero to two cycles per second. The driven eccentric could also be adjusted to provide wobble amplitudes from zero to several degrees. Figure 9-1 is an illustration of the wobble table.

Air-Bearing Test Device

The air-bearing test device was designed to duplicate the free gyroscopic motion of the orbiting spacecraft.

The air-bearing test device comprised three main sections: the spherical air bearing, the gyro platform, and the azimuth assembly (see Figure 9-2).

The spherical air bearing provided minimum friction and eliminated as nearly as possible the influence of all external torques. It was capable of supporting up to 600 pounds with an air supply of 30-70 psi, and provided for a spin axis tilt of $\pm 40^\circ$.

The gyro platform provided space for mounting the spin-up gas bottles and controls, the power supply, balancing weights and the spindle required for mounting the azimuth assembly.

The azimuth assembly provided for mounting of the precession jet gas bottles, the azimuth positioning system with eyes and controls, the readout eyes, the nutation damper, and the elevation positioning system with eyes and controls. It also provided a yoke with a transverse spindle for mounting the elevation pointing instrumentation.

The gyro platform spin rate was maintained by three gas jets mounted tangentially to the outer surface of the rotating section. In order to maintain a fixed position, the upper structure was driven by a torque-motor in the opposite direction to the gyro platform at the required speed.

The principal moments of inertia of the fixture were of the same order of magnitude and in approximately the same ratio to each other as those of the rotating section of the spacecraft.

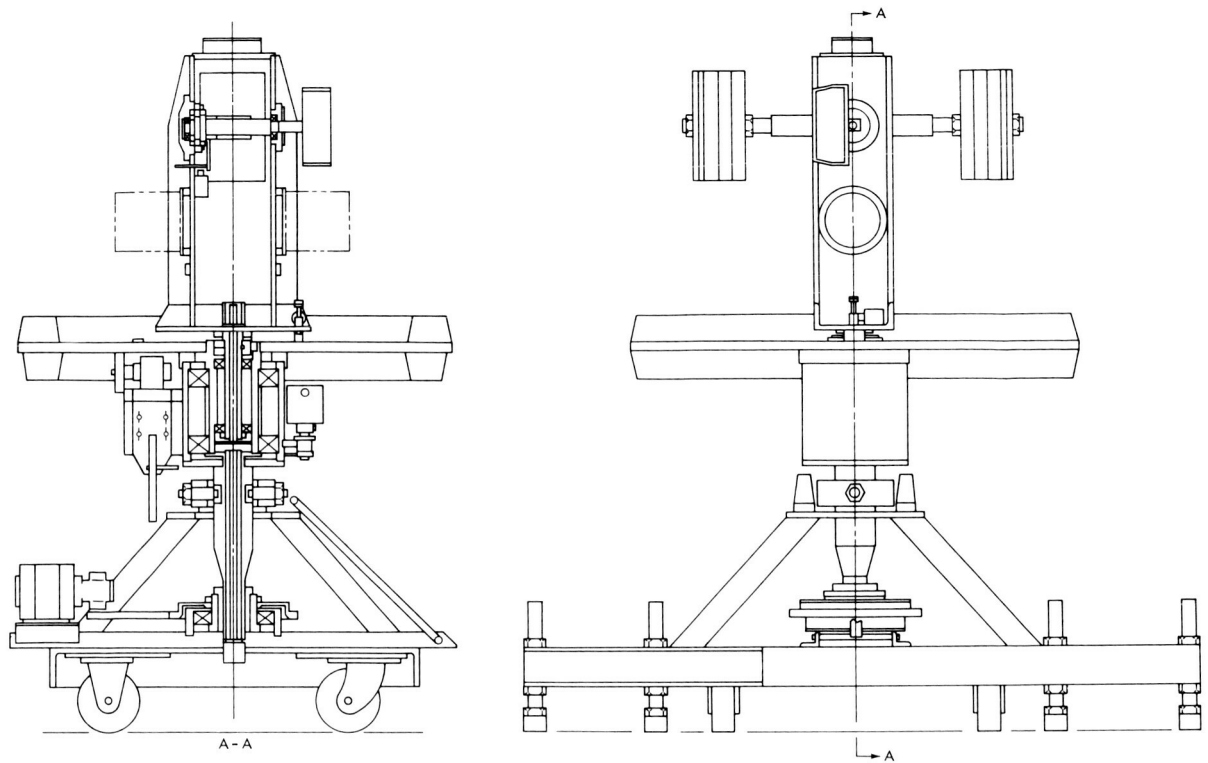


FIGURE 9-1.—Wobble table fixture.

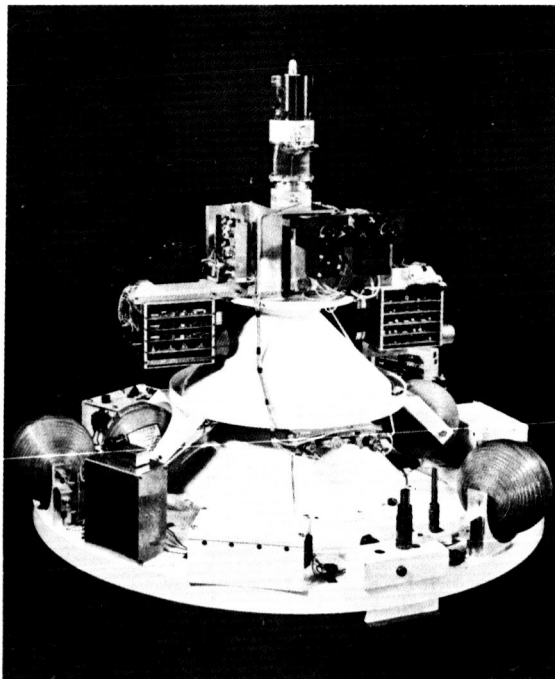


FIGURE 9-2.—Air bearing test fixture.

Indoor Sunlight Source

The artificial indoor sunlight source comprised an aerial camera objective lens, a pair of condensing lens, and a 2100-watt projection bulb as a light source. The power source was a 60-cycle, 220-volt line providing an output of zero to 60 volts dc which was rectified and filtered. Since the intensity of the indoor light source was about 3 percent of the intensity of the sun, it was necessary to use optics of about 30 times more light gathering area than those in the flight model spacecraft for indoor testing. The artificial sun source was positioned at the proper distance from the instrument to simulate a 32-minute diameter *sun*.

Coelostat

In order to do test work using the actual sun on nonportable devices such as the wobble table and the air-bearing device, a sun-tracking reflection system was designed to reflect sunlight into the laboratory. This was accom-

plished using a three-mirror system. The primary mirror was mounted upon a polar axis drive rotating at one revolution per day such that sunlight was always directed through a window into the laboratory. A secondary mirror system was used inside the laboratory consisting of two mirrors, one mirror fixed to the ceiling and one moveable mirror mounted on the floor. The fixed ceiling mirror directed the sunlight reflected from the outdoor primary mirror to the floor mounted moveable mirror. The moveable mirror allowed the final direction of the beam to be adjusted toward either the actual spacecraft or one of the test fixtures so that pointing and monitoring test work could be accomplished.

Periscope

A periscope fixture was designed to allow known sun angle changes to be obtained for checking out the pitch readout and the pitch control eyes. The fixture consisted of two mirrors mounted parallel to each other. One mirror was fixed and one was rotated by means of a micrometer. The micrometer mirror motion was calibrated such that the apparent sun angle shift caused by the micrometer motion could be accurately read.

Shadow Alignment Tool

The shadow alignment tool was a very simple device built to determine when the upper structure of the spacecraft was normal to the solar vector. It consisted of an occulting disk on a long arm with calibrated rings at the base of the arm mounting. This device was also useful for determining the zero pitch angle position of the pitch axis.

Thermal-Vacuum Chamber

The thermal-vacuum chamber provided a simulated space environment for testing the spacecraft and its various systems. It was capable of a wide range of temperature and pressure. The chamber was 10 ft. in diameter, large enough to accommodate the spacecraft and test table. It had three windows to permit observation and allow the use of the artificial sunlight source. The door of the chamber was

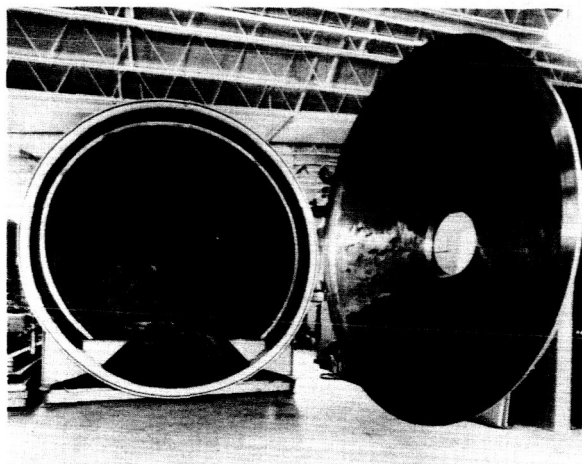


FIGURE 9-3.—Thermal vacuum chamber.

full-opening, allowing convenient installation and removal of the spacecraft (see Figure 9-3).

Electrical lead-throughs were provided to operate the spacecraft, monitor the spacecraft operation, control the pitch and spin gas control systems and monitor temperatures and pitch readout.

Spin Table Test Device

Two spin table test fixtures were built. One was a table fixture, the other was floor mounted. Both tables had a rotatable shaft supported by a rigid base. The shaft had an attach clamp fitting to mate with the attach clamp fitting of the spacecraft. A torque motor installed on the shaft rotated the spacecraft to simulate expected orbital motion. The spacecraft control input connections were provided by slip rings connected through the spin table to an output connector. This permitted the spacecraft to be operated from a console while rotating.

Balancing Machine

A balancing machine was required to place the spacecraft in static and dynamic balance.

The balancing machine had a platform on which the spacecraft could be mounted and rotated. The platform was part of a pendulum yoke. Dynamic unbalance in the payload produced motion in the pendulum yoke, both in phase and amplitude. Unbalance moments

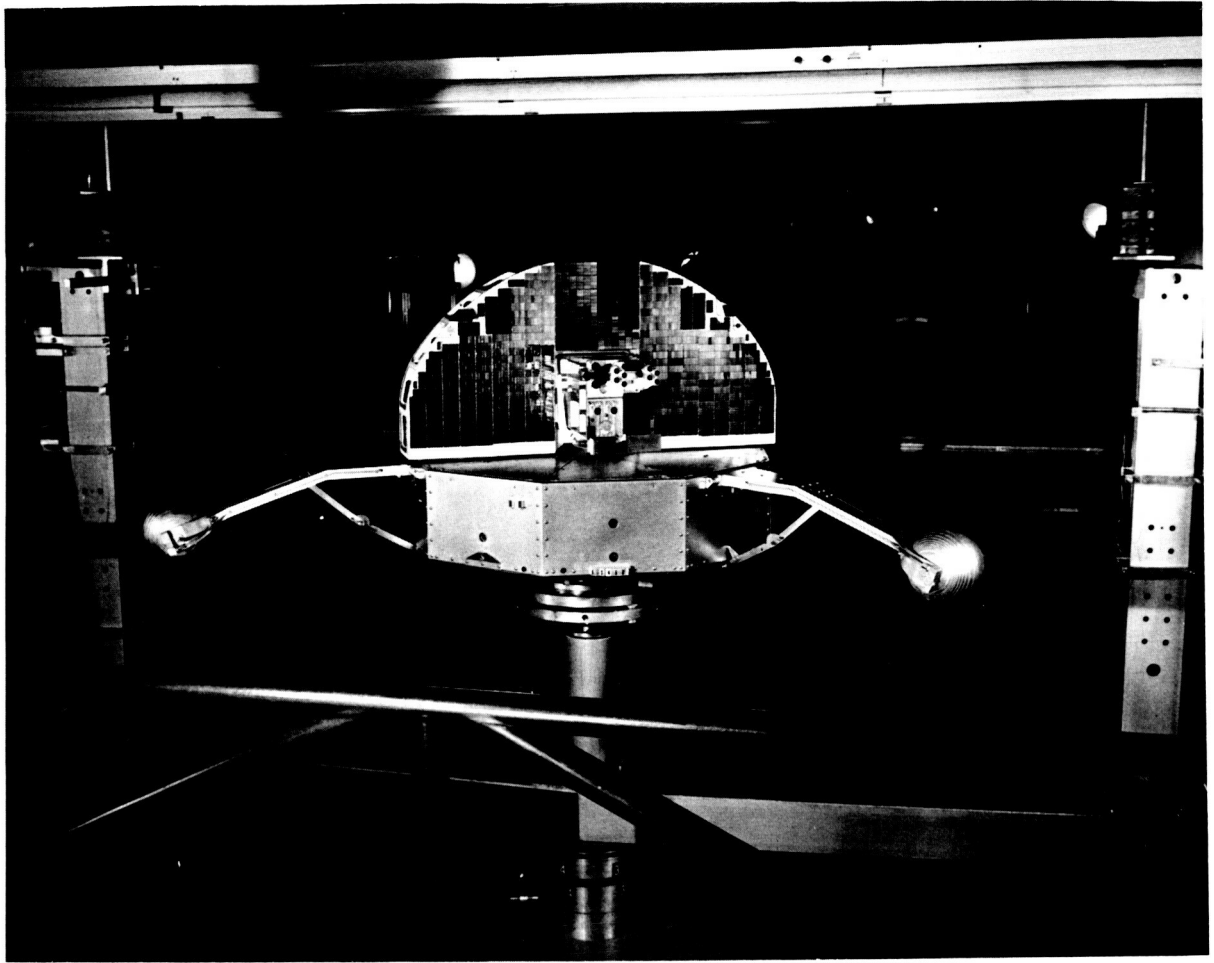


FIGURE 9-4.—Balancing machine.

were determined and proper balancing weights positioned on the spacecraft.

The balancing machine is shown in Figure 9-4.

Pendulum Platform Test Fixture

The pendulum platform was used as a test apparatus because it could be made to perform the circular translational motion experienced by the damper during nutation.

The pendulum platform test fixture (Figure 9-5) consisted of a circular steel plate suspended from a rigidly mounted plate of similar dimensions by three adjustable lengths of steel wire. Multiple wire supports were utilized to eliminate tip and tilt of the platform during lateral oscillation. Frequency of oscillation of the platform was controlled by adjusting the length

of the support wires. The weight of the steel plate could be made such that when the platform was oscillating at a given frequency and amplitude, its total energy (kinetic plus potential) was equal to the kinetic energy of nutation of the spacecraft. Thus, in damping out oscillations of the test fixture, the damper dissipated the same amount of energy as in the actual application. By adjusting the frequency of vibration and the weight of the steel plate, this apparatus could also be used to evaluate damper performance on the air-bearing fixture.

Magnetotropometer

The magnetotropometer was built by Ball Brothers Research Corporation to measure the torque exerted on an object by the magnetic

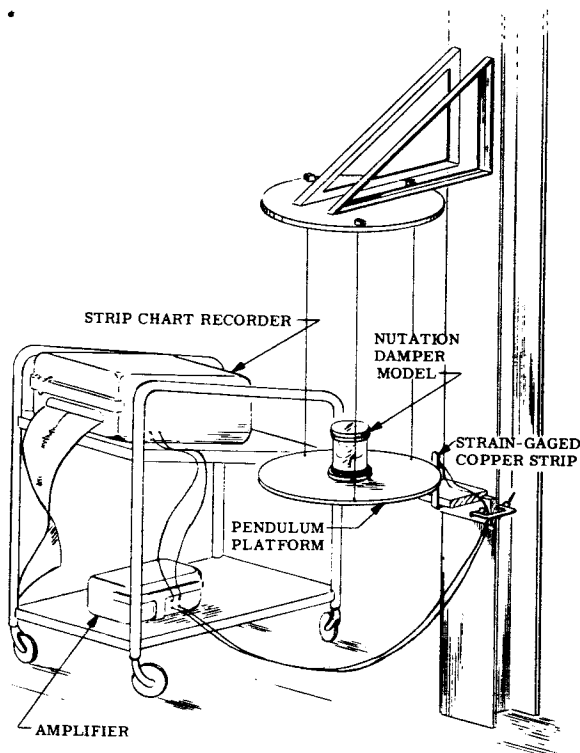


FIGURE 9-5.—Pendulum platform test setup.

field of the earth, thus determining the magnetic dipole moment of that object (see Figure 9-6).

The hull was 45 inches in diameter. The vertical clearance in the hull was about 22 inches. Both tub and hull were deep enough to allow 150 pounds weight in the floating hull. magnetotropometer could accommodate the entire sail structure of the OSO I spacecraft with pointed instruments installed.

Figure 9-6 shows the cylindrical hull floating in a tub of oil. The hull is held in the center of the tub by three slack chains which drape from three stationary posts to a tripod attached to the hull. The chains are attached to the tripod at a small distance from its axis. Therefore, if the hull and tripod are rotated from rest position, the chains apply a restoring torque. If the dimensions of the apparatus, the weight of the chains and the displacement angle are known, the restoring torque can be calculated. Then, knowing the orientation of the object and the direction and intensity of the magnetic field of the earth, one component of the unknown dipole moment can be calculated.

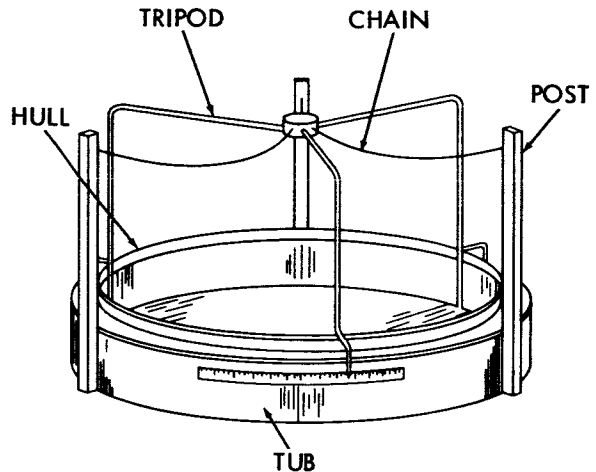


FIGURE 9-6.—Magnetotropometer.

DEVELOPMENTAL TESTING

Control System

In the early stages of development it was necessary to prove that the initial concepts of the control system components were sound. Testing was accomplished on the wobble table and the air-bearing table.

Electronic Subsystem

A mockup electronic system was built from easily obtainable components with no concern given to acquiring or using qualified space flight components. This mockup system was used to demonstrate soundness of the design concept.

The next step in testing the electronic subsystem involved the use of engineering model printed circuitry designed for the spacecraft, but not packaged as anticipated for the final design. This circuitry was installed on the air-bearing test device to demonstrate the performance capability of a two-axis control system.

Pointing Control Subsystem

The various components of the closed loop pointing control were tested individually before the complete loop was checked out. These tests involved measuring the open-loop transfer characteristics of the servo-amplifiers, the power handling capability of the servo motor

driving circuits and the angular characteristics of the error detectors.

At the completion of the subsystem tests the entire pointing control servo system was bench tested in preparation for the system testing on the wobble table and the air-bearing table.

To determine if the accuracy of the pointing control system was well within the specified requirements, tests were conducted on the wobble table. These tests were conducted by driving the wobble table at the anticipated orbital spin rates with the wobble angles applied by means of eccentric adjustment. The pointing control was shown to be adequate for anticipated disturbances.

The upper structure solar acquisition system, as originally designed, proved to be inadequate. Testing was accomplished by spinning the wobble table up to the required spin rate with the pointing control electronics turned off, allowing the upper structure to be dragged up to the wheel rotation rate by the friction of bearings, slip rings, and torque motor brushes, and then turning on control system power.

The solar acquisition method was changed and the new design was demonstrated to be adequate.

Gas Control Subsystem

The gas handling components of the gas control system (spin control and pitch control) were tested before being integrated into the control systems on the test apparatus. This involved the leak testing of valve, regulators, and fittings, the flow testing of regulators and filters, calibration of pressure transducers and nozzle design testing to determine nozzle efficiency and impulse properties.

Nutation Damper

All quantitative results obtained from the preliminary tests of the nutation damper on the pendulum platform test fixture were based on single-plane motion of the system with the platform performing simple pendulum motion. The following information justified this procedure.

During simple pendulum motion, the total energy of the platform is equal to either the maximum kinetic energy or maximum potential

energy. The latter quantity is given by the expression:

$$V = gh = Wl(1 - \cos \theta) = \frac{Wl(1 - \cos^2 \theta)}{1 + \cos \theta} = \frac{Wl \sin^2 \theta}{1 + \cos \theta} \quad (9.1)$$

where

h = maximum change in height of platform
 l = length of pendulum arms
 θ = angular displacement of platform from equilibrium position
 w = weight of platform
 g = gravitational acceleration

When performing circular translational motion of the same angular amplitude, the platform has the same potential energy as given in Equation (9.1). In addition, it also possesses the following kinetic energy:

$$T = \frac{1}{2} mv^2 = \frac{1}{2} mr^2 \omega^2 = \frac{1}{2} m(l^2 \sin^2 \theta)(g/l \cos \theta) = \frac{Wl \sin^2 \theta}{2 \cos \theta} \quad (9.2)$$

where

m = mass of platform
 v = linear velocity of platform
 r = radius of the circle described by the center of gravity on the platform during this motion
 ω = angular velocity of center of gravity about the center of the circle

and the other symbols are the same as in Equation (9.1). The total energy of the platform in this case is:

$$E = T + V \quad (9.3)$$

It is readily seen from Equations (9.1) and (9.2) that for small θ , the kinetic and potential energies of the platform performing such motion are twice that which it possesses when performing simple pendulum motion of the same angular amplitude. It is shown in the derivations on

page 248 that the spherical pendulum damper dissipates energy at twice the rate of the simple pendulum damper. The damping rate of the damper-platform system is, therefore, the same during either type motion.

It can also be determined from Equations (9.1) through (9.3) that the total energy of the platform during either type motion is directly proportional to the weight of the platform. Since the energy dissipation rate of the damper is independent of this parameter, it was reasoned that the time constant of the damping rate for a given damper-platform configuration should be directly proportional to platform weight. Therefore, rather than change the platform weight to duplicate the total energy of either the spacecraft or air-bearing fixture, a single series of tests with fixed platform weight was conducted. Nutation damping rate of the damper for a particular spacecraft or air-bearing fixture configuration was then calculated from the results of these tests. This approach seemed advisable in view of the constantly changing spacecraft and air-bearing fixture designs during the development program. The calculated values for damping rates using this simplified approach were verified by the results obtained from the analog computer in the mathematical analysis of the problem.

Instrumentation was provided to measure the amplitude and frequency of oscillation of the platform. A thin metal strip was positioned against the edge of the platform in such a way that any lateral oscillation of the platform resulted in bending of the metal strip. A strain gauge was attached to the strip to measure this deflection. Output of the strain gauge, which was an approximately linear function of platform displacement, was amplified and fed into a strip-chart recorder. The strip-chart plot of platform displacement as a function of time was then used to calculate the rate at which free oscillations of the platform were damped out.

Damping Efficiency—In the first series of tests conducted on the pendulum platform test fixture, the cylindrical pendulum bob was used in the nutation damper. At the time of these tests the theoretical analysis of the damper-

platform motion had not been performed and only the approximate results of the steady forced vibration analysis were available. Furthermore, the spacecraft design program was still young and values for spacecraft parameters being quoted at that time were changed considerably during subsequent design modifications. The numerical results of these early tests were therefore not directly applicable to final damper design. They were important primarily in bracketing damper parameters more closely than had been possible from the results of the steady forced vibration analysis included in Chapter 2. The importance of performing a mathematical analysis of damper-platform motion to optimize damper parameters rather than using experimental results for this purpose was also made evident by the results of these tests:

1. Maximum damping efficiency did not occur when the natural frequency of vibration of the damper was equal to the disturbing frequency as had been indicated by the steady forced vibration analysis. The slight difference in value of these two frequencies at the tuned condition was observed to be influenced by the damping constant of the damper.

2. The damping constant, in turn, was observed to have an optimum value for fixed values of the two frequencies. When using fluid with a kinematic viscosity of 100 centistokes or greater, the damping rate of the system decreased with increasing viscosity. On the other hand, fluid of less than 10-centistoke viscosity was determined to be too thin since *beating*, or the alternate transfer of energy between the damper and platform, occurred. These results indicated the necessity for a parametric study in order to arrive at an optimum damper configuration. This study was obviously more readily conducted on an analog computer than with the test setup.

The series of tests performed using the spherical pendulum bob served two purposes; namely, to verify the results of the mathematical analysis of system motion and to determine experimentally the association between damping constant of the damper and the kinematic viscosity of the fluid used to fill the damper. This latter determination was neces-

sary because of the lack of theoretical information available that was pertinent to this particular geometry. Stokes' equation, relating damping force and kinematic viscosity for a sphere immersed in a fluid, yielded damping constants considerably smaller than those calculated from experimental results. This discrepancy was due to the small volume of fluid used. The flow of fluid over the surfaces of the cylindrical container probably contributed significantly to the damping force. Three different series of tests were made with this configuration using 10-, 50-, and 100-centistoke silicone fluid in the damper. With the natural frequency of the damper held fixed, the frequency of vibration of the pendulum platform test fixture was varied by changing the length of the support wires of the platform. From the recorder plot of platform amplitude versus time, the time constant for the decay rate of the amplitude was determined in each case. The graph of time constant versus platform frequency for the tests using 50- and 100-centistoke fluid is presented in Figure 9-7. A similar plot for the tests using 10-centistoke fluid could not be made since beating occurred during these tests, and the amplitude of vibra-

tion of the platform did not decay exponentially.

From the curves in Figure 9-7, the following conclusions can be made:

1. The time constant for the decay rate of the amplitude decreases with decreasing fluid viscosity, but a moderate variation in viscosity will not seriously affect damper efficiency.

2. As fluid viscosity (and therefore damping constant) decreases, the damper must be more sharply tuned to disturbing frequency to obtain good damping efficiency.

3. Maximum damper efficiency occurs when the natural frequency of the damper is slightly less than the disturbing frequency.

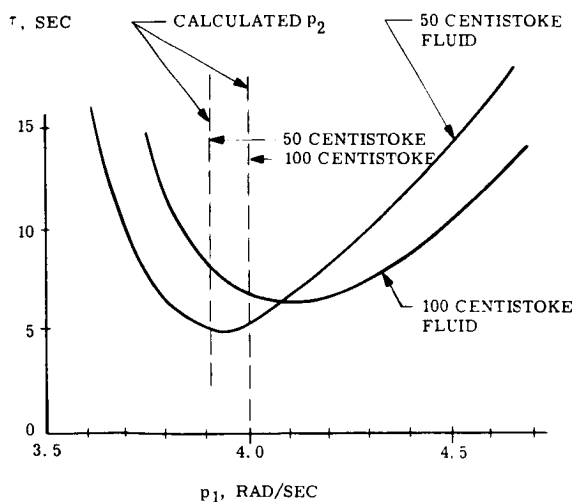
4. For the range of fluid viscosities tested, a time constant of twice the optimum value is obtained when the disturbing frequency is approximately 10 percent above or below nominal value.

These qualitative results of the experimental tests were later substantiated by the results of the mathematical analysis of the problem.

In order to determine the minimum viscosity fluid that could be used without encountering beating, additional tests were conducted with 20 and 30-centistoke fluid. Beating occurred using the 20-centistoke fluid and did not occur when the 30-centistoke fluid was used. Since only a small difference in time constant could be detected with a change of 10 centistokes in fluid viscosity, further optimization of damping constant was not attempted. The 30-centistoke fluid was selected for use in the spacecraft damper.

To obtain an accurate evaluation of damper performance a simple setup was devised. This instrumentation consisted of a photodetector (a 1-cm x 2-cm silicon cell) mounted on the nonrotating upper section and partially hidden from the constant-direction light source by a knife-edge whenever the spin axis of the test fixture was nearly vertical. Slight variation in position of the spin axis resulted in variation of the amount of the solar cell surface exposed to light. Since output current of the solar cell is proportional to the amount of light it detects, it was a simple matter to use this output signal to determine test fixture attitude during nutation.

With the attitude control system operating and the wheel spinning at nominal rate, torques



NOTE

- p_1 = undamped natural frequency of platform
 p_2 = undamped natural frequency of damper

FIGURE 9-7.—Time constant versus platform frequency.

were applied to the test fixture to induce nutation. Because of its relatively large amplitude and low frequency, the resulting motion of the fixture was easily followed visually. When the nutation damper was caged, this motion continued indefinitely. With the damper uncaged, nutation was quickly damped out, and the fixture remained "at sleep" until another torque impulse was applied.

The recorder plot of nutation amplitude versus time was compared with similar plots obtained for the same damper during tests on the pendulum platform. The time constant of about 8 seconds measured for the air-bearing tests checked satisfactorily with the value calculated from the results of the pendulum platform tests, taking into account the greater amount of energy the damper had to take out on the air-bearing fixture for vibrations of equal amplitude. Thus, these tests proved conclusively that the damper would damp out nutation of a gyroscopic body at the rate predicted from the results of the pendulum platform tests and two-body free vibration analysis.

Frequency Tests.—Since the frequency of vibration of the damper during orbital flight could not be checked by direct measurement in the laboratory, it was important that the frequency equations derived for the damper in the gravity field be fully substantiated by tests. It was reasoned that if the undamped frequency of the damper in the inverted position could be predicted accurately from the knowledge of the necessary damper parameters and the undamped frequency in the upright position, the effect of gravity on frequency would be firmly established.

The procedure for calculating frequency of the inverted damper from a known frequency of the upright damper was also followed in tuning the damper to nutation frequency of the air-bearing fixture. This was necessary because of the low frequency of nutation. The wire pendulum arm required to obtain the tuned frequency of the fluid-filled damper in the inverted position was unable to support the weight of the pendulum bob in air. The undamped frequency in the inverted position could, therefore, not be measured directly. Calculation of undamped frequency from the measured damped fre-

quency was not feasible because of the difficulty of measuring damped frequency of the pendulum in the sealed, fluid-filled assembly. Attempts were made to measure damped frequency with one end of the case removed, but it was found that the damping constant of the fluid with the end removed was different from its value in the sealed unit.

Verification of the frequency equations for the pendulum in the upright and inverted positions was accomplished with a spring wire of sufficient diameter to support the bob weight in the inverted position. Frequency measurements were made by visual observation and the use of a stop watch. Within the accuracy of this measurement technique, the two measured frequencies were found to agree with the calculated values. During these tests, the end of the wire attached to the bob was adjusted to various positions within the adjustment range of the setscrew. This determined the extent of the double pendulum effect created whenever the wire end was not positioned at the center of gravity of the bob. It was found that by computing an effective wire length from the measured upright frequency in each case, the frequency of the inverted damper could be predicted with one percent accuracy. The double pendulum effect was therefore considered to be negligible.

Integrated Control Systems

The laboratory tests of the gas control system and observation of the concurrent behavior of the sun sensing eyes, torque motors, electronic control circuits and the nutation damper required a freely rotating platform. For this reason the entire engineering model of spacecraft control systems was tested on the air-bearing test fixture.

The control system was tested for pointing accuracy, solar acquisition, spin-up, pitch correction, day-night cycling, data acquisition and data transmission. The test fixture was given wobble during simulated night condition and nutation during day conditions. All possible motions of the actual spacecraft were simulated and the resulting systems reactions were measured.

Many minor changes and adjustments were made to the engineering model. One unforeseen problem that would have caused the actual spacecraft to fail was detected as a result of the air-bearing table testing. The problem, an unstabilizing interaction between the pitch control and a large nutational disturbance was quite easily solved by the addition of a time delay in the turn-on of the pitch control gas jets.

Mechanical Structure

During the developmental testing program for the control system, a concurrent effort was underway to develop the mechanical structure for the spacecraft. After this structure was built and assembled, static structural load testing was performed where necessary. Tests were also run to determine that the erection of the extendable arms at launch spin rates did not damage the spacecraft or produce any appreciable wobble input to the spacecraft.

The first such test was performed using the structural wheel assembly mounted on a spin table. Later, when the arm configuration was changed, dummy wheel compartments were constructed on the spin table using production quality rim panels and further tests were made. These tests consisted of spinning the table to a predetermined velocity with the arms in the stowed position and then firing electrically operated explosive pin pullers to release the arms to the extended position. All tests were recorded with a high speed motion picture camera.

Throughout these tests the panels were observed to deflect considerably at peak loads, but no permanent deformations occurred and no design changes were made.

The structural wheel model was loaded with dummy packages to simulate control system components and experiments, and was subjected to vibration tests and steady-state acceleration. During steady-state acceleration, strain gages were utilized in various areas. The results of this test caused a redesign of the upper wheel casting. To prove the integrity of the redesign, the casting was statically loaded and deflections were measured. No strain gage measurements were made after redesign.

Other parts subjected to strain gage measurements were the arm mounting rim panels, decks, and ribs. Other than in the upper wheel casting, the highest stress measured was 1100 p.s.i. on a deck panel. All other stresses were so low as to be negligible.

Data Acquisition and Command System

The subassemblies of this system were subcontracted by Ball Brothers Research Corporation. All developmental testing of these subassemblies was performed by the subcontractor. Ball Brothers Research Corporation subjected each subassembly to a rigorous qualification procedure upon receipt from the subcontractor.

A detailed analysis of the antenna design was made by running patterns of the actual array on an antenna tower utilizing the full scale structural model spacecraft as a test model.

PROTOTYPE TESTING

Subsystems Tests

All of the prototype spacecraft subsystems, i.e., components of the control systems, mechanical structure, and data acquisition and command system, were tested extensively in the laboratory prior to formal environmental acceptance testing. These tests primarily involved thermal cycling over the expected orbit temperature extremes to assure that the thermal compensation work had been properly performed, and testing at the expected power supply voltage extremes to assure proper performance under the anticipated worst case conditions.

Conformance to the component environmental test specification was required of all spacecraft subsystems and components. This specification required that the subsystem or component be successfully operated before, during (in most cases), and after a high and low temperature thermal-vacuum test, and a three-axis vibration test.

The prototype subassembly thermal-vacuum test specifications were as follows:

Temperature	Duration (hours)	Vacuum (mm Hg or less)
+50° C-----	24	10-2
-20° C-----	24	10-2

TEST PROGRAM

- The component vibration test specifications were as follows:

Frequency range (cps)	Acceleration (peak g)	Displacement (inches)
5-17-9	-----	0.4
17.9-50	6.4	-----
50-1000	15.6	-----
100-2000	21.6	-----

Total duration—15 minutes.

Wheel Structure

After the fourth complete vibration test of the prototype mechanical structure, two decks carrying large, heavy experiments were found to have cracks in the side flange bend radii. These failures were considered to be insignificant because test results were far in excess of flight load expectations at this point. However, a program of careful investigation of the bend area on other decks was instigated.

There was one other consideration of the loading under actual flight conditions, the nature of the loads on the deck was changed by the thrust acceleration, as shown in Figure 9-8.

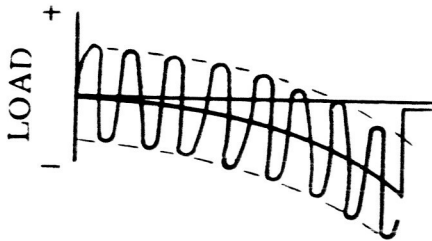


FIGURE 9-8.—Flight loading vibration plus thrust.

Superimposing the vibration on the thrust curve shows that the load is predominantly downward on the deck. Thus, full cycle reversals during launch would not have occurred as frequently as during vibration testing. Furthermore, the time duration of vibration input during flight was only 5 minutes, compared to 45 minutes of testing time. This point is made not to suggest that the design criteria should have been changed, but to indicate that in evaluating the cracks which appeared there was justification in accepting the design as qualified.

Elevation and Azimuth Castings

The first tests conducted on the prototype casting were commenced in August 1960 at the

Martin Company in Denver. These tests consisted of 1220 minutes of low-level sinusoidal input sweeps in three axes, 15 minutes of high-level sinusoidal input sweeps in three axes, 12 minutes of random vibration input in three axes, and 10 minutes of 15 g sustained acceleration input in the thrust axis.*

Figure 9-9 shows the structural model in place on the shaker, and Figure 9-10 shows the structural model in place on the centrifuge.

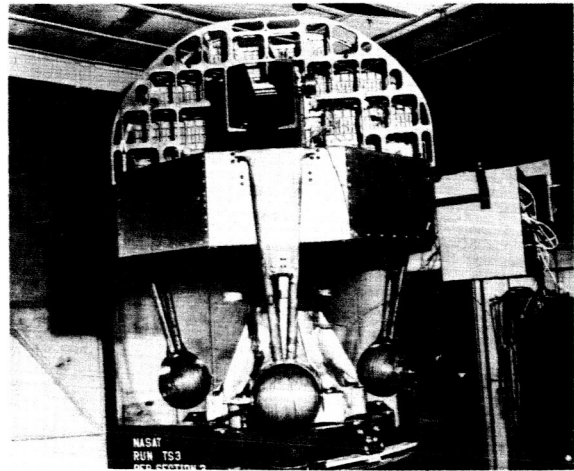


FIGURE 9-9.—Structural model vibration test.

During these tests no failures or problems occurred in either casting. The devices used to lock out both elevation and azimuth freedom failed and required redesign, but these did not directly affect the castings.

During the acceleration tests, seven strain gage readings were obtained on various points of the azimuth casting and four readings on the elevation casting. No unduly high stress levels were indicated.

The structural model azimuth casting was later used for a vibration test of the honeycomb solar cell panels. The setup consisted of the casting hard-mounted on a fixture, with the sail structure and solar cell panels attached. The assembly was not balanced. The test was run for 17 minutes with sinusoidal swept inputs up to 17 g. At this time, during a transverse

*See *Martin Test Report*, Env. 1192, Proof Testing of the Ball Brothers Prototype Model S-16, Orbiting Solar Observatory Satellite, 10 September 1961.

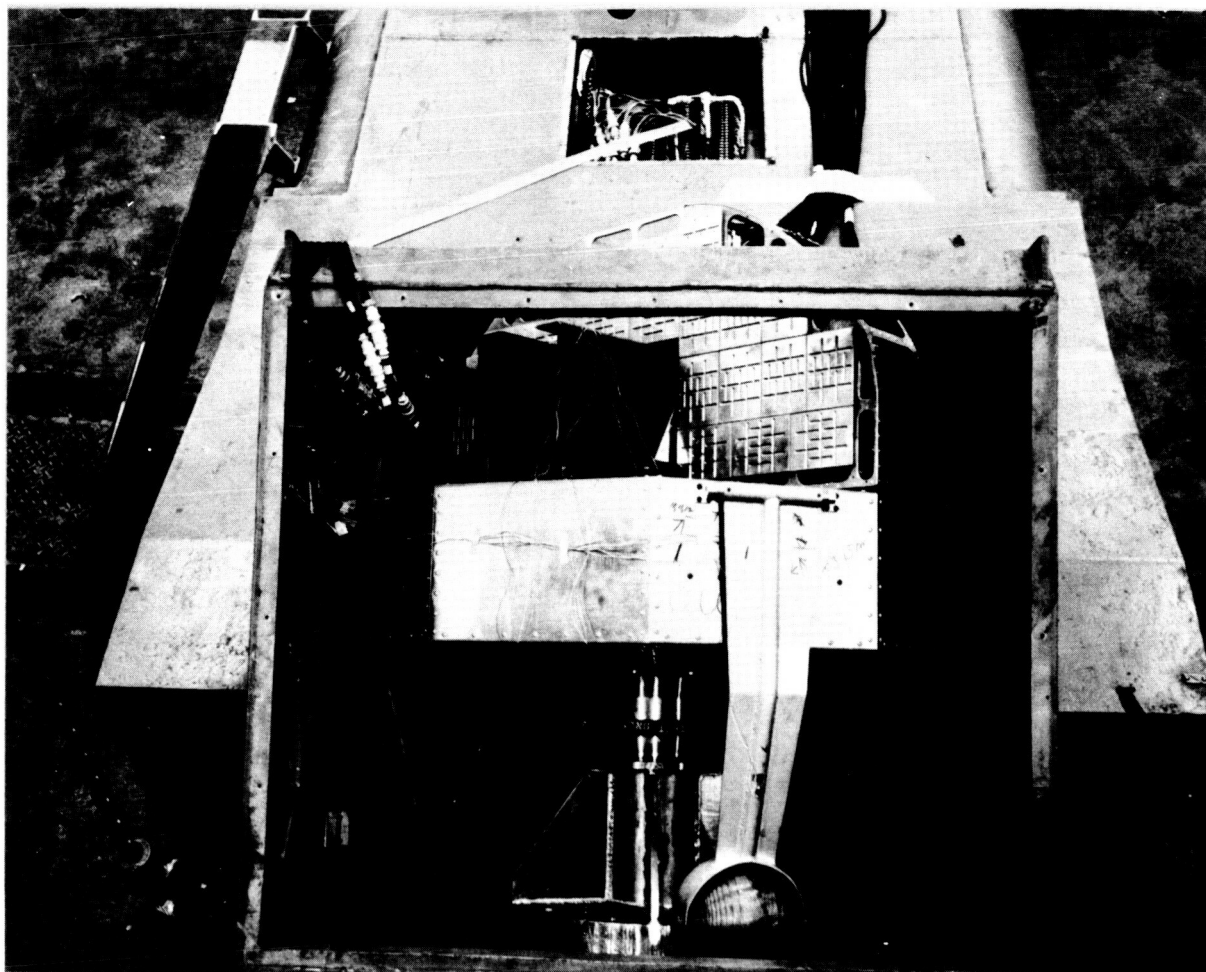


FIGURE 9-10.—Structural model accelerator test.

test, the control accelerometer mounted on the fixture fell off, and the vibrator exciter attempted to put infinite forces into the fixture. The exciter was turned off by overload cutouts, but the input loads reached were not known. After this occurrence, the azimuth casting was found to be badly cracked in a lower corner radius. The crack was considered to be due to a test failure rather than a component failure.

Bearings

After the vibration and acceleration tests at Martin Company, the shafts and bearings were disassembled for inspection. The bearings still functioned properly, but the azimuth shaft was badly galled particularly at the lower bearing. As a result of this condition the shaft fit toler-

ances were changed. There were no further problems during subsequent testing.

Sail Structure

No problems of a structural nature were encountered in the sail structure during the structural model tests. For these tests, the original modular stamped solar cell panels were used, and two of the modules had dummy solar cells bonded in place.

After the structural model tests, the sail structure was removed and independent vibration tests were conducted on it. It was found that during thrust axis vibration, the horizontal tie members were flexing with large amplitudes, causing excessive crosstalk output in the transverse plane. Stiffeners were added to the hori-

zontal members and the test was re-run with satisfactory results. This same test was repeated when the honeycomb panels replaced the modular stamping. Even after the test failure, there was no damage to the sail structure.

Vibration tests were conducted on the nutation damper to evaluate the caging mechanism. As a result of these tests the original design was modified. At first an O ring was utilized for a bob seat during caging but it was found that the bob rotated, twisting the pendulum wire and deforming it. Also, the previous design utilized a solenoid for releasing the bob, but this would not successfully withstand vibration. The solenoid was replaced with a squib actuated pin puller unit, and the caging seat was changed to metal and later serrated to prevent the bob from rotating during shake.

Summary

After all prototype subassemblies had received rigorous qualification and testing, they were installed in the prototype spacecraft and integration tests were begun.

Comprehensive System Tests

After all components and subsystems had been individually tested and successfully integrated, the first comprehensive, covers-off, operational system test was performed. This was an extensive test, in accordance with a formal test procedure, designed to provide assurance that all subsystems in the prototype model were working properly, and that no adverse interactions between subsystems and instruments had occurred.

In these tests, every measurable parameter of significance to spacecraft performance was measured. All work was accomplished with the spacecraft mounted on the spin table.

Comprehensive system tests were performed on the prototype spacecraft before during and after environmental tests.

Prior to the actual environmental testing, it was necessary to balance the prototype spacecraft, measure the moments of inertia, and determine the location of the center of mass.

Static and Dynamic Balance

The wheel structure had to be aligned to spin about the figure axis and the structure had to be aligned about the spin axis to prevent a dynamic unbalance which would have caused a wobble in the orbiting payload. The upper structure had to be dynamically in balance to prevent the night spin-up of the upper structure from introducing wobble. Dynamic balance was also necessary to keep the spacecraft within the alignment tolerances of third stage thrust for the stowed launch condition.

Balance Tolerances

Static Balance—Launch Condition.—According to Douglas Report No. 7843144, the center of mass of the spacecraft had to lie within 0.0033 inches of the axis of the master diameter of the attach fitting. This meant that the spacecraft static balance had to be within 25 oz-in.

Dynamic Balance.—Launch Condition.—According to Douglas Report No. 7843144, a principal inertial axis had to lie within θ radians of the axis of the master diameter of the attach fittings, where

$$\theta = 0.001 \frac{I_s + 4.5}{I_s - I_T}$$

I_s is the moment of inertia about the spin axis in slug-ft², and I_T is the transverse moment of inertia in slug-ft².

For the OSO I it was determined that

$$I_s = 19.7 \text{ slug-ft}^2$$

$$\text{and } I_T = 15.3 \text{ slug-ft}^2,$$

$$\text{so } \theta = 0.0055 \text{ radian, or } 18.9 \text{ minutes.}$$

This meant that the dynamic balance had to be within 1800 oz-in.²

Orbital Condition

In orbit, static balance is not important. However, all methods producing good dynamic balance also produced good static balance. Dynamic unbalance of the wheel causes the spacecraft to wobble. If the wobble becomes too large, errors in experiment pointing causing higher servo power result. Therefore, an arbitrary limit of one minute of wobble was

chosen. This meant that the wheel had to be dynamically balanced within 144 oz-in.²

Dynamic unbalance of the sail would have caused a wobble at night which would have become nutation when the upper structure acquired the sun the following day. An arbitrary limit of 10 minutes was chosen for night wobble. This meant that the wheel and sail together had to be dynamically balanced within 2150 oz-in.²

Combined Tolerance

Control over stowed balance and orbital balance is not independent. All the balance requirements were met as follows:

1. With the arms extended, the wheel was balanced within 10 oz-in. static and 140 oz-in.² dynamic.
2. With the arms extended and the sail locked in the launch position, weights were added to the sail to balance the spacecraft within 10 oz-in. static and 300 oz-in.² dynamic.
3. With the arms stowed in launch position and the sail locked in launch position, the spacecraft was checked to assure a balance within 20 oz-in. static and 100 oz-in.² dynamic.

Theory of the Balancing Method.

Consider a right-circular cylindrical object that is to be balanced statically and dynamically about its figure axis. It is possible to bring this object into complete balance by attaching two weights, one at each end. The problem is to find out experimentally what weights to put on and where to put them.

Figure 9-11 shows a cylindrical object of radius a , spinning about its vertical figure axis at ω radians per second and free to rock about

one horizontal axis. The apparatus can be changed to use the lower pair of trunnions, allowing rocking about a different horizontal axis. If the object is balanced, it has no tendency to rock about either axis.

To determine the effect of an unbalanced condition, a small mass, m , is placed on the upper edge of the cylinder at an angle, α , from a reference line drawn on the cylinder. This mass will cause rocking torques in two ways. First, the weight of m causes a torque about the rocking axis of $mga \sin(\omega t + \alpha)$. Second, the mass applies a centrifuge force of $m\omega^2$ at a distance $l_4 - l_2$ above the rocking axis, producing a torque of $m\omega^2(l_4 - l_2) \sin(\omega t + \alpha)$. The total rocking torque, then, is

$$ma[g + \omega^2(l_4 - l_2)] \sin(\omega t + \alpha). \quad (9.4)$$

If the mass were placed at the bottom edge of the cylinder, l_4 would be replaced by l_3 . If the apparatus were changed so that it pivoted on the lower trunnions, l_2 would be replaced by l_1 .

Note that even if a weight is so placed that its static and dynamic effects just balance and it produces no rocking torque about one pair of trunnions, it does produce a rocking torque about the other. Thus, if the cylinder is out of balance, it tends to rock about at least one pair of trunnions.

Further note that even if the cylinder is initially out of balance it is possible to place some weight on one end of the cylinder in such a position that the rocking torque about one pair of trunnions due to the initial unbalance is just cancelled by the rocking torque due to the weight.

To simplify the notation, let

$$\left. \begin{aligned} a[g + \omega^2(l_4 - l_2)] &= A, \\ a[g + \omega^2(l_3 - l_2)] &= B, \\ a[g + \omega^2(l_3 - l_1)] &= C, \\ a[g + \omega^2(l_4 - l_1)] &= D. \end{aligned} \right\} \quad (9.5)$$

and

An experimental method of balancing the cylinder if it has an unknown initial unbalance may now be discussed.

With the upper trunnions in use, attach an experimentally determined weight, m_1 , to the

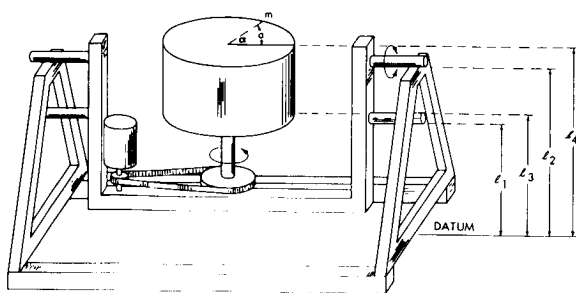


FIGURE 9-11.—Simplified balancing machine.

top rim in such a position, α , that the apparatus does not rock. This weight applies a torque of

$$m_1 A \sin (\omega t + \alpha)$$

Now shift to the lower trunnions and attach some weight, m_2 , to the bottom rim at β so that the apparatus does not rock. The two weights together apply a torque of

$$m_1 D \sin (\omega t + \alpha) + m_2 C \sin (\omega t + \beta).$$

Two weights must now be found, one for the top rim and one for the bottom rim, such that the two together will apply the same torque as m_1 about the upper trunnions and the same torque as m_1 and m_2 together about the lower trunnions. If m_1 and m_2 are removed, and these new weights installed, the apparatus will not rock about either pair of trunnions. The cylinder will be balanced.

Designate the top weight, M_T , and its position, θ . Designate the bottom weight, m_b , and its position, ϕ . They apply a torque about the upper trunnions of

$$m_T A \sin (\omega t + \theta) + m_B B \sin (\omega t + \phi),$$

and a torque about the lower trunnions of

$$m_T D \sin (\omega t + \theta) + m_B C \sin (\omega t + \phi).$$

Then

$$m_1 A \sin (\omega t + \alpha) = m_T A \sin (\omega t + \theta) + m_B B \sin (\omega t + \phi), \quad (9.6)$$

and

$$m_1 D \sin (\omega t + \alpha) + m_2 C \sin (\omega t + \beta) = m_T D \sin (\omega t + \theta) + m_B C \sin (\omega t + \phi). \quad (9.7)$$

Multiply Equation (9.6) by $\frac{D}{A}$, and subtract Equation (9.7) to get

$$m_B \left(\frac{DB}{A} - C \right) \sin (\omega t + \phi) = -m_2 C \sin (\omega t + \beta). \quad (9.8)$$

From this equation, it is seen that $\phi = \beta$ and

$$m_B = m_2 \frac{AC}{DB - AC}$$

or

$$m_B = m_2 \frac{[g + \omega^2(l_4 - l_2)][g + \omega^2(l_3 - l_1)]}{\omega^4(l_2 - l_1)(l_4 - l_3)} \quad (9.9)$$

Now we remove m_1 and m_2 and place m_B (whose value is now known) exactly where m_2 was. Then, still pivoting on the lower trunnions, find a weight for the top rim that stops the rocking. This is m_T , and the cylinder is balanced.

The steps in balancing the cylinder, then, are these:

1. Using the upper trunnions, find a trial weight which, when mounted at the proper point on the top rim, stops the rocking. Leave it on.

2. Using the lower trunnions, find a trial weight which, when mounted at the proper point on the bottom rim, stops the rocking.

3. Remove and weigh the bottom trial weight. Replace it with a weight greater by a known factor. Remove the top trial weights.

4. Still using the lower trunnions, find a weight which, when mounted at the proper point on the top rim, stops the rocking. Mount it. The cylinder is balanced.

Balancing Method

Aligning the Table.—Before the spacecraft was mounted on the balancing machine, alignment of the table to the true bearing axis of the spindle was checked with a dial indicator. The indicator was set up to measure vertical motion 4.5 inches from the center of the table. The table was adjusted for a reading less than 0.0003 inch TIR. The table was then aligned within 0.12 minutes of arc.

Mounting the Spacecraft on the Table.—Since the balancing machine measures unbalance about its own bearing axis, it is important that the spacecraft be accurately centered on the machine.

On the OSO I the only convenient indicating surface was the alignment nubbin at the top of the sail. With the spacecraft on the balancing machine, but not rotating, indication was from the balancing machine frame to this nubbin. The sail was rotated and the nubbin was adjusted to run within 0.0007 inch TIR.

This meant that the nubbin was well aligned to the spacecraft bearing axis. Then, with the sail locked, the balancing machine spindle was rotated and the spacecraft was moved horizontally on the table until the nubbin ran within 0.0007 inch TIR. This meant that if the nubbin was to lie on both axes, with both axes perpendicular to the attach fittings, then the two axes were coincident. The spacecraft was now aligned to the balancing machine.

Selecting the Pivot Points.—The best sensitivity and accuracy were attained when $l_1=l_3$ and $l_2=l_4$. Since the most convenient places to mount balance weights on the wheel were the top and bottom rims, the two pivot points (the flexures that take the place of trunnions) were set at the heights of the top and bottom rims.

Tuning the Balancing Machine.—With the spacecraft installed and the pivot points set, the tuning springs and weights were adjusted so that the natural frequency of the rocking motion about each pivot was within one percent of the spindle rotation frequency. Since the operation was moderately damped, closer tuning was not necessary.

Balancing the Wheel.—

1. The sail was tied to the balancing machine yoke so that it could not rotate.

2. Using the upper pivot, the magnitude and phase of the rocking torque was measured. A weight of the proper size was taped on the wheel at the proper position on the top rim of the wheel to reduce the rocking torque. This step was repeated with trimming weights until a five-gram change caused overshoot.

3. Using the lower pivot, Step (2) was repeated on the bottom rim. The bottom weight was then removed and weighed. A weight 12 times as great was then put in the place of the original weight.

4. Steps (2) and (3) were repeated until the wheel was in balance within the effect of a five-gram weight for both pivots. This usually required about three reiterations.

5. The position and size of the two weights was noted and a location where permanent weights could be installed to give the same effect as the two temporary weights was found. (Usually the best place was the inside of the

wheel compartment covers.) The permanent weights were fabricated and installed.

6. Steps (2) through (5) were repeated until the wheel was in balance within the effect of a five-gram weight with only permanent weights installed. This usually required two reiterations, and the wheel was then balanced.

Balancing the Sail.—With the sail locked, the wheel balancing steps were repeated with weights being placed on the sail rather than on the wheel. More reiterations were required because of the difficulty in putting weights where they were required. This was continued until the wheel and sail together were balanced within the effect of a ten-gram weight. The spacecraft was then in balance.

Checking Balance in Stowed Condition

With the sail locked and the arms down, a check was made to be sure that the spacecraft was in balance in the launch condition within effect of a twenty-gram weight.

Moments of Inertia

Spacecraft moments of inertia were measured to provide numerical data for use in tuning the nutation damper, the setting of the spin-up circuit parameters and the spin and pitch gas jet nozzle ON times.

The measurements were made by suspending the entire spacecraft from a point and measuring pendulum motion periods.

Spin-Axis Moment.—Figure 9-12 shows the experimental setup. The spacecraft was suspended from three vertical wires of length, s , attached at a horizontal distance, r , from the center of gravity. The spacecraft was caused to oscillate as a torsional pendulum, and the frequency of oscillation, ω , was measured. For small amplitudes of oscillation, the moment of inertia is

$$I = \frac{mgr^2}{s\omega^2} \quad (9.10)$$

where m is the mass of the spacecraft and g is the local value of gravitational acceleration. A small correction was made for the mass and moment of inertia of the wires.

With the sail locked to the wheel, the moment was measured with arms out and with

arms stowed. With the sail unlocked so that it could not oscillate, the moment of the wheel was measured with the arms extended. The effect of gas in the bottles was calculated.

Location of Center of Gravity.—As shown in Figure 9-13, the spacecraft was attached to a bracket and suspended by a single wire. The position of the wire was adjusted until the spin axis of the spacecraft was horizontal. This position was measured.

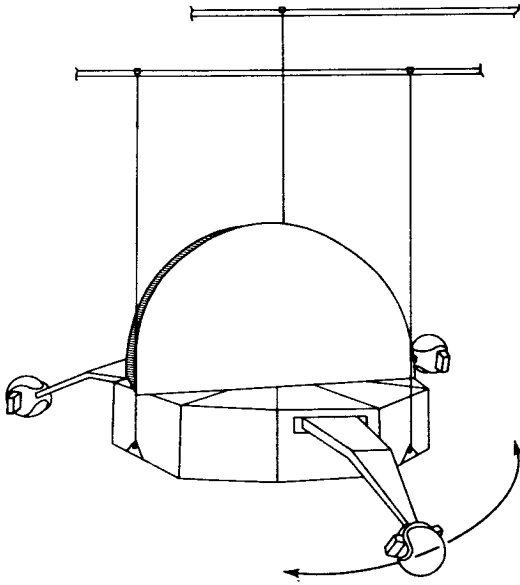


FIGURE 9-12.—Experimental setup for spin moment determination.

The mass and c.g. location of the bracket and the mass of the spacecraft were known, so the c.g. location of the spacecraft could be calculated.

This measurement was made with arms extended and with arms stowed. The effect of gas in the bottles was calculated.

Transverse Moments.—The single wire used in the c.g. determination was replaced by two wires (see Figure 9-14) spaced equally from the combined c.g. of the spacecraft and bracket. The spacecraft was swung as a torsional pendulum, and Equation (9.10) was used to determine the moment of inertia of the assembly about a vertical axis through the combined c.g. From this moment and the known moment of the bracket and the known c.g. locations, the

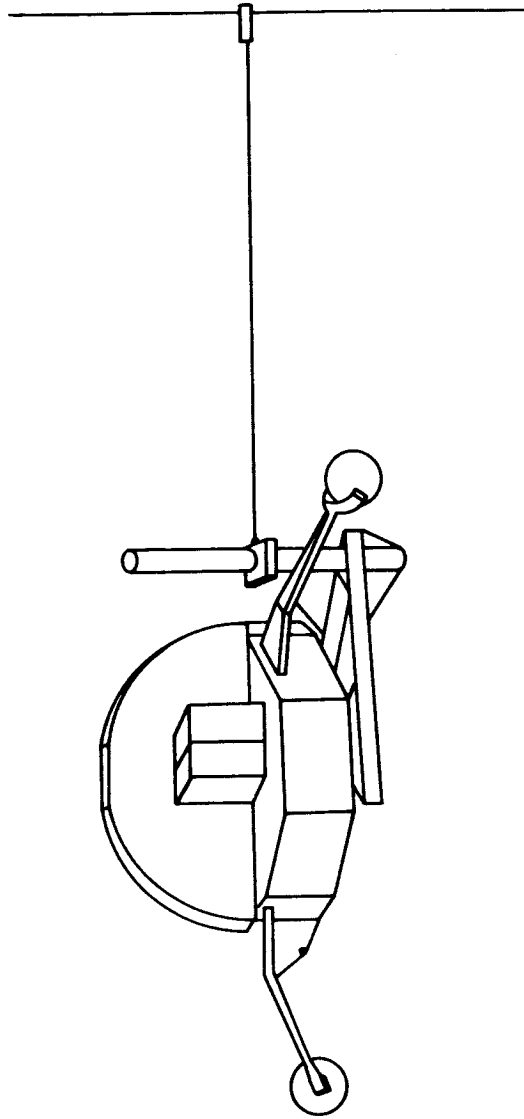


FIGURE 9-13.—Experimental setup for CG location.

transverse moment of the spacecraft was calculated.

This measurement was made with arms extended and with arms stowed, with the sail in a vertical plane and the sail in a horizontal plane, and with each of the three arms toward the floor. The position of the sail made very little difference. The measurements with different arms toward the floor were almost identical. The effect of gas in the bottles was calculated.

Results—

Spin moment, wheel, arms extended:

Bottles full: 24.39 slug-ft²

Bottles empty: 22.29 slug-ft²

Spin moment, wheel and sail, arms extended:

Bottles full: 27.66 slug-ft²

Bottles empty: 25.56 slug-ft²

Spin moment, arms stowed:

Bottles full: 19.71 slug-ft²

Transverse moment, arms stowed:

Bottles full: 15.26 slug-ft²

Transverse moment, arms extended:

Bottles full: 17.72 slug-ft²

Bottles empty: 16.56 slug-ft²

Center of gravity, arms extended:

Bottles full: 12.68 in. above interface

Center of gravity, arms stowed:

Bottles full: 11.85 in. above interface

Environmental Tests

Prototype environmental testing was performed in four phases: acceleration, vibration, thermal, and thermal-vacuum. During the course of these test phases, modified and complete comprehensive systems tests were performed. These tests, which were made only when *go/no-go* information was desired, were referred to as *quick-look* tests.

The quick-look testing determined that the subsystems and instruments were or were not working within some nominal performance.

Acceleration Tests

After the spacecraft was balanced, the prototype spacecraft was tested under steady state acceleration conditions simulating the launch thrust at an amplitude one and one-half times the anticipated launch thrust level. The test was conducted on the Martin-Denver large centrifuge. The photograph of Figure 9-15 shows the prototype spacecraft in the test fixture.

The spacecraft was put in the launch condition and was left that way during the first four minutes of the five-minute test run. The timers were started one minute before the termination of the simulated thrust loading, and the timers were allowed to finish their sequencing operation after the thrust was terminated.

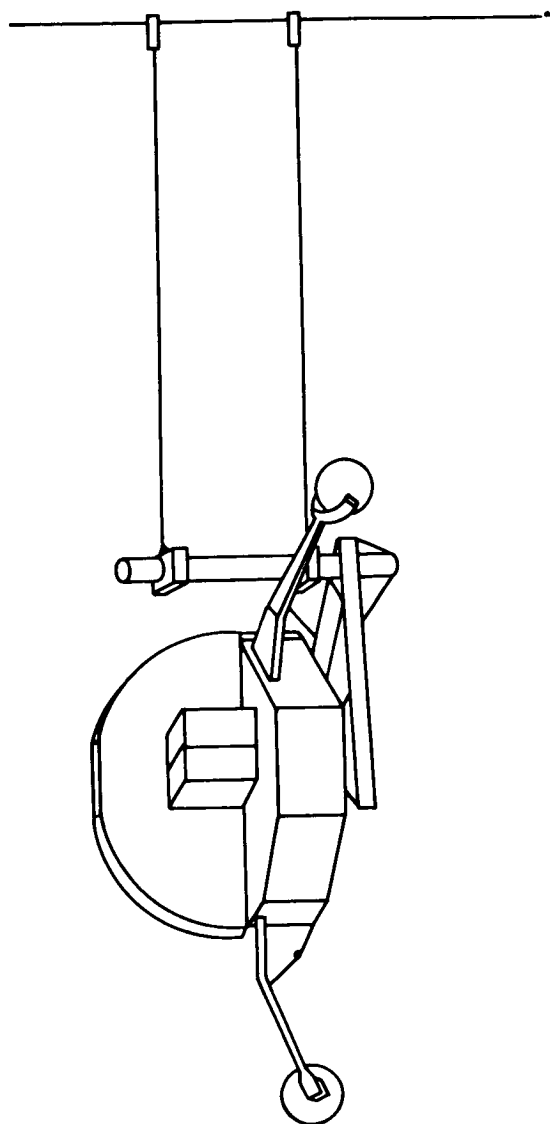


FIGURE 9-14.—Experimental setup for transverse moment determination.

All sequencing operations and all systems were then checked for proper operation.

The spacecraft was then given another comprehensive system test with no failures being found. The balance was checked and found unchanged.

Vibration Tests

Test Equipment.—The prototype model testing was accomplished at Ball Brothers Research Corporation using the equipment listed in Table 9-1.

TEST PROGRAM

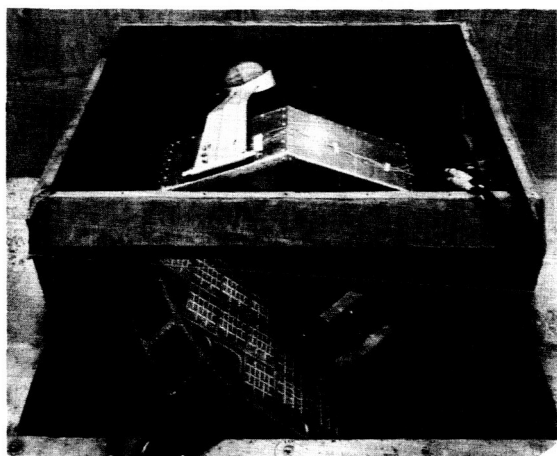


FIGURE 9-15.—Centrifuge test.

Table 9-1

Test Equipment Used for Prototype Testing

Exciter-----	Ling Model 275 rated at 7500 lb peak force.
Power supply---	Ling Model PP 40/60, 40 KVA amplifier.
Control-----	Ling Model R1007 sine and random control console.
Tape facility---	Ampex Model FR-1000, 7-channel FM record with 2-channel FM playback.
Transducers----	Endevco Model 2221C accelerometers. CEC Model 4102A velocity pickups. Endevco Model 2106 force gauges.
Associated transducer equipment.	Endevco Model 2216 amplifiers. Endevco Model 2622 power supplies. Mosely Model 58 and Model 2D x-y plotters. Hewlett-Packard Model 302A wave analyzer. Mosley Model 60B logarithmic converters. Mosley Model 80 strip-chart recorder.
Exciter Controller	B and K Model 1016 servo controller.

Testing Specifications.—The specified tests to be accomplished and the specifications are as follows.

Sinusoidal Test.—

Thrust Axis Specification

Frequency (cps)	Acceleration (g)	Velocity (in/sec)
5-50	2.3	-----
50-500	-----	2.6
500-2000	21.0	-----
2000-3000	54.0	-----

Transverse Axis Specification

Frequency (cps)	Acceleration (g)
5-50	0.9
50-500	2.1
500-2000	4.2
2000-3000	17.0

Random Motion Thrust and Transverse Specification.—

Vibration Axis	Frequency Range (cps)	Test Duration (min.)	PSD Level (g ² /cps)	Appr. Accel. (g-rms)
Thrust (Z axis)	20-510	4	0.07	*5.85
	510-2000	4	0.07	*10.40
		Total	8	
Transverse (X axis) and (Y axis)	20-510	4	0.07	5.85
	510-2000	4	0.07	10.40
		Total	8 min.	(each axis)

*Within amplitude limit of vibration generator.

Test Arrangements

The test arrangements for the various configurations are described below.

Thrust Axis Tests.—The prototype spacecraft was clamped with the actual payload third-stage interface retaining clamp to a test fixture having the proper matching interface. The clamp bolts were torqued to the proper level for the actual explosive bolt interface clamps as specified by the Douglas design specification. The test fixture was bolted to the exciter table such that the input forces were applied through the test fixture to the spacecraft. The test fixture consisted of a 3-inch plate which bolted to the exciter, an aluminum casting resembling the frustrum of a cone tapering from 20 inches in diameter at the bottom to 10 inches in diameter at the top, and a machined adapter flange which provided the interface fitting. For the effective weight tests described below, force gauges were inserted between the adapter flange and the casting.

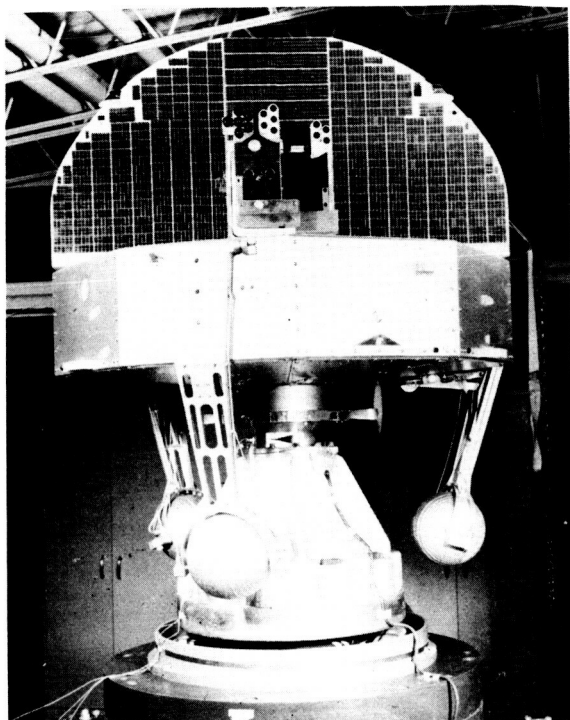


FIGURE 9-16.—Thrust axis vibration test.

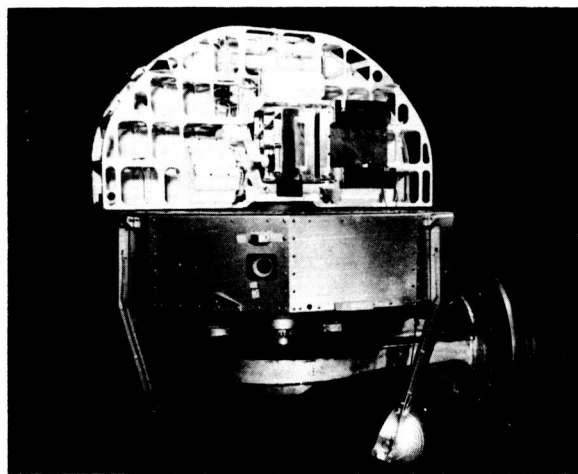


FIGURE 9-17.—Thrust axis vibration test (closeups of spacecraft).

Figures 9-16 and 9-17 are photographs of the spacecraft installed on the vibration exciter with the test fixture interface for thrust axis vibration.

Transverse Axis Tests.—The spacecraft was clamped to a cast magnesium sliding fixture.

The fixture was driven in a sliding manner on two granite surface blocks oriented such that the spacecraft arms could be positioned between them. The slide fixture was designed to allow the centerline of the exciter to be as close as possible to the center of gravity of the spacecraft. For the effective weight tests, the slide fixture was the same except that it was made from two pieces which could be bolted together with force gauges between the two pieces. Figures 9-18 and 9-19 illustrate the test arrangement and show the interface fixture.

Results

A description of each test is included in the order in which the testing was accomplished and the results of these tests are as follows:

Transverse X-Axis Sine Input (Low-Level Trial Run).—The test was run at $\frac{1}{2}$ g input from 5 to 50 cps and at 1 g input from 50–300 cps with 100 psi pressure in gas bottles. No failures occurred.

Transverse Y-Axis Sine Input (Low-Level Trial Run).—Same test as the transverse X-axis sine input above. No failures occurred.

Thrust Axis Sine Input (Low-Level Trial Run).—The test was run at 1 g input from 21 to 3000 cps with the gas bottles pressurized to 180 psi, 30 psi of which was helium. No failures occurred and a leak test afterwards indicated no bottle leakage.

Thrust Axis Sine Input (Specification Level—First Run).—The test was run at the specified input level with the gas bottles at 100 psi. Some of the spacecraft weight was supported by rubber cords to prevent the exciter table from bottoming due to the 1 g static load of the spacecraft. This test loosened many screws in the sail structure mountings. A pin connection at a connector was opened. A photomultiplier tube was broken in the Goddard wheel experiment, and the high-voltage supply to the tube was shorted. The Goddard pointed instrument carriage pin and a lead at a terminal board were broken. Two photomultipliers in the Rochester wheel experiment were broken, and one of the Ames surface emissivity plates was loosened.

The damage was repaired before vibration testing was continued.

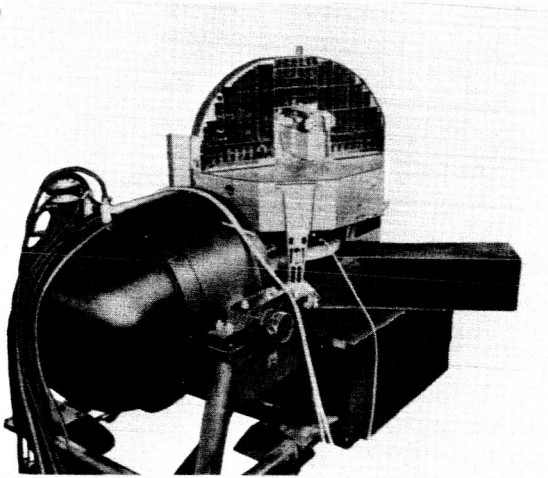


FIGURE 9-18.—Transverse axis vibration test.

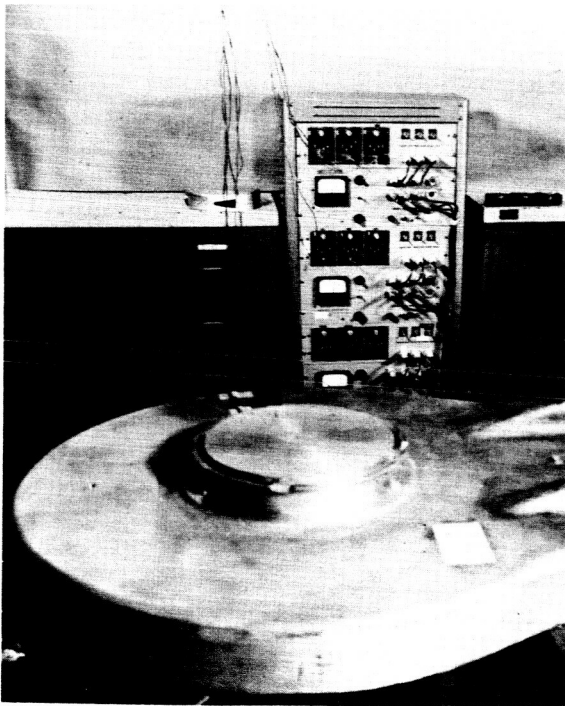


FIGURE 9-19.—Transverse axis vibration fixture and accelerometer amplifiers.

Transverse X-Axis Sine Input (Specification Level—First Run).—The test was run with specified input levels with 100 psi gas bottle pressure. No failures occurred.

Transverse Y-Axis Sine Input (Specification Level—First Run).—The test was run at speci-

fied input levels with 100 psi bottle pressure. Gas bottle leak tests after the test indicated no leakage. Balance checks after the test indicated no change in balance.

Transverse X-Axis Sine Input (Specification Level—Second Run).—This test was run with specified input levels with 3000 psi gas bottle pressure. No failures occurred.

Transverse Y-Axis Sine Input (Specification Level—Second Run).—The test was run with 3000 psi gas bottle pressure. No failures occurred.

Transverse Y-Axis Random Input.—Random testing involved a procedure known as *equalization*. The problem involved with a random input is that of applying equal energy per frequency bandwidth to a device having many resonant motions. If a noise generator random signal of equal energy distribution were applied directly to the spacecraft, the power distribution per bandwidth would not be uniform because of load resonances. Thus, the output amplitude from a noise generator must be tailored to conform to the reciprocal of the exciter and spacecraft response. To accomplish this adjustment of the input signal, the process of equalization was necessary.

Equalization is accomplished in the following manner: A low-level sinusoidal sweep is run to record the load response. By means of peak and notch filtering, the noise generator output is adjusted to match the reciprocal of the measured response. A low-level random burst then is applied through the filter system and the load response is recorded on tape. A tape loop is cut and run through an analyzer. Any necessary small adjustments can then be made without subjecting the load to continuous random vibration. A full-level random burst is then run.

Some difference is usually found between the low and high level random input response of a multi-resonant structure. This may be due to nonlinear resonances that are present under high-level input conditions. A high-level response could appear at a filter notch frequency, and the combination could result in an energy concentration at that frequency which could be damaging to the specimen under test. For this reason, the high-level burst is run for a short

duration and is recorded. A tape loop is cut and run through the analyzer and further filtering adjustments are made. At this point, the specified level random test may begin. Usually minor adjustments are made during the first few seconds of the actual test run due to the fact that the tape loop may not contain enough time for proper statistical averaging of the high-level response. These adjustments must be only minor due to the complex overlapping nature of the filtering networks.

The transverse Y-axis random test was accomplished after the equalization discussed above. The test was run with full spin and pitch gas bottle pressure. Checkout revealed no failures.

Thrust Axis Sine Input (Second Run).—The test was run with full bottle pressure at specification level. The Goddard pointed experiment carriage pin was broken. The test had to be repeated.

Thrust Axis Random Input.—Equalization was accomplished and the test was run at specified input levels with the gas bottles fully pressurized. No failures occurred.

Thrust Axis Sine Input (Third Run).—This full-level test was repeated since some equipment had not yet survived the test. This time no failures were noted. Leak tests indicated no bottle leakage.

Thrust Axis Effective Weight.—Because some of the third-stage engines used in the Thor-Delta vehicle configuration had been shown in test firings to resonate during the burning time at a frequency between 550 and 650 cps, it was considered necessary to test the spacecraft payload extensively at this bandwidth. In order to do this, the effective weight of the spacecraft had to be known.

Effective weight is that quantity derived by dividing the measured force input to the spacecraft by the resulting measured acceleration produced. To do this, it is necessary to separate into components the reactive and the direct forces, and accelerations as measured for both the fixture and the composite spacecraft and fixture, and to measure the phase angles involved. As a simplified view of the problem, what is required is the knowledge of

the effect upon the spacecraft of the actual force input expected.

Effective weight measurements were made by measuring the phase and amplitude of the forces and accelerations involved. This measurement was made only at 550 cps because one of the force gauges became inoperative after the measurement had been made at that frequency. The effective weight measured was 27 lb.

Transverse X-Axis Effective Weight.—The effective weight was somewhat variable over the 550 to 650 cps bandwidth (about 20 percent variation). The average was about 21 lb.

Transverse Y-Axis Effective Weight.—Since the highest effective weight measurement in the X-axis direction was measured at 550 cps, the Y-axis was measured only at that frequency. The weight was about 27 lb.

Resonant Burning Dwell Tests.—After effective weights had been determined for each axis, a slow sweep was run from 550 to 650 cps in 30 seconds at an input level of 600 lb, divided by the resultant effective weight for the thrust axis, and 100 lb divided by the resultant effective weight for the transverse axis. The 600 lb and 100 lb figures represent the estimated thrust axis and transverse axes components of the third-stage engine thrust force, respectively.

The problems that were encountered with both the spacecraft equipment and the scientific instruments during the prototype vibration testing serve to point out the importance of this testing. The input levels at which vibration testing should be accomplished are argumentative, but the necessity for the testing is certainly obvious in the light of the test results during this program.

A complete comprehensive system test was performed after the complete vibration test.

Thermal Tests

Thermal tests of the prototype were made in the thermal-vacuum chamber, and rough vacuum was used to expedite thermal changes. The chamber is 10 ft in diameter. Figures 9-20, 9-21, 9-22, and 9-23 show the 10-ft-chamber and the operating equipment for the chamber.

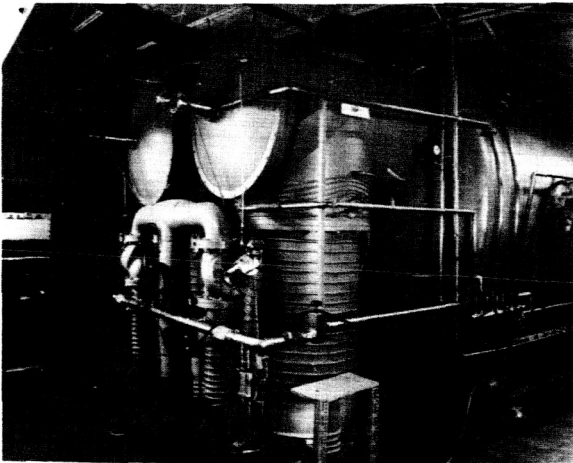


FIGURE 9-20.—Large thermal vacuum chamber with 32-inch dual-diffusion high vacuum pumping system.

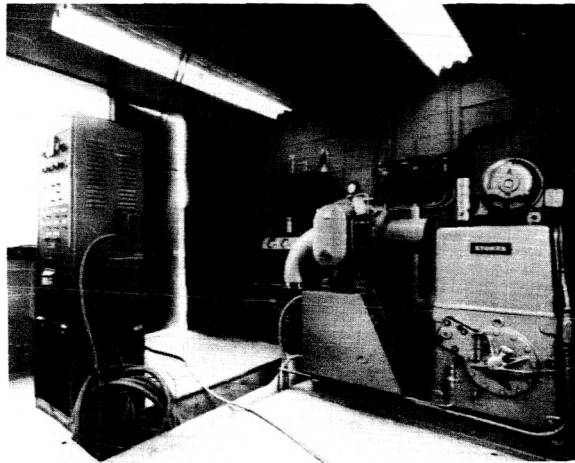


FIGURE 9-22.—Roughing pump system.

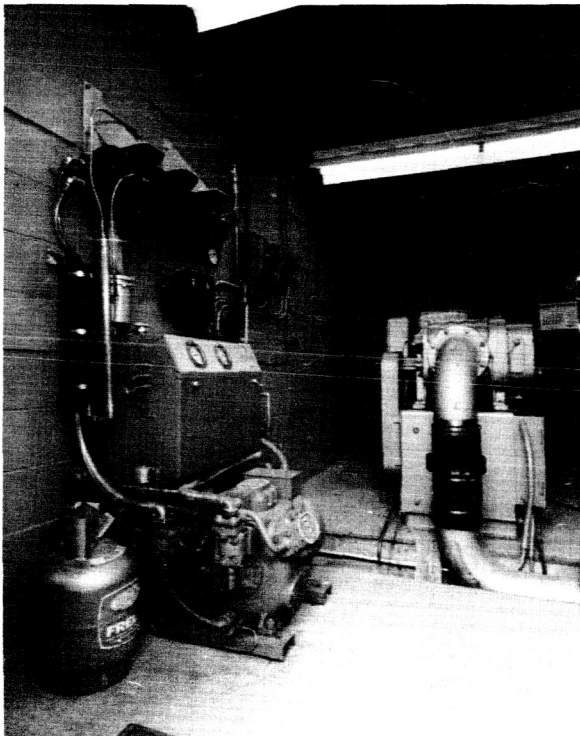


FIGURE 9-21.—Freon cooling system.

In Figure 9-23 the spin table can be seen supporting the spacecraft in the vacuum chamber.

The first temperature half-cycle was made at 35°C for six hours. Four hours were required to reach temperature stabilization. This por-

tion of the test was a temperature soak in which the spacecraft was nonoperative.

The second temperature half-cycle was made at -10°C for six hours. Thirteen hours were required to stabilize at this temperature. This portion of the test was also a temperature soak with the spacecraft nonoperative.

The third half-cycle involved six hours to reach stabilization at 35°C. After stabilization, six hours were required to perform spacecraft operational sequencing. All systems were operated with the exception of the experiments which were limited to operation in a high vacuum. The artificial sun, directed through the front window of the chamber, and smaller light sources inside the chamber were used in the operation of the spacecraft. Electrical leads through the chamber facilitated operation and instrumentation of the spacecraft.

The fourth half-cycle involved ten hours to reach stabilization at -10°C. One hour after stabilization the test was stopped for investigation of problems in the spacecraft. When the test was resumed the temperature was again stabilized at -10°C. For eight hours after stabilization, the spacecraft was subjected to a complete operational test.

In summary, this thermal test sequence resulted in a total of 12 hours soak time at 35 degrees and a total of 15 hours at -10°C. After the thermal tests, a complete comprehensive system test was performed.

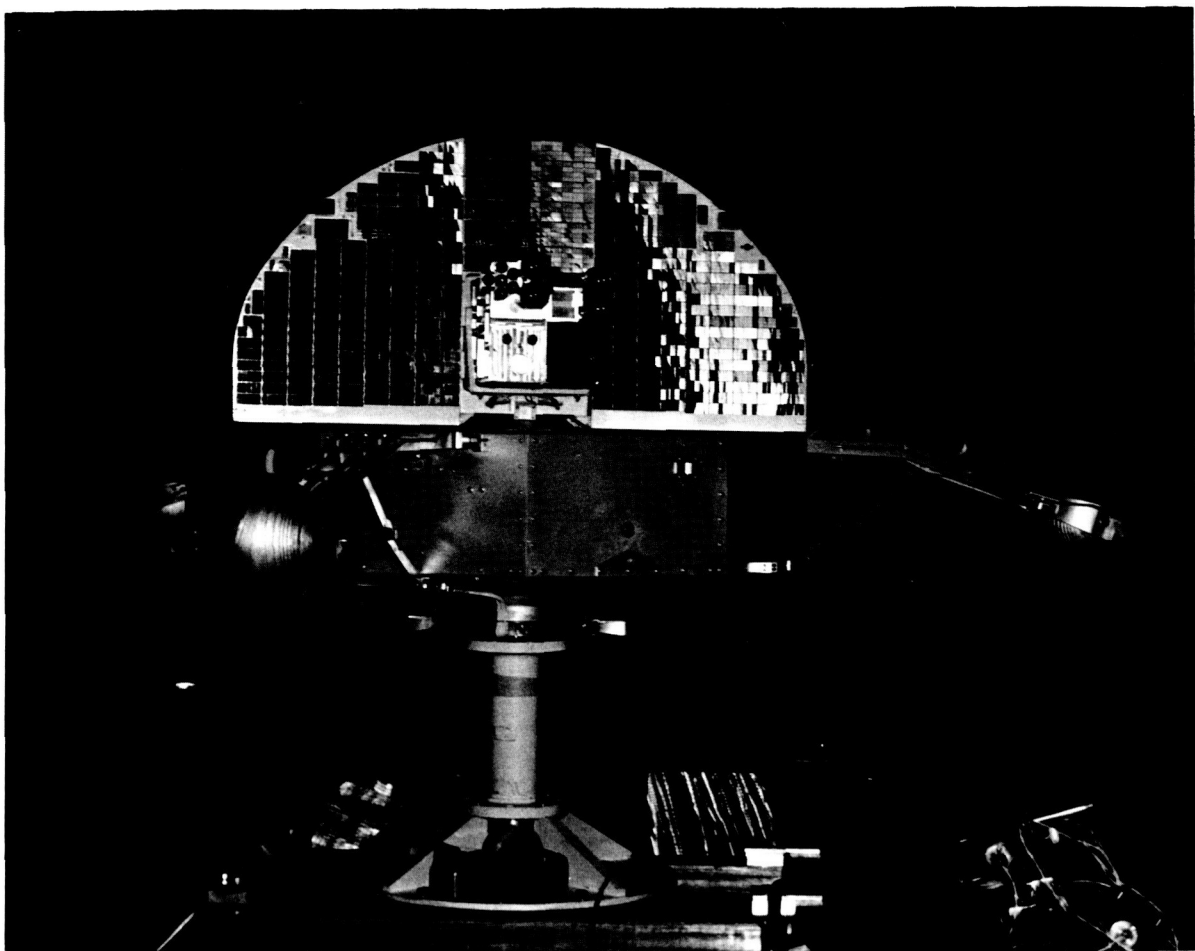


FIGURE 9-23.—Spacecraft on spin table in vacuum chamber.

Thermal-Vacuum Tests

Thermal-vacuum testing of the prototype spacecraft was conducted in the vacuum chamber. This chamber has three windows to permit observation. One of these windows was used to allow operation with the external artificial sun.

Electrical lead-throughs were provided to operate the spacecraft, monitor spacecraft operation, control the pitch gas system, control the spin gas system, monitor pitch readout, and to monitor temperatures.

The vacuum chamber had a full-opening door allowing convenient installation and removal of the spacecraft. A telemetry antenna was placed inside the chamber to command spacecraft operation and to receive telemetry

signals. This antenna was connected to coaxial lead-throughs in the chamber.

The following documents were applicable to the thermal-vacuum test:

1. *Vacuum Test Records* 61-23 and 62-24.
2. *Test Procedure—Satellite Payload Thermal-Vacuum Test of NASA S-16A OSO.*
3. *Thermal-Vacuum Test Specification—S-16 OSO Satellite, S61-6.*
4. *Test Specification No. S61-1.*
5. *Test Procedure No. TP61-1.*
6. *Test Procedure No. TP61-11.*

The spacecraft was maintained in an environment of 10^{-5} mm Hg pressure throughout the test except when the gas jets were fired and when temperature changes were made.

Prototype spacecraft test specifications required a minimum test duration of seven days

at 35° C and a minimum test duration of five days at -10°C.

Throughout the test, chamber temperature and pressure were recorded hourly and during any temperature and pressure changes that occurred between hourly recordings.

The temperature sensor mounted on the spacecraft wheel hub was used to maintain the specified spacecraft temperature throughout the thermal-vacuum test. The spacecraft spin table temperature and the chamber thermal liner temperatures were also monitored.

The first attempt to run the thermal-vacuum test started with the high-temperature cycle on July 26, 1961, and was stopped on July 28. This test ran for 47 hours at 35°C, and was halted to repair spacecraft instruments. This attempt was not included as part of the required seven-day test.

The second attempt was started July 31. This attempt was stopped to correct problems occurring in the spin jet system. The total elapsed time of this attempt was 22 hours. None of this time was applied to the seven-day test requirement.

The final and successful attempt to run this test was started with a pressure of 1×10^{-5} mm Hg and a temperature of 35°C on August 2, and ran successfully until complete for a duration of nine days and three hours.

Throughout this test run, the prototype spacecraft was operated continuously in simulated orbiting day-night cycles. Real-time data was transmitted to the telemetry ground station allowing records to be made of all real-time spacecraft output channels. Complete playback records were made at intervals approximating the orbit intervals expected. The spacecraft day-night operation was simulated by using a timer to operate the day-night switching circuitry within the spacecraft.

By controlling small lamps inside the chamber, the pitch control system, the spin-up system, and the pitch readout system were tested periodically. The spacecraft was operated throughout the thermal-vacuum test according to the OSO I spacecraft thermal-vacuum test procedure and the thermal-vacuum test specifications.

Upon completion of the high-temperature

portion of the thermal-vacuum test, the cold cycle was started. The vacuum chamber pressure was raised to 30 mm Hg to accelerate the temperature change from 35° to -10°C. This temperature change required eight hours. Temperature of -10°C and pressure of 1×10^{-5} mm Hg was achieved on August 12. The test was successfully run at -10°C and pressures lower than 10^{-5} mm Hg for a total duration of 119 hours. During this time, the prototype spacecraft was successfully operated in the same manner previously described.

Upon completion of thermal-vacuum testing, a complete and final comprehensive systems test was made, and the prototype spacecraft was completely qualified.

FLIGHT MODEL TESTING

Subsystems Tests

All of the flight model spacecraft subsystems and major components were tested to the component environmental test specification. This specification required that the components or subsystems be successfully operated before, during (in most cases), and after three-axis vibration tests and high and low thermal-vacuum tests.

Basic Electronics Packages

The basic electronics for the control systems were packaged in three separate boxes. (See Figures 4-14 and 4-15 for two of these.) These are as follows:

Servo Amplifier Box.—This package, mounted on the sail structure, consisted of the servo preamplifiers, lead networks, ac amplifiers, modulators and demodulators, voltage regulators required for the circuits, and positioning readout amplifiers.

Power Amplifier Box.—This package, also mounted on the sail structure, contained the pulse-width modulators, power amplifiers, pitch control electronics, and necessary voltage regulator circuits.

Wheel Electronics Box.—This package, mounted in the wheel, contained the spin control electronics, undervoltage circuitry, day turn-on circuitry, spin-rate monitoring circuit, and necessary voltage regulators.

Each of these electronics packages was tested to the flight model environmental test specifications which are defined as follows:

Sinusoidal Sweep Vibration Tests.—The tests were conducted by sweeping the applied frequency from the lowest to the highest frequency once for each of three mutually perpendicular axes. The time rate of change of frequency was four octaves per minute making each sweep about 2.5 minutes duration for a total time of 7.5 minutes on each subassembly.

The specifications were as follows:

Vibration axis	Frequency (cps)	Acceleration (peak g)
Thrust	5-50	1.5
	50-500	7.1
	500-2000	14
	2000-3000	36
Lateral (each of two axes perpendicular to thrust axis and perpendicular to each other).	5-50	0.6
	50-500	1.4
	500-2000	2.8
	2000-3000	11.3

The subassemblies had to operate properly before, during, and after the vibration tests to be considered qualified.

Thermal-Vacuum Tests.—These tests were conducted as follows. The subassemblies were brought to an equilibrium temperature of -20°C at a vacuum of at least 10^{-5} mm Hg and were left at these conditions for 24 hours. At the completion of this cold temperature soak, the temperature was raised to $+50^{\circ}\text{C}$ with the vacuum maintained, and was held at $+50^{\circ}\text{C}$ for 24 hours. The subassemblies were required to operate properly before, during, and after the thermal-vacuum tests to be considered qualified.

Detector Subassemblies

The coarse pointing eyes, fine pointing eyes, fine readout eyes, spin eyes, pitch control eye, and the pitch readout eye were tested to the same vibration and thermal-vacuum test specifications as the electronic packages and were further tested to a temperature shock specification. The reason for this was that, in general, the detectors are relatively unprotected from direct solar radiation and thus are subject to rapid temperature changes as the spacecraft moves from orbit night to orbit day and vice

versa. These subassemblies were allowed to reach an equilibrium temperature at -20°C in a cold chamber and then moved directly to a hot chamber at $+80^{\circ}\text{C}$ and allowed to come to equilibrium. The procedure was repeated three times. Angular response curves were run before and after the temperature cycling to determine that no mechanical or optical changes or damage had occurred. Response curves were also run before and after the vibration and thermal-vacuum tests.

Nutation Damper

Both the flight model and flight model spare nutation dampers were assembled and tested in the following manner:

The nutation damper bellows were subjected to a helium leak-rate test. No detectable leaks were found in the bellows used in the flight and flight spare nutation dampers. After the bellows were leak-tested, the nutation damper upper housing, end bells, and bellows were assembled. The entire upper housing was helium leak-tested after assembly of the end bells and bellows.

The nutation damper pendulum frequency was calculated using the measured spacecraft moments of inertia and the anticipated spin rate. The equations for the pendulum frequency are given in Chapter 2. The moments of inertia used for calculating the pendulum frequency were:

Axial moment of inertia, $I = 24.54$ slug-ft²

Transverse moment of inertia, $I_a = 17.30$ slug-ft²

Transverse moment of inertia, $I_b = 16.15$ slug-ft²

The spin rate used for calculating the pendulum frequency was:

π radians per second, or $\frac{1}{2}$ rotation per second.

The flight model nutation damper specifications were:

Radius of bob, 0.926 inches

Bob weight, 0.845 lbs

Diameter of wire, 0.024 inches

Pendulum wire length, 3.08 inches

Calculated tuning frequency, 2.04 cps

TEST PROGRAM

The flight spare nutation damper specifications were:

- Radius of bob, 0.926 inches
- Bob weight, 0.825 lb
- Diameter of wire, 0.024 inches
- Pendulum wire length, 3.12 inches
- Calculated tuning frequency, 2.03 cps

Calibration of the pendulum was accomplished by suspending the pendulum from a special optics bench fixture and adjusting the pendulum length until the natural frequency of the pendulum agreed with the calculated frequency. This frequency adjustment was made by counting the frequency of interruption of a light beam by the pendulum with a photomultiplier tube. The output of the photomultiplier was compared with a standard frequency on a two-channel strip-chart recorder. Comparisons and adjustments were made until the bob frequency agreed with the calculated frequency. When the bob was calibrated, the mechanical adjustment was sealed with Locktite to prevent any frequency change under vibration or temperature extremes.

Adjustment of the pendulum bob caging was accomplished by screwing the bob wire into the upper housing until the bob was nearly restricted from swinging by the brass caging ring. When

this adjustment was made, the top pendulum screw was locked with a jam screw and sealed with C-7 epoxy.

The bottom housing was oriented and assembled complete with caging piston and spring, but without the squib block. The bottom housing was then bolted to the upper housing. Locktite was used to secure the bolts.

The squib block was assembled and two capstan nuts were adjusted for proper caging position. The squib block was mounted on the damper and the damper was caged.

Prior to filling, the damper was carefully weighed. After filling, the damper was weighed again to assure that the damper was properly filled with oil, and the dampers were leak-tested once again with a helium leak detector.

The nutation dampers were then connected to the filling system, illustrated by the schematic of Figure 9-24 and vacuum-filled with Dow Corning 200, 30-centistoke silicone fluid. This filling was done with hot, air-free oil in a vacuum to avoid air being trapped inside the damper. The damper oil was then cooled to -10°C to complete the filling. After the dampers were filled, the filling system was valved off. The dampers were then heated to 85°C to cause a pressure rise which was absorbed by the bellows.

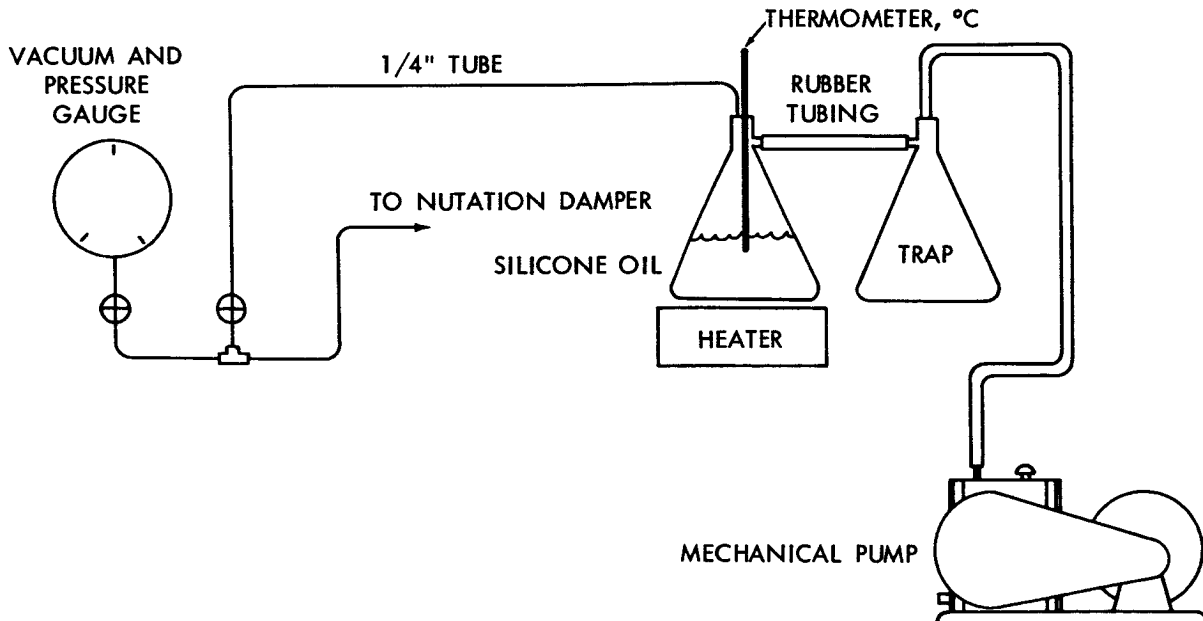


FIGURE 9-24.—Nutation filling system schematic.

The pressure was less than 17 psi. If the pressure in either damper had risen to over 20 psi, the bellows would have been judged defective. Without the bellows the damper would have burst with a slight temperature rise over the -10°C filling temperature. After the pressure test the dampers were again brought to a temperature of -10°C and sealed. Each temperature cycle was one hour in duration.

Vibration test inputs for the nutation dampers were programmed to correspond to data taken from the azimuth casting accelerometer during the transverse axes prototype vibration test. The duration of the tests were 5 minutes in each axis. The thrust axis vibration test was of 8 minutes duration. The frequencies and "g" levels are listed below:

<i>Frequency (cps)</i>	<i>Acceleration (peak g)</i>
5-40	2.5
40-80	17.5
80-100	8.5
100-300	7.05
300-3000	11.25

After the vibration tests were completed, the nutation dampers were tested to determine that the caging mechanism had held the penulum bob securely. The caging release squib actuators were then fired to check the proper operation of the caging release mechanism. The squib block was then removed, cleaned, and replaced. The nutation dampers were painted with the spacecraft white paint used on most of the upper structure which faced away from the sun in orbit.

Thermal-vacuum tests were run with the dampers at pressures lower than 10^{-5} mm Hg and temperatures of 0°C and 80°C . Each temperature was maintained for 24 hours. No silicone fluid leakage occurred during the thermal-vacuum tests.

Comprehensive System Test

The flight model spacecraft was tested in accordance with the same format documented test procedure used for the prototype system tests. Prior to performing the various tests necessary to qualify the flight model, it was necessary to balance the spacecraft. The flight model was balanced statically and dynamically

using the same method as was used on the prototype spacecraft.

Vibration Tests

The flight model spacecraft was subjected to input levels 50 percent less than those used for the prototype spacecraft, and the test duration was one-half of that used for the prototype model. The effective weights measured on the prototype were used for the flight model resonant-burning dwell tests. The flight model vibration tests went well. The first test was repeated when a lead wire shorted. The lead wire had been too tightly routed around a sharp corner of a grounded component retainer clip. The insulation had cold flowed, probably accelerated by vibration, and shorted. The lead wire was re-routed and the complete test series was run with no failures.

The general concept of environmental testing is that the anticipated environment should be realistically simulated and that the level of simulation be increased by some safety factor margin in order to gain confidence that any equipment surviving the simulated conditions will survive in the actual environment.

The general impression of the series of vibration tests performed at Ball Brothers is that the test specifications were unduly rigorous. Since as much time and money can be wasted in testing to unreasonably rigorous specifications as are wasted in testing to specifications that are too mild, it is recommended that data be gathered from future flights with a view toward providing more realistic test specifications.

Thermal-Vacuum Test

This test was performed to assure that the spacecraft would operate in the orbital environment. Flight model test specifications required a minimum test duration of three days at 25°C and a minimum test duration of three days at 0°C . Test specifications also required a vacuum of 10^{-5} mm Hg, or less.

Throughout the test the spacecraft hub temperature, the spin table temperature, and the chamber thermal liner temperature were monitored to provide control data for maintaining the specified temperature of the spacecraft. The control temperature sensor was the

wheel hub sensor. Temperature and pressure readings were recorded at least every hour throughout the test.

During the ten-day test the vacuum chamber pressure varied from 9×10^{-6} mm Hg to 5×10^{-7} mm Hg.

Final preparation for the test began on December 4, 1961. The spacecraft was mounted on the spin table in the thermal-vacuum chamber. The chamber was pumped down to 10^{-5} mm Hg and returned to atmospheric pressure to clean the spacecraft slip rings which were conducting intermittently. Several chamber evacuations were made between December 5, and December 8, while spacecraft problems were being corrected. The high-temperature portion of the test started December 8, at 1730 and ended December 12 at 1200, for a total duration of three days, $18\frac{1}{2}$ hours. On December 13, the cold cycle was started, but a return to atmospheric pressure was necessary to change the bearings on the spin table. The low-temperature portion of the test started again December 12 at 0800 with a pressure of 1×10^{-6} mm Hg and a temperature of 0°C . This portion of the test continued until 1100 on December 18, for a test duration of three days, three hours, and 55 minutes.

On January 12, 1962, a short low-temperature thermal-vacuum test was made to test the spin jet system. This test ran approximately one day.

Throughout the OSO I flight model thermal-vacuum tests the spacecraft was operated continuously in simulated day-night orbiting cycles. Real-time data were recorded, and complete playback records were made. Spacecraft day-night operation was controlled as explained previously. By controlling the small lamps located inside the chamber, the light operated control subsystems were checked out.

Magnetic Compensation Tests

Introduction

Any object which possesses a magnetic dipole moment tends to act like a compass. That is, the earth's magnetic field exerts a torque on it which causes it to tend to line up with the field.

If the OSO I spacecraft possessed a net dipole moment, a magnetic torque would have to be counteracted by torque from the gas jets. Rough calculations† indicated that the gas expenditure would be tolerable if the net dipole moment were less than 500 gauss-cm³. Therefore, it was necessary to determine the moments of the various magnet-containing parts of the spacecraft and to add compensating magnets to reduce the net moment to an acceptable level. These tests were performed on Ball Brothers Magnetotropometer which is described in detail on page 246.

Calculation of Scale Factor

Referring to Figure 9-25, let

- θ = deflection angle
- r = inner radius
- R = outer radius
- $2a$ = length of chain
- W = weight per unit length of chain
- T_0 = Tension in chain at lowest point
- $2b$ = span of chain
- $c = T_0/W$

For a catenary chain with both ends at the same altitude, we have

$$\frac{a}{c} = \sin \left(\frac{b}{c} \right) \quad (9.11)$$

From the figure, we determine that

$$(2b)^2 = r^2 + R^2 - 2rR \cos \theta, \quad b = 1/2(r^2 + R^2 - 2rR \cos \theta)^{1/2} \quad (9.12)$$

A solution of Equations (9.11) and (9.12) for c is desired. This pair of equations cannot be solved analytically. However, b may be evaluated for known values of θ and R and given values of θ . Then, using these values of b and the known value of a , Equation (9.11) and a table of hyperbolic sines may be used to solve for c . Then

$$T_0 = wc = \text{horizontal force at } P \text{ in the direction of the chain} \quad (9.13)$$

†BBRC TN 61-68.

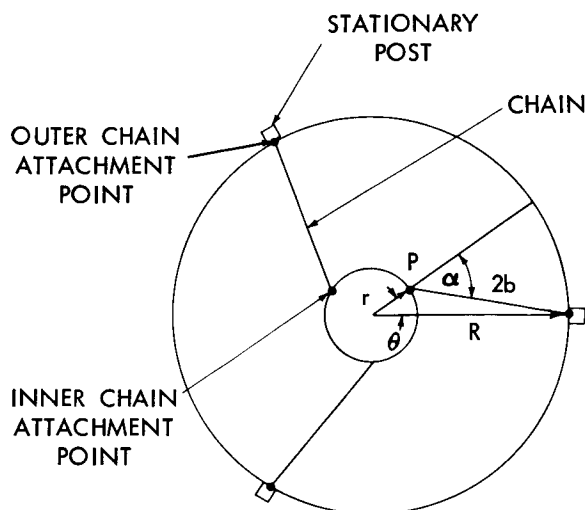


FIGURE 9-25.—Geometry of magnetotropometer.

The tangential force at P is then,

$$T = T_0 \sin \alpha \quad (9.14)$$

Now, since the sine of the interior angle at P is equal to $\sin \alpha$ from the law of sines, we have

$$\frac{\sin \alpha}{R} = \frac{\sin \theta}{2b}$$

or

$$\sin \alpha = \frac{R \sin \theta}{2b} \quad (9.15)$$

Combining Equations (9.13), (9.14), and (9.15) results in

$$T = \frac{wcrR \sin \theta}{2b}$$

The torque exerted on the inner ring by each chain is Tr ; so the total torque exerted by the three chains is

$$L = \frac{3wcrR \sin \theta}{2b} \quad (9.16)$$

For this instrument, the following values are determined:

$$\begin{aligned} r &= 1.07 \text{ cm} \\ R &= 62.50 \text{ cm} \\ 2a &= 65.20 \text{ cm} \\ w &= 18.8 \text{ dyne-cm}^{-1} \end{aligned}$$

Therefore, we have

$$L = 1885 \frac{c}{b} \sin \theta \quad (9.17)$$

When L is calculated for several values of θ , for $\theta < 40^\circ$, L lies within 2 percent of 53.50 (when θ is expressed in degrees).

The outside circumference of the tub, which carries the scale, is 383 cm. This gives 1.063 cm per degree. The scale factor, then, is 50.3 dyne-cm of torque per cm of scale deflection.

Resolution

A lack of repeatability was shown from experience to amount to ± 0.4 cm deflection.

This may have been due to uneven loading, hull warpage, or convection currents. In any case, the small difference in repeatability limited the useful resolution of the magnetotropometer to 20 dyne-cm torque.

Accuracy

The accuracy of the magnetotropometer depended only on the accuracy with which its dimensions were measured. The error in each measurement was estimated, and the error that each could contribute to the magnetotropometer reading at 30° deflection was calculated to be:

Measurement	Error (percent)	Reading error (at 30°) (percent)
r -----	3	2
R -----	0.5	5
$2a$ -----	0.1	1
w -----	0.2	0.2

By combining with these errors the 2 percent error introduced by using a linear scale, and by

Magnetic Field at Magnetotropometer

Area of coil----- $NA=3.99(10^5) \text{ cm}^2$
turns
Current----- $i=0.088 \text{ amperes}$
Scale deflection----- 16.0 cm

Torque..... (16.0) (50.3) = 805
dyne-cm
Torque..... 0.1 NA i H
H = 0.229 gauss

The largest part of the magnetic moment of the spacecraft was due to the magnetically focused photomultiplier in the Goddard pointed x-ray spectrometer. This moment was about 6200 gauss-cm³ in the direction of the pitch axis. Therefore, the first step in compensation was to mount the magnet structure from an identical photomultiplier at one end of the sail, with its moment opposite to that of the spectrometer. All further measurements were made with this compensating magnet considered as part of the spectrograph.

Measurements were made on each instrument which contained a magnet with the following results:

Instrument	Components of magnetic dipole moment		
	Forward	Right	Up
Goddard X-Ray Spectrograph (with compensating magnet)	— 83	358	454 gauss-cm ³
California Neutron Experiment	Spinning	Spinning	— 236
Upper Commutator	646	0	0
Alternate Pointed Experiment Commutator	0	— 646	0
Total	563	— 288	218

The magnitude of the total magnetic dipole moment was 670 gauss-cm³. A permanent magnet with a moment of 610 gauss-cm³ was made, and was mounted on the sail with its orientation opposite to the total moment of the rest of the spacecraft. Considering the errors involved, the net magnetic dipole moment of the spacecraft was probably less than 200 gauss-cm³.

The behavior of OSO I in orbit indicated that the spacecraft had a net magnetic dipole moment of about 800 gauss-cm³ in the direction of the pitch axis. This moment produced a torque which caused the spin rate to vary sinusoidally. The amplitude of the variation was 0.037 rps. Its period was 50 days, the same as the period of the precession of the orbit relative to the solar vector.

PRELAUNCH TESTING

After the completion of the balance checks and a final comprehensive test, the flight model spacecraft, the prototype spacecraft, and all required ground support equipment were shipped to the launch site.

The prototype model spacecraft was placed on the launch vehicle on the launching pad at the Atlantic Missile Range and RF compatibility checks were made. No problems were encountered. The flight model spacecraft was given a final comprehensive operational check-out at the launch site. No transportation damage was found. The flight model spacecraft was then given a preflight checkout and delivered to the Douglas Aircraft Company balancing facility for third-stage and spacecraft assembly balancing. The third stage and spacecraft were then placed on the launch

vehicle and several preflight checks were run to provide assurance that the spacecraft continued to be in a launch-readiness state. No problems were found during this period and after eight days of delay due to vehicle problems, the spacecraft was successfully launched.

RF Compatibility Tests

Prior to the installation of the flight spacecraft and third-stage rocket, a dummy third stage and the prototype spacecraft were installed. The shroud was installed and RF compatibility checks were run. The object of this testing was to insure that the spacecraft/ground station communication loop was working and was secure from other RF interference, and that the spacecraft RF link did not interfere with other RF links in use during the launch activities. The payload compatibility checks uncovered no problems in either area. The testing was done in three phases.

1. With the tower in place with the spacecraft in both launch and orbit conditions.
2. With the tower in place with the payload in launch condition and with the shroud in place.
3. With the entire vehicle and spacecraft system in launch condition with the tower removed.

Preflight Tests

Preflight testing accomplished on the flight spacecraft was to provide assurance of continued flight readiness. The preflight testing was somewhat more rigorous than the quick-look checks made during environmental testing, but less rigorous than the comprehensive tests. Input simulation of radiation to be measured or detected together with the proper output response of the systems was the primary objective of this test procedure.

No problems were encountered during any of the preflight checkouts run on the flight spacecraft. The preflight checkout was written into the launch vehicle system countdown activities. The test equipment required was the spacecraft main control console, the set of attaching instrument checkout fixtures, and the several radiation sources used for instrument checks.

Operational Fly-Over

Prior to the OSO I launch, an operational test was performed to determine the compatibility of the spacecraft communications system with the Fort Myers tracking mission and the data receiving mission. This ground station is a typical Minitrack network station. The testing was performed by carrying the OSO I structural model extended below a helicopter as shown in Figure 9-26. During this fly-over, the spacecraft was transmitting normally as though in orbit and was capable of receiving command signals from the ground station. The spacecraft radiated and received signals which were attenuated by a 50-db pad to compensate for the 300:1 difference in the distances involved.

Another purpose of the fly-over test was to permit the receiving station personnel to become familiar with the signal format and the command system operation. The test also determined the extent of the problems which might occur in tracking the modulated carrier of the spacecraft.

This operational flight test at Fort Myers

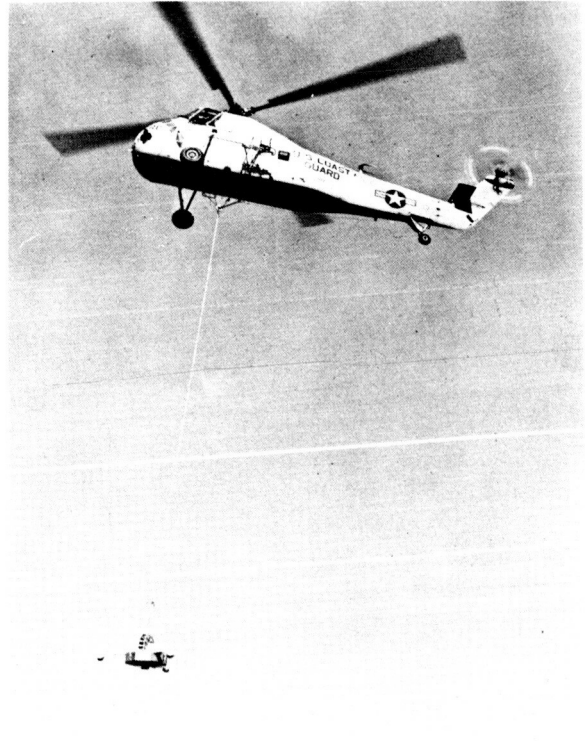


FIGURE 9-26.—Operational fly-over.

verified the compatibility of the ground station equipment and the spacecraft equipment.

MISCELLANEOUS TESTS

Component Testing

Components were chosen on the basis of highest quality, commercial availability, and reliability proven by past history under rocket and spaceflight environmental conditions. An incoming quality control inspection program was instituted to measure the important electrical and mechanical parameters to assure that only those components meeting the purchase specifications would be accepted.

Many components used in the control and communications systems had to be of higher quality than was guaranteed by the routine incoming inspection tests. Therefore, a selection criterion was established and only components that had measured parameters near the average for that component were selected. Furthermore, those components handling large amounts of power were required to meet parameter specifications after being power-aged.

Each component test specification was derived from a knowledge of the importance of each parameter in the actual circuit application. All component selection was accomplished on the basis of the requirements of the individual system application in addition to the basic component quality requirement.

Thermal Stability Tests

The temperature of the spacecraft is dependent on the radiation heat balance between the spacecraft and its surroundings and on the heat dissipated in the craft by such things as electronic gear. The rate of change of spacecraft temperature is described by Equation (9.18):

$$\frac{dT_{\oplus}}{d\tau} = \frac{q_i - q_0}{mc} \quad (9.18)$$

where

$T_{\oplus}$ = temperature of spacecraft

τ = time

$q_i = \Sigma E_i$ = summation of the heat energy flow in

$q_0 = \Sigma E_0$ = summation of the heat energy flow out

mc = heat capacity of the spacecraft

At thermal equilibrium $\frac{dT}{d\tau} = 0$, and Equation (9.18) reduces to

$$q_i = q_0 \quad (9.19)$$

From the detailed analysis of Equation (9.18) as applied to the OSO spacecraft, the need was seen for surface preparations that would have α/ϵ ratios of about 1, 5, and as low as possible to cool the solar-cell panel.

Ball Brothers Research Corporation constructed surface preparations with the following characteristics:

Paint No. 80U:

$$\left. \begin{array}{l} \alpha = 0.32 \text{ to } 0.34 \\ \epsilon = 0.32 \text{ to } 0.34 \end{array} \right\} \frac{\alpha}{\epsilon} = \beta \approx 1$$

Paint No. 63W:

$$\left. \begin{array}{l} \alpha = 0.2 \text{ to } 0.25 \\ \epsilon = 0.83 \text{ to } 0.85 \end{array} \right\} \frac{\alpha}{\epsilon} = \beta \approx 0.25 \text{ to } 0.3$$

Aluminum:

$$\left. \begin{array}{l} \alpha = 0.25 \text{ to } 0.30 \\ \epsilon = 0.04 \text{ to } 0.05 \end{array} \right\} \frac{\alpha}{\epsilon} = \beta \approx 5$$

The 63W (white) and the 80U (flat) coating are enamel-like coatings developed for spacecraft use. They had high thermal stability, excellent resistance to ultraviolet damage, high film flexibility, low evaporative loss in high vacuum, and known optical properties. When properly applied and cured, these coats have excellent durability, water repellency, and weatherability.

The aluminum is mechanically polished aluminum. These aluminum surfaces are the plates on top and bottom of the spacecraft body proper.

Optical Properties

The 63W had total hemispherical emissivity (ϵ) of 0.83 to 0.85 and a normal solar absorptivity (α) of 0.20 to 0.25, giving an α/ϵ ratio of about 0.3. The 80U had a total hemispherical emissivity of 0.32 and a normal solar absorptivity of 0.32 to 0.34, giving an α/ϵ ratio of about 1. Since (ϵ) is a surface property

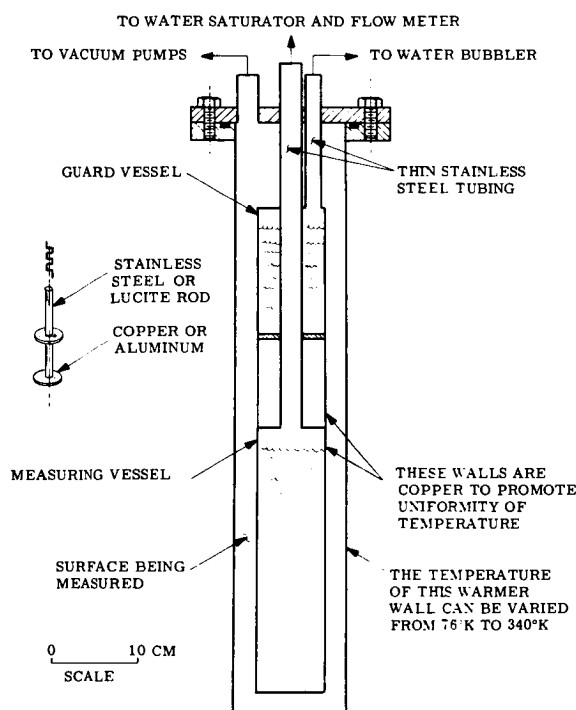


FIGURE 9-27.—Hemispherical emissivity fixture.

and would change with serious resin evaporation the (ϵ) was monitored very carefully for 88 hours in vacuum (4×10^{-6} mm Hg). No change was measured.

The total hemispherical emissivity was determined on an apparatus (Figure 9-27) using cylinder-cylinder geometry based on the Christiansen equation.†

$$q = \frac{\epsilon_2 \epsilon_0 A_i \tau}{\epsilon_0 + \epsilon_i (1 - \epsilon_0) A_i / A_0} (T_2^4 - T_1^4) \quad (9.20)$$

The α_0/ϵ ratio was determined in the apparatus shown in Figure 9-28. This apparatus was based on Equation (9.19) as applied to a thin disk geometry:

$$q_0 = q_i \quad (9.19)$$

or

$$2\pi r^2 \epsilon \tau (T_2^4 - T_1^4) = S_\odot \pi r^2 \alpha_\odot \quad (9.21)$$

With the outer wall at 76°K, Equation (9.21) reduces to

$$T_2^4 = \frac{1}{2} \frac{S_\odot}{\sigma} \left(\frac{\alpha_\odot}{\epsilon} \right) \quad (9.22)$$

† C. Christiansen, *Ann. d. Phys. U. Chem.*, Vol. 19, p. 267, 1883.

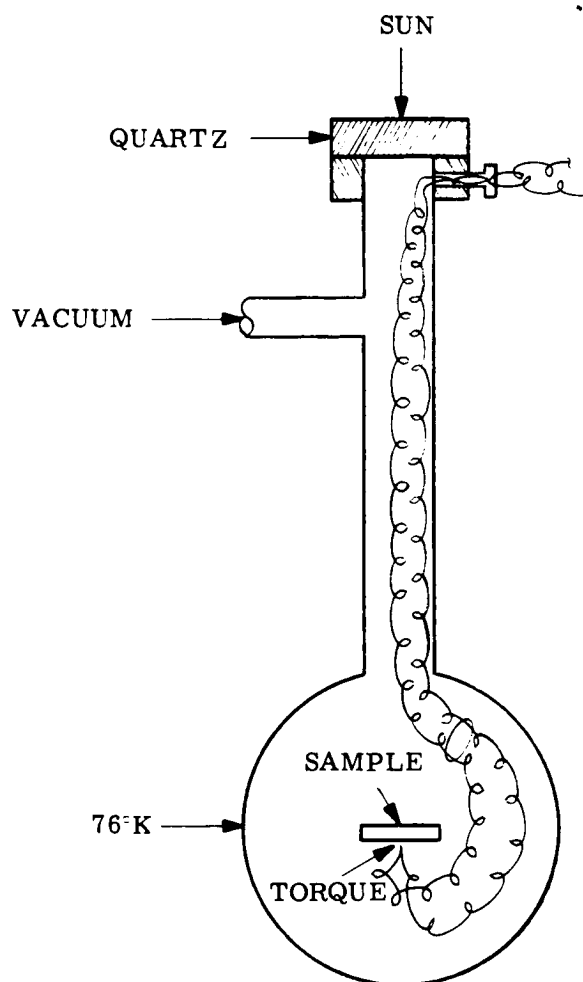


FIGURE 9-28.—Radiometer.

when the sample has reached a steady-state temperature, where

- $S_\odot$ = solar constant
- σ = Stefan-Boltzmann constant
- T_2 = temperature of the sample
- $\alpha_\odot$ = solar absorptivity
- ϵ = total hemispherical emissivity

The vacuum weight losses of the various enamels were determined to evaluate their possible influence on the ultraviolet equipment on the spacecraft. Figures 9-29 through 9-32 show the weight loss per unit area and total weight loss in vacuum for 63W and 80U.

The coatings were exposed to an Osram HBO-500 Hg lamp in air at three times solar

TEST PROGRAM

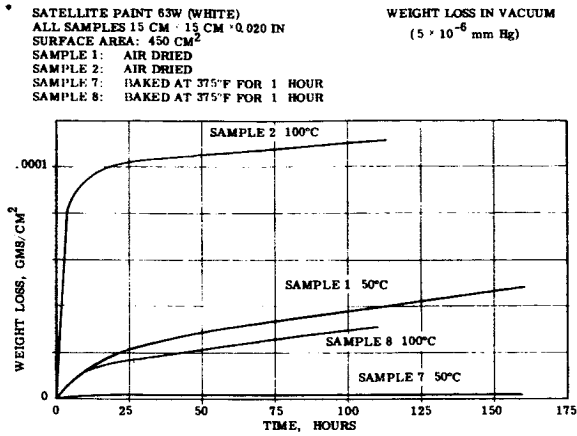


FIGURE 9-29.—Weight loss per unit area in vacuum of BBRC point 63W.

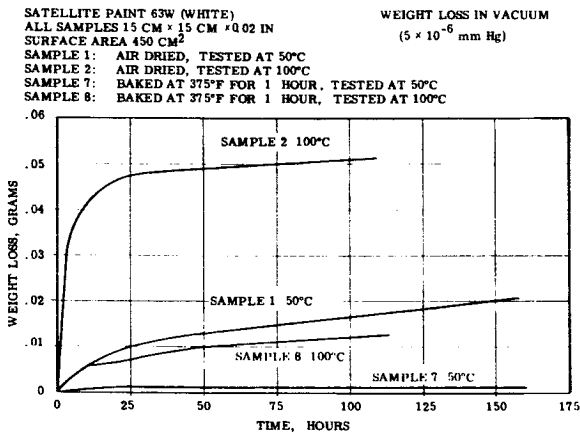


FIGURE 9-30.—Weight loss in vacuum of BBRC point 63W.

intensity for about 500 hours. (The quartz envelope of the lamp cuts out wavelengths less than about 0.2 μ . There was no visible evidence of discoloration even on the white (63W), but there was some slight mud-cracking of the surface, too slight to seriously influence the temperature of the spacecraft.

Application of Coatings

The coatings were applied over most clean metallic surfaces by spraying, flow coating, brushing, or dipping. A clean (degreased) surface with all loose particles removed was necessary for optimum film adhesions. However, for hard service the metal surfaces were

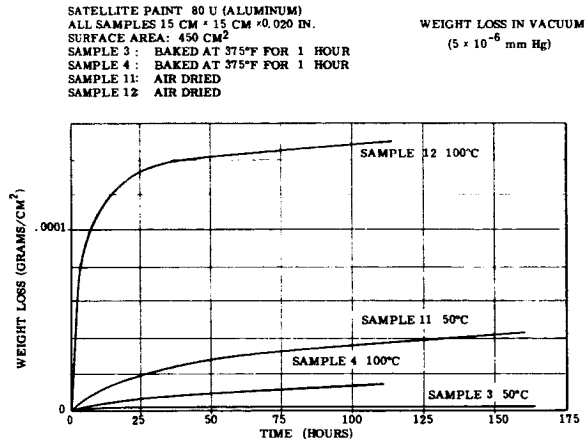


FIGURE 9-31.—Weight loss per unit area in vacuum of BBRC point 80U.

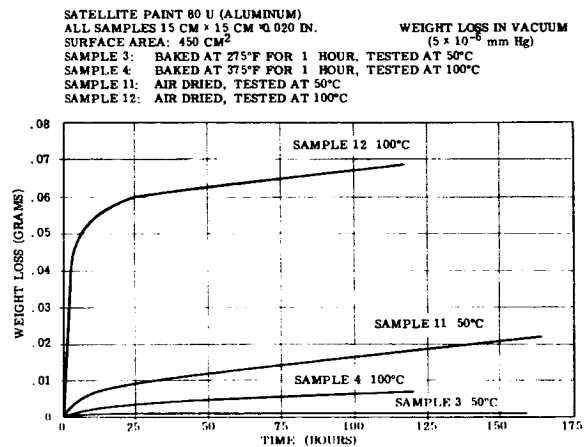


FIGURE 9-32.—Weight loss in vacuum of BBRC point 80U.

prepared as if to paint, up to the point where an organic type primer was used. Aluminum alloys were prepared per MIL-C-5541 (for example, the Okite process), and were then either chromacoated or primed per MIL-C-15328A before final coatings were applied. Excellent film flow in spraying was obtained by thinning slightly with Butyl Cello-solve.

Air dry produced a Sward hardness of about 10 and after a minimum bake of 30 minutes at 400°F gave a maximum hardness of about 25. For any future applications for use above 350°F it is recommended that an additional bake-out of 15 minutes at 480°F be used to reduce the thermoplasticity or tack. If applications are

not for long-time use in ultraviolet, the addition of Octasol cobalt drier will lower the thermo-plasticity.

Samples of the enamel formulation were put on $\frac{1}{16}$ -inch-thick 2024 aluminum. An acetylene torch was applied to the unpainted side. The aluminum substrate melted away leaving a paint film free-standing. The paint did not

support combustion. Other samples were immersed in liquid nitrogen with no noticeable change. Two samples were exposed to outdoor environmental conditions for nearly seven months. The samples were in direct sunlight and facing south. No noticeable change in properties was observed. The surfaces remained remarkably clean of contamination.

APPENDIX A

ANTENNA PATTERNS

The following antenna patterns were recorded on the Ball Brothers Research Corporation antenna range. There are 18 transmitter patterns and three principal-plane receiver patterns.

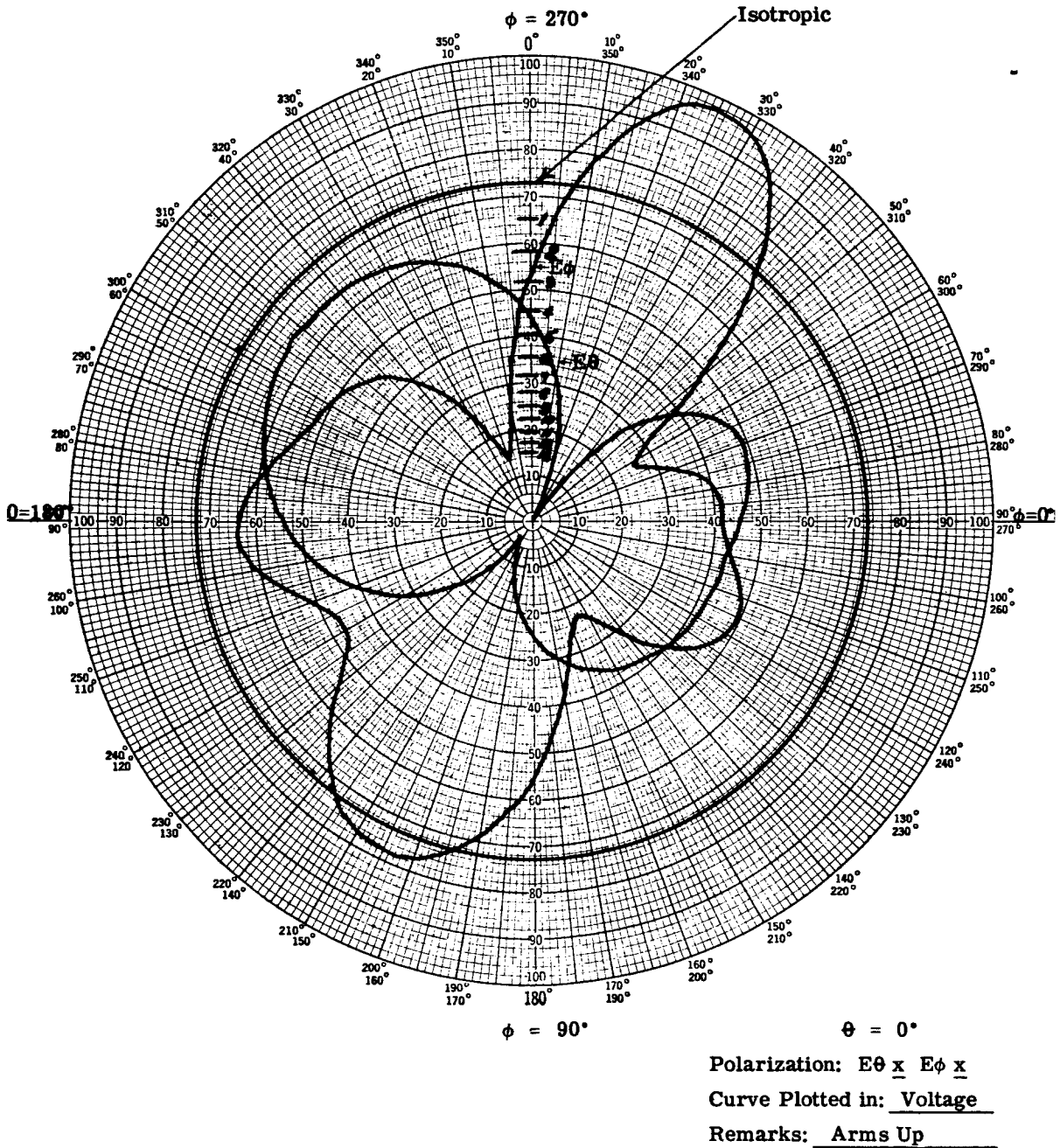


FIGURE A-1.—Transmitter frequency antenna pattern No. 1.

ORBITING SOLAR OBSERVATORY

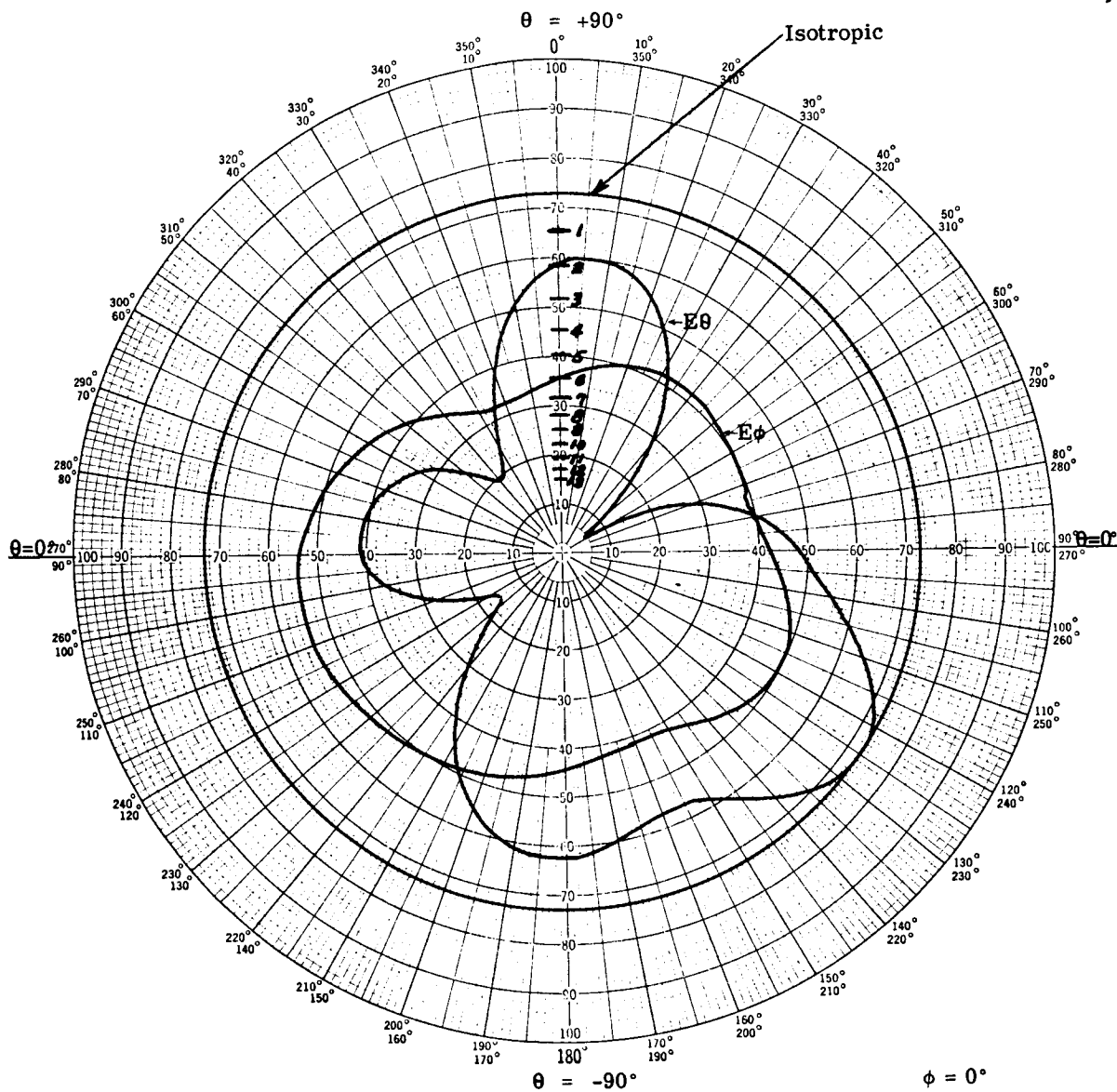


FIGURE A-2.—Transmitter frequency antenna pattern No. 2.

APPENDIX A—ANTENNA PATTERNS

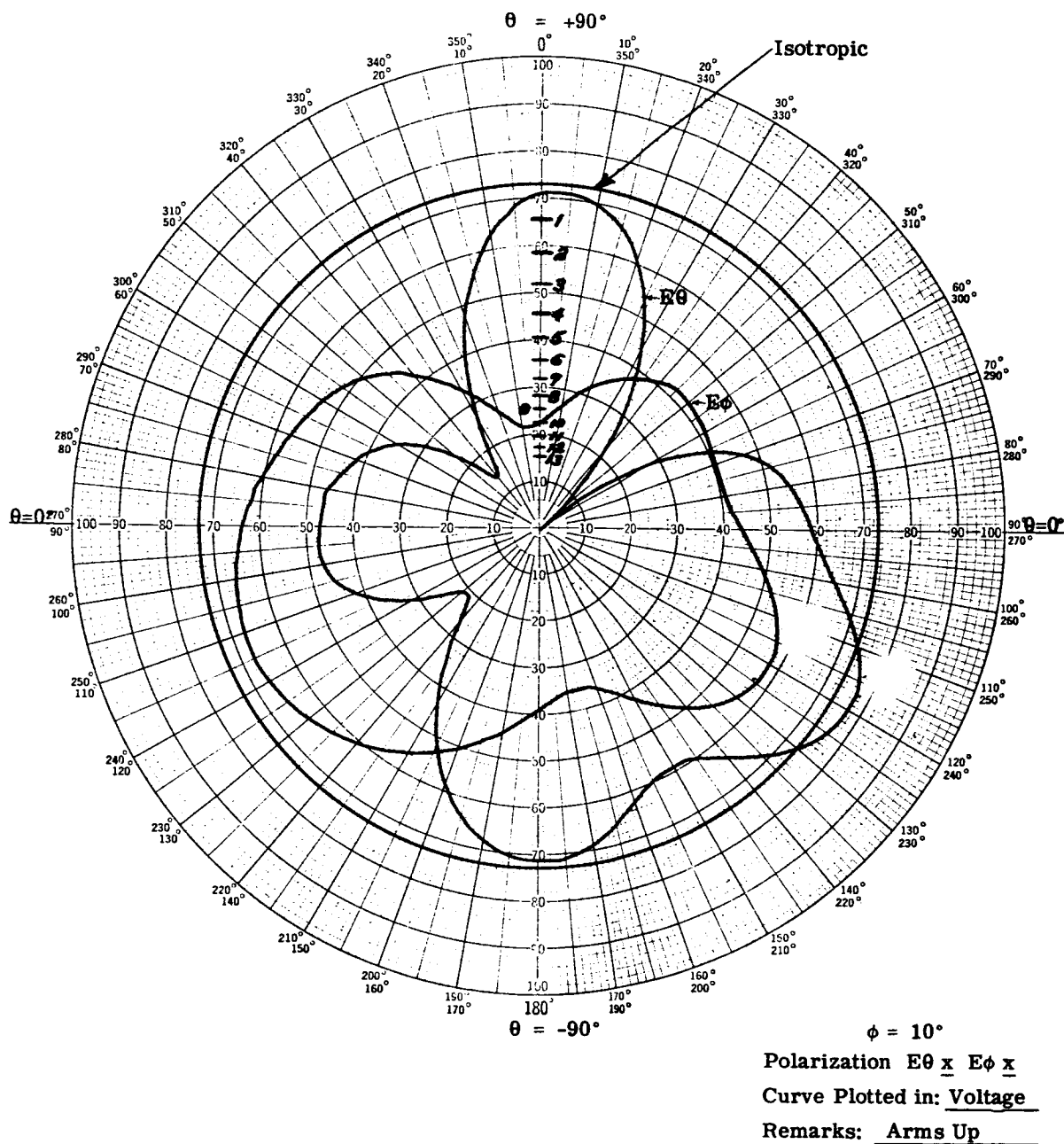
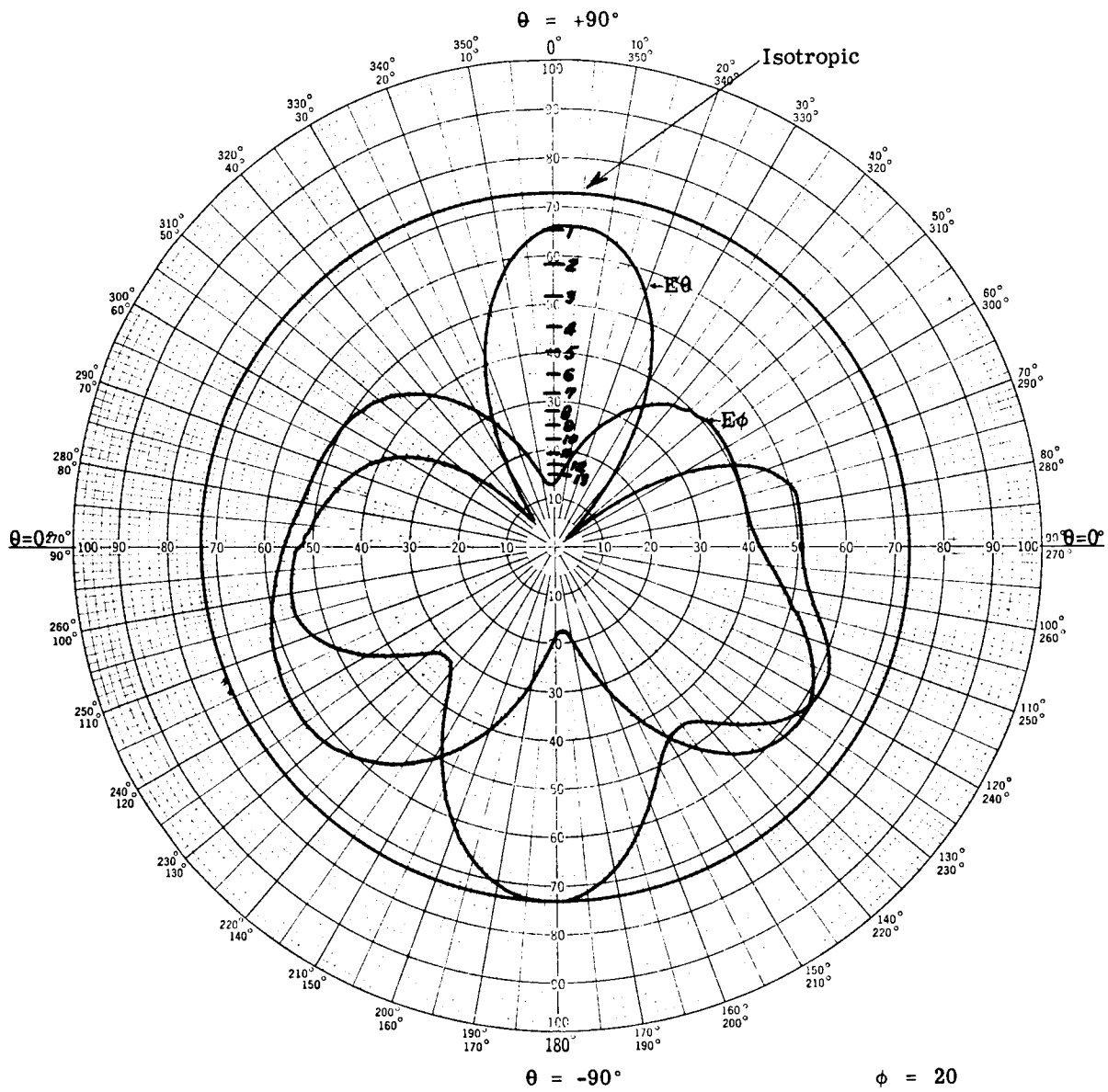


FIGURE A-3.—Transmitter frequency antenna pattern No. 3.

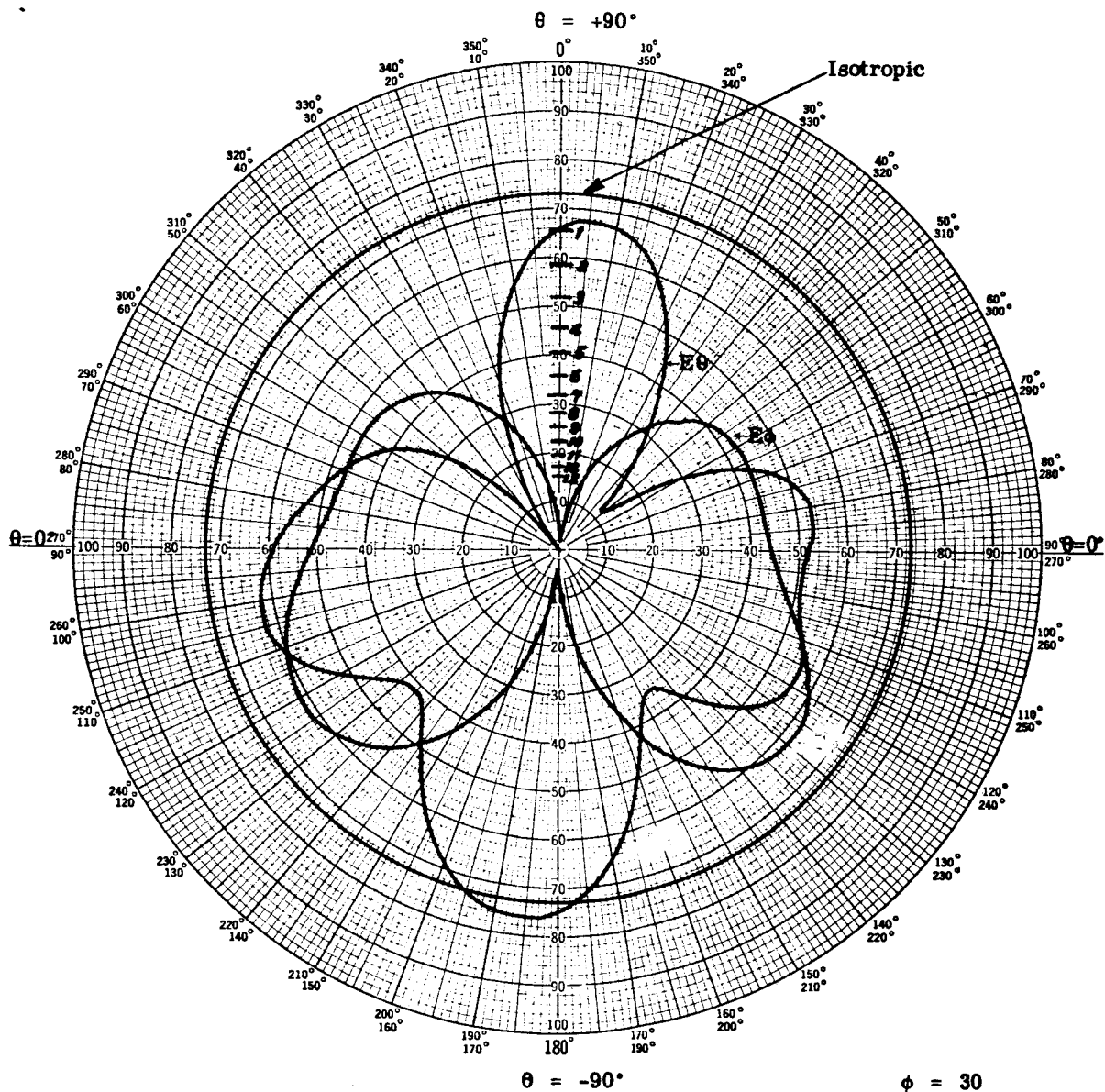
ORBITING SOLAR OBSERVATORY



Polarization $E_{\theta} \times E_{\phi}$
Curve Plotted in: Voltage
Remarks: Arms Up

FIGURE A-4.—Transmitter frequency antenna pattern No. 4.

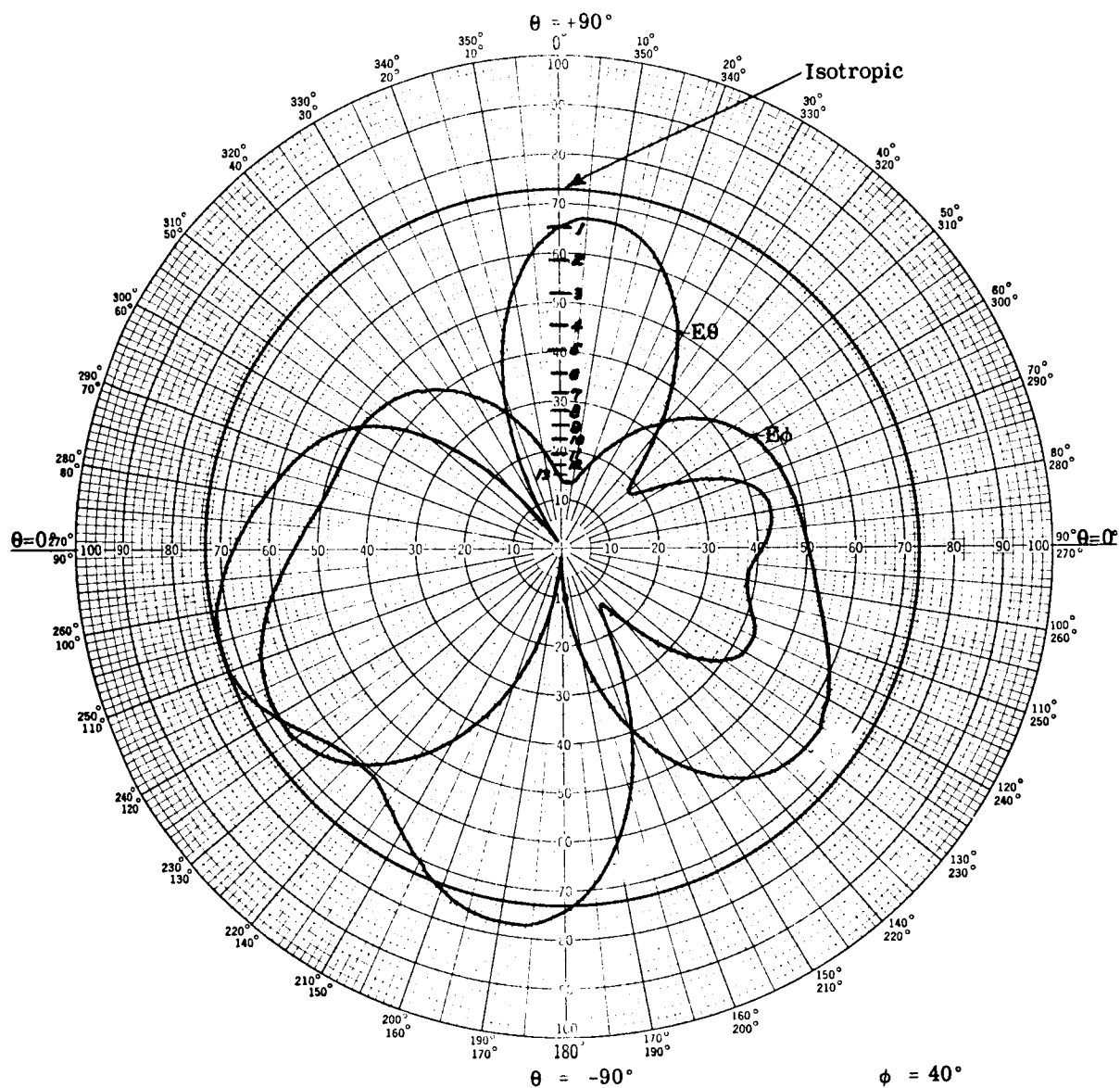
APPENDIX A—ANTENNA PATTERNS



Polarization $E_\theta \times E_\phi \times$
 Curve plotted in: Voltage
 Remarks: Arms Up

FIGURE A-5.—Transmitter frequency antenna pattern No. 5.

ORBITING SOLAR OBSERVATORY



$\phi = 40^\circ$
Polarization: $E_\theta \times E_\phi$
Curves Plotted in: Voltage
Remarks: Arms Up

FIGURE A-6.—Transmitter frequency antenna pattern No. 6.

APPENDIX A—ANTENNA PATTERNS

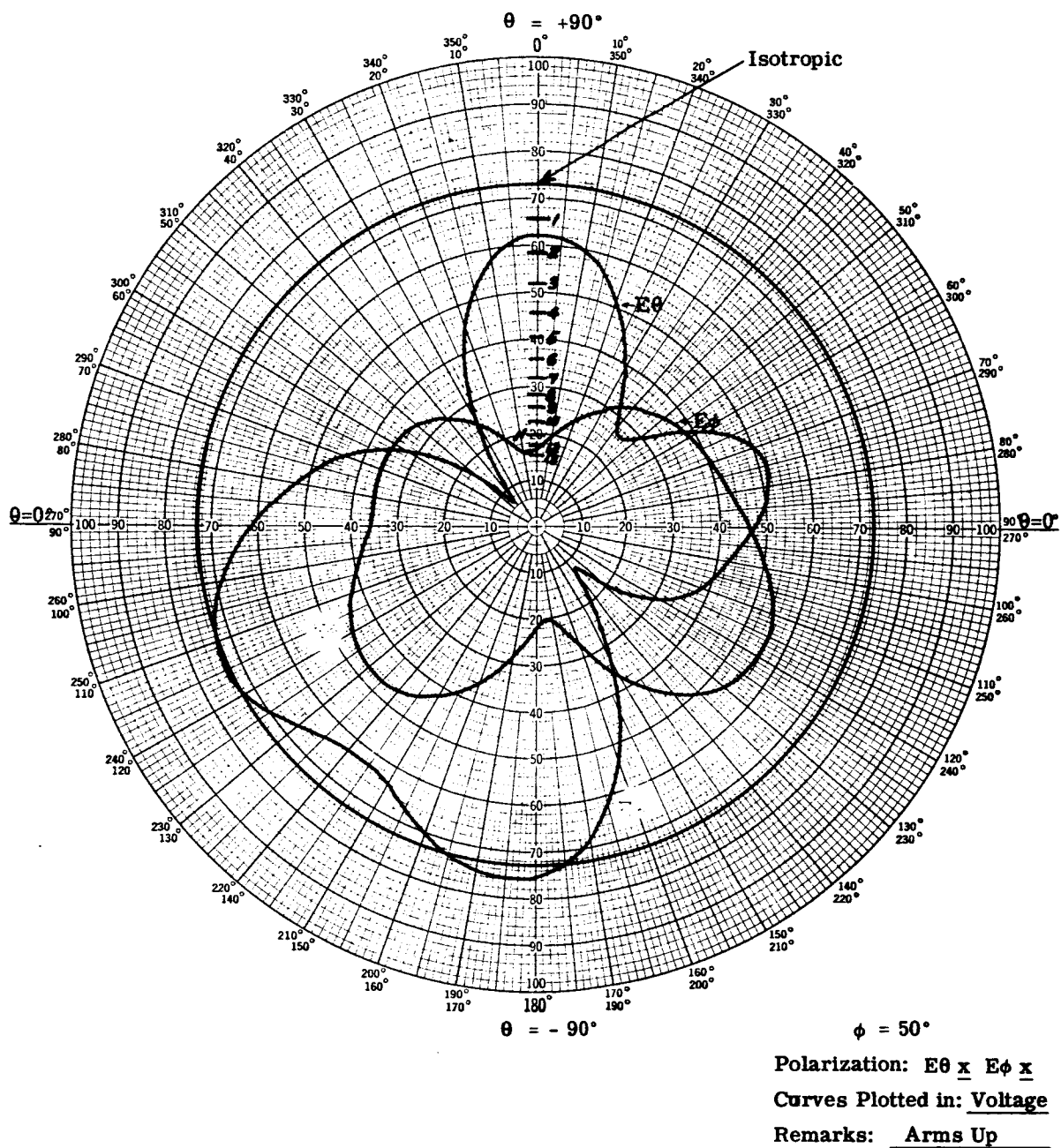
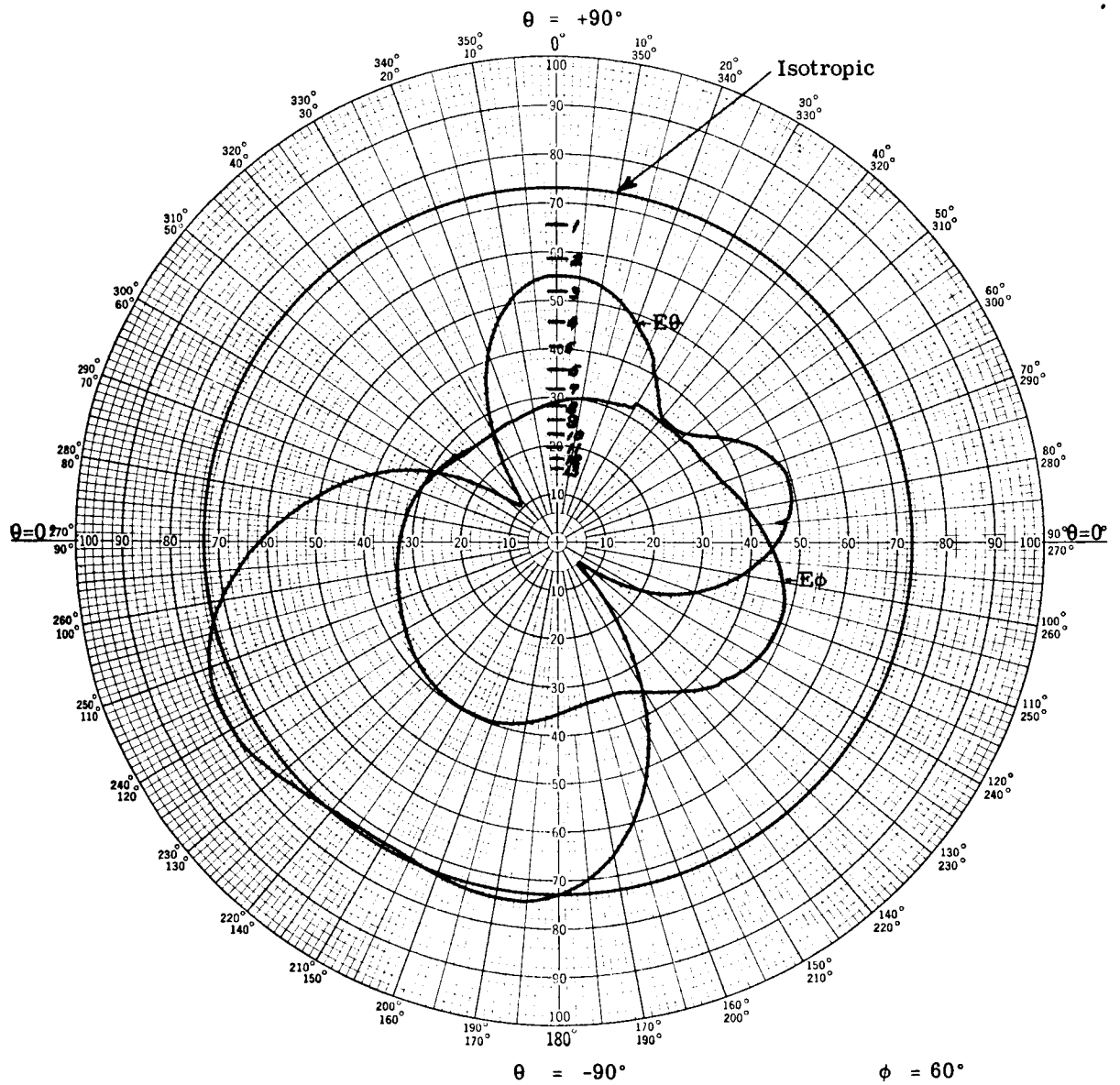


FIGURE A-7.—Transmitter frequency antenna pattern No. 7.

ORBITING SOLAR OBSERVATORY



$\phi = 60^\circ$

Polarization: $E\theta \times E\phi$

Curve Plotted in: Voltage

Remarks: Arms Up

FIGURE A-8.—Transmitter frequency antenna pattern No. 8.

APPENDIX A—ANTENNA PATTERNS

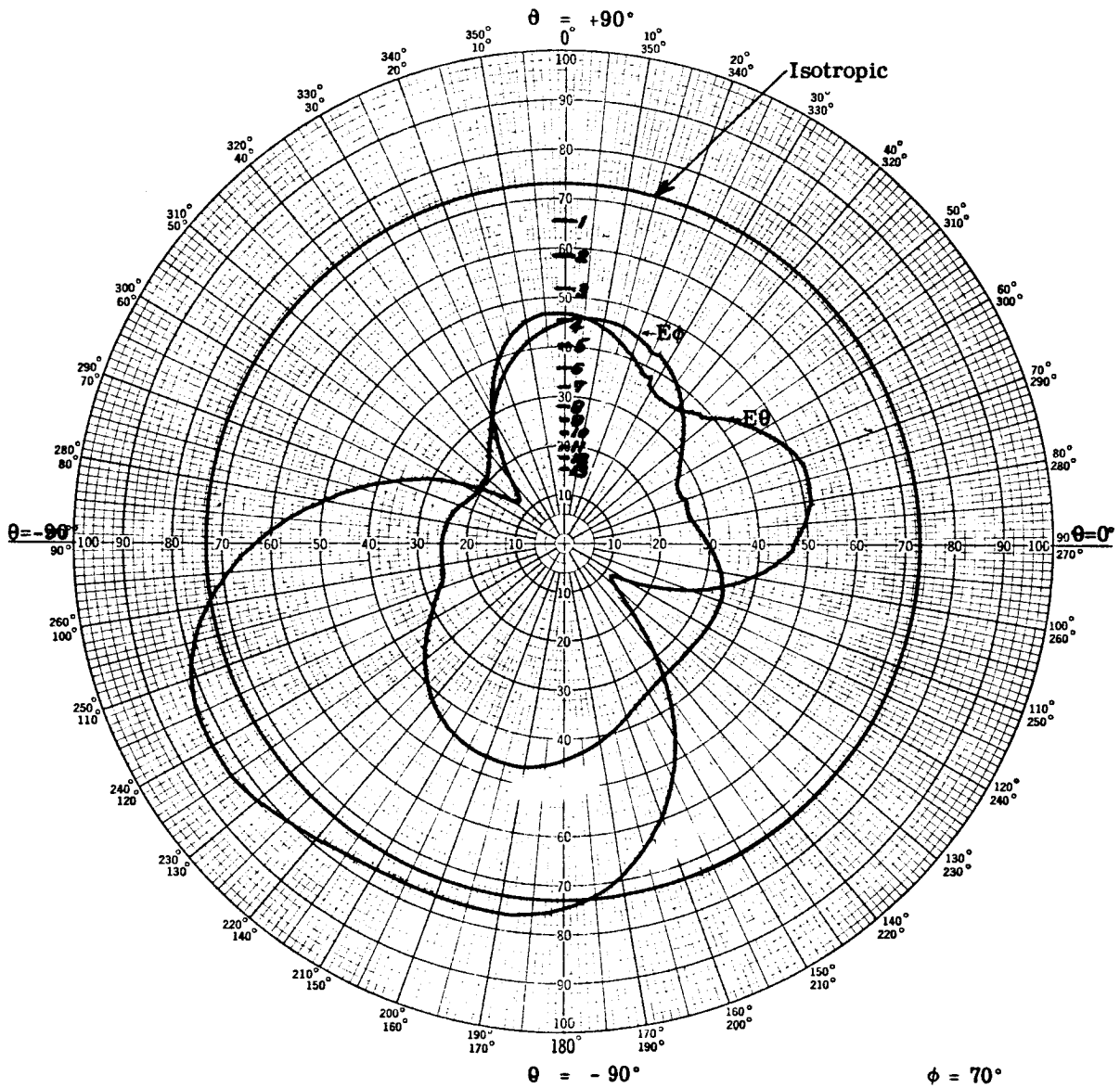
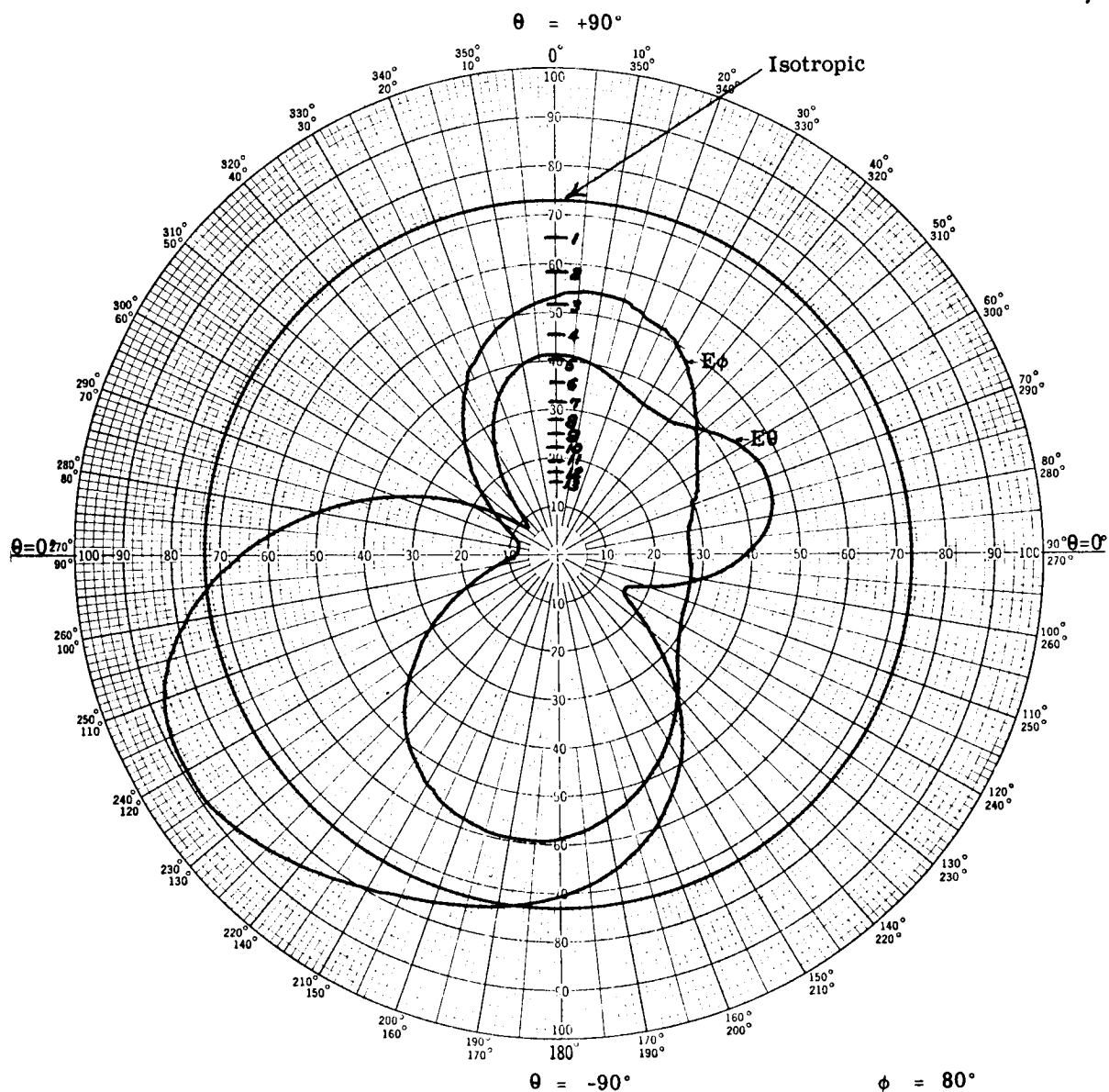


FIGURE A-9.—Transmitter frequency antenna pattern No. 9.

ORBITING SOLAR OBSERVATORY



$\phi = 80^\circ$

Polarization: $E\theta \times E\phi \times$

Curves Plotted in: Voltage

Remarks: Arms Up

FIGURE A-10.—Transmitter frequency antenna pattern No. 10.

APPENDIX A—ANTENNA PATTERNS

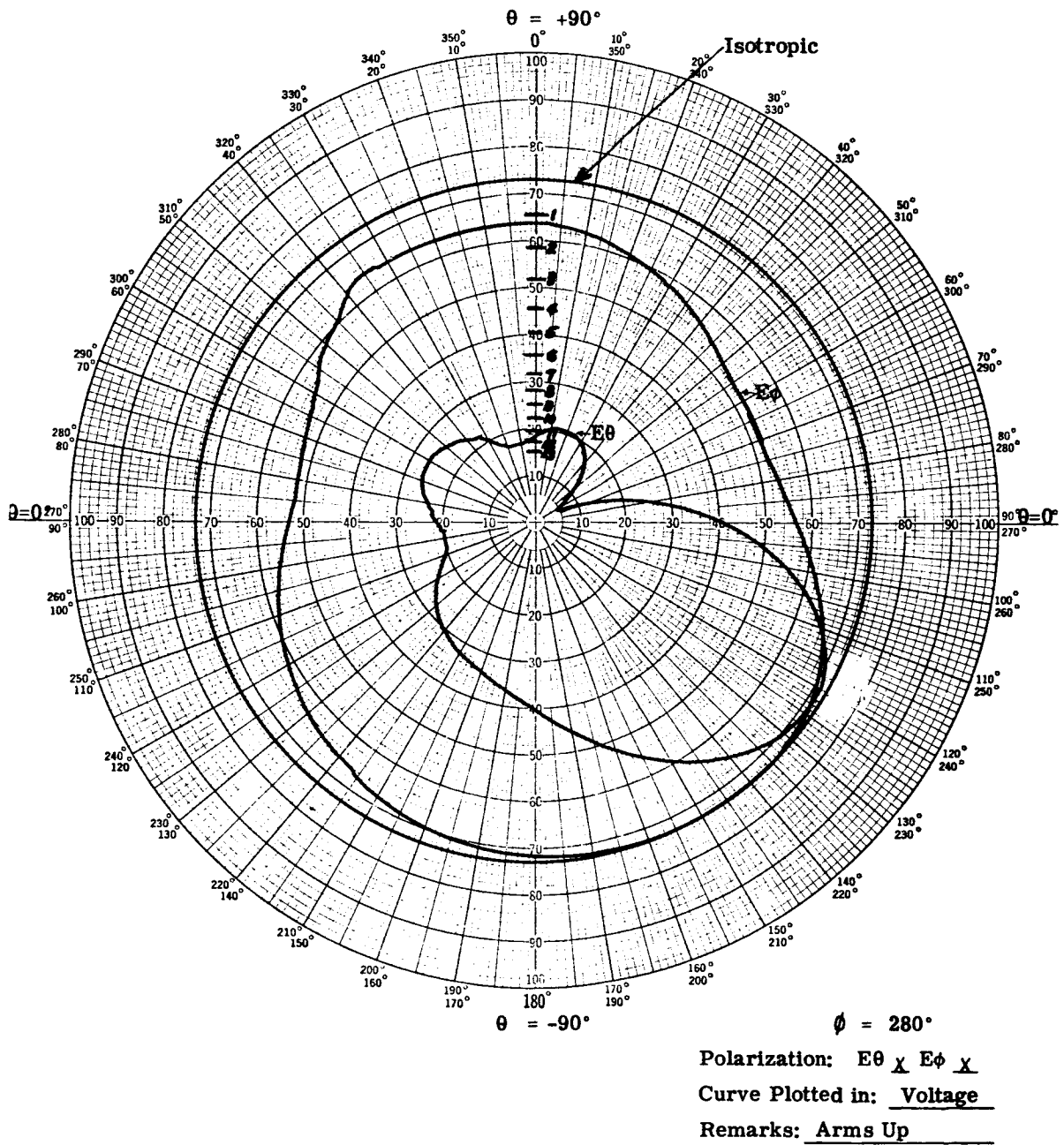
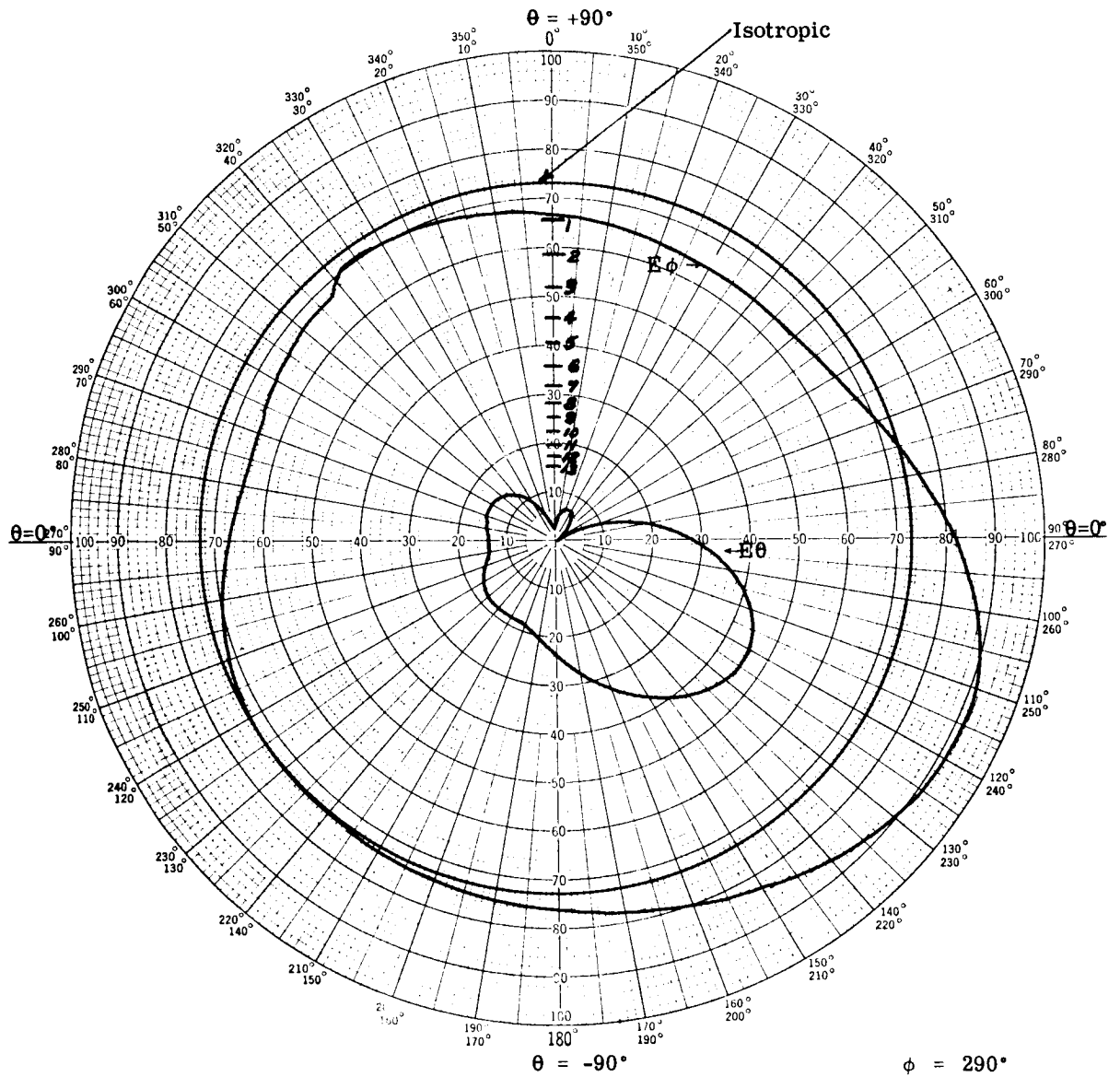


FIGURE A-11.—Transmitter frequency antenna pattern No. 11.

ORBITING SOLAR OBSERVATORY



$\phi = 290^\circ$

Polarization: $E\theta \times E\phi$

Curve Plotted in: Voltage

Remarks: Arms Up

FIGURE A-12.—Transmitter frequency antenna pattern No. 12.

APPENDIX A—ANTENNA PATTERNS

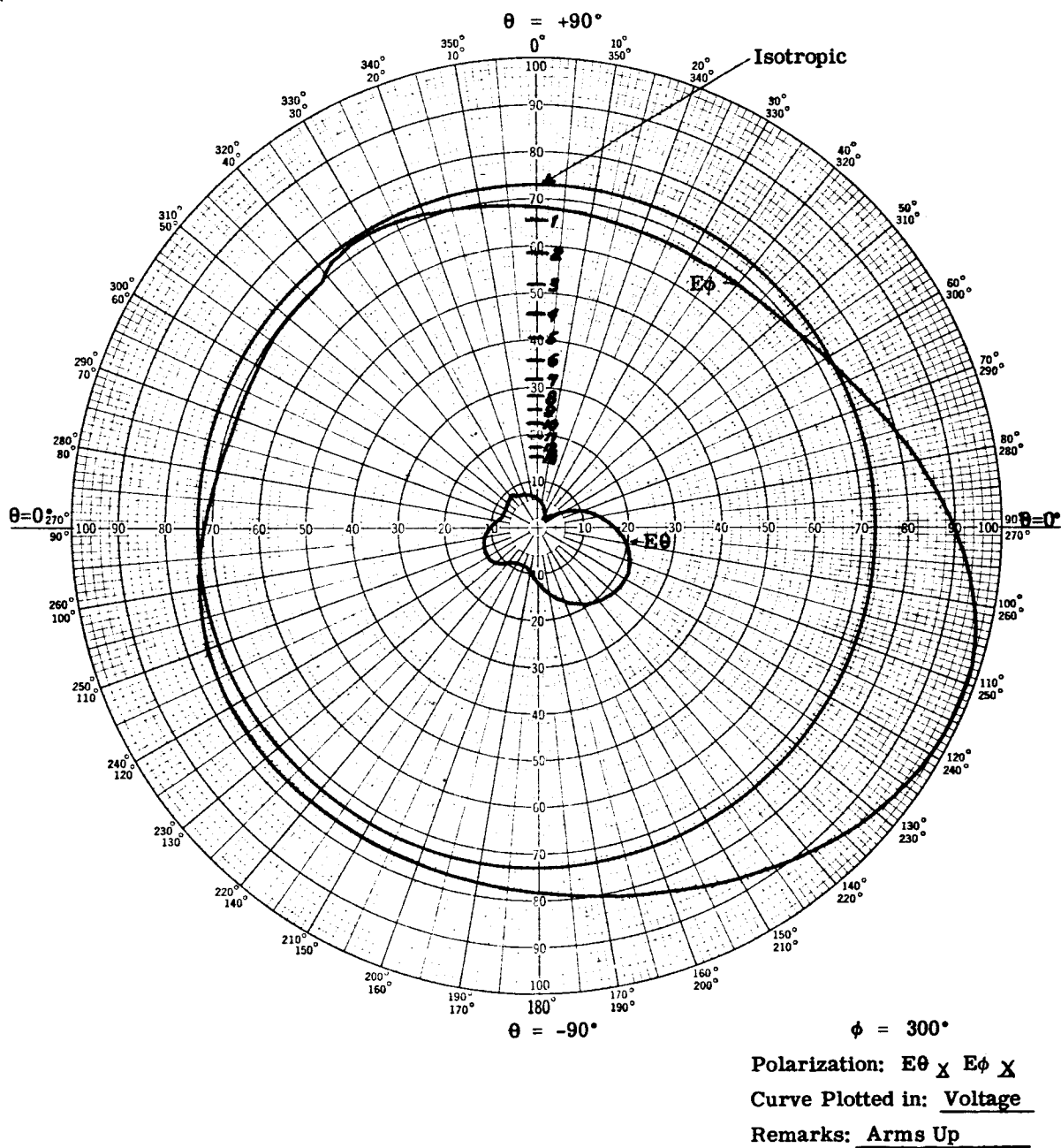
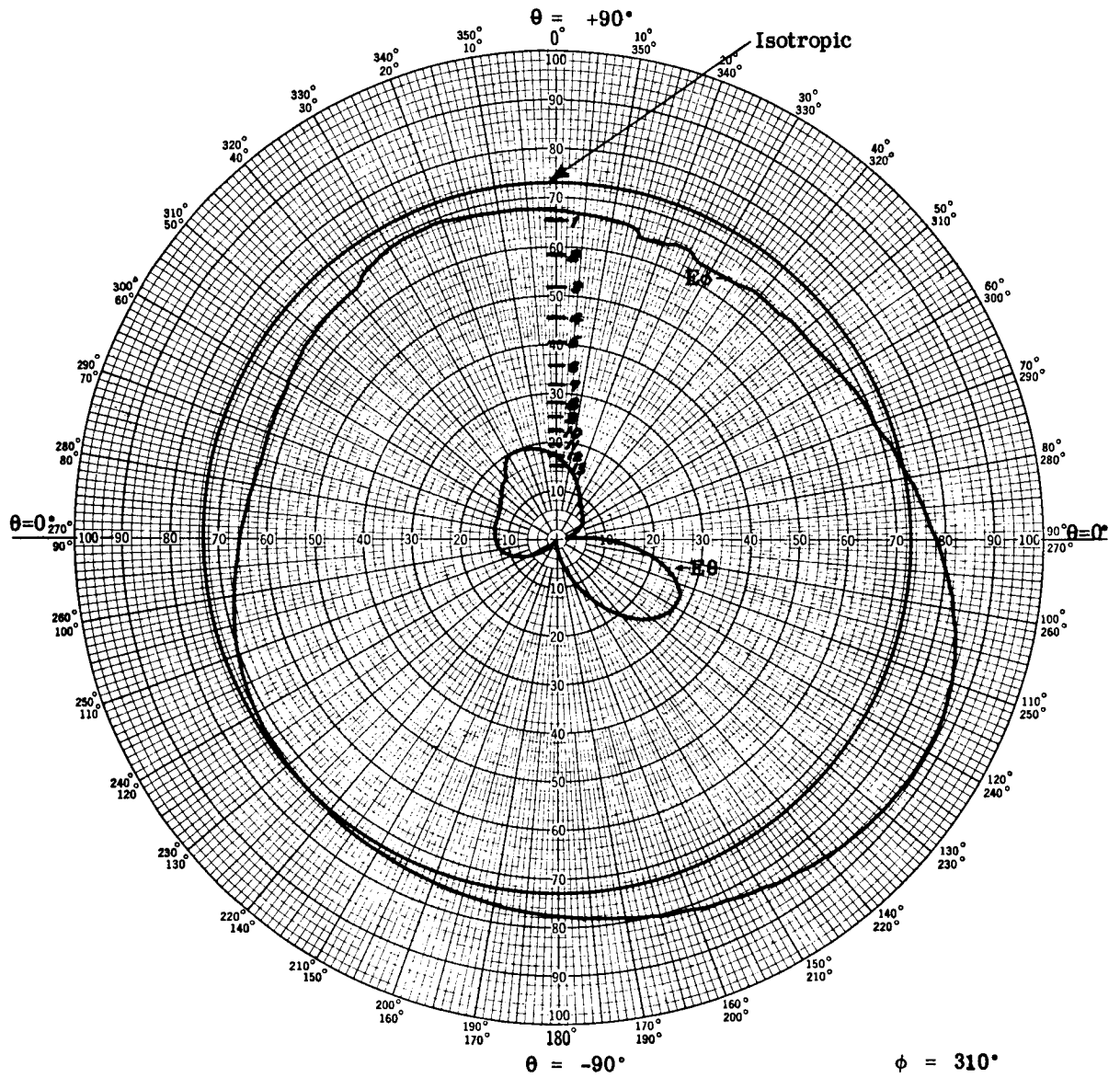


FIGURE A-13.—Transmitter frequency antenna pattern No. 13.

ORBITING SOLAR OBSERVATORY



Polarization: $E\theta \times E\phi$
 Curve Plotted in: Voltage
 Remarks: Arms Up

FIGURE A-14.—Transmitter frequency antenna pattern No. 14.

APPENDIX A—ANTENNA PATTERNS

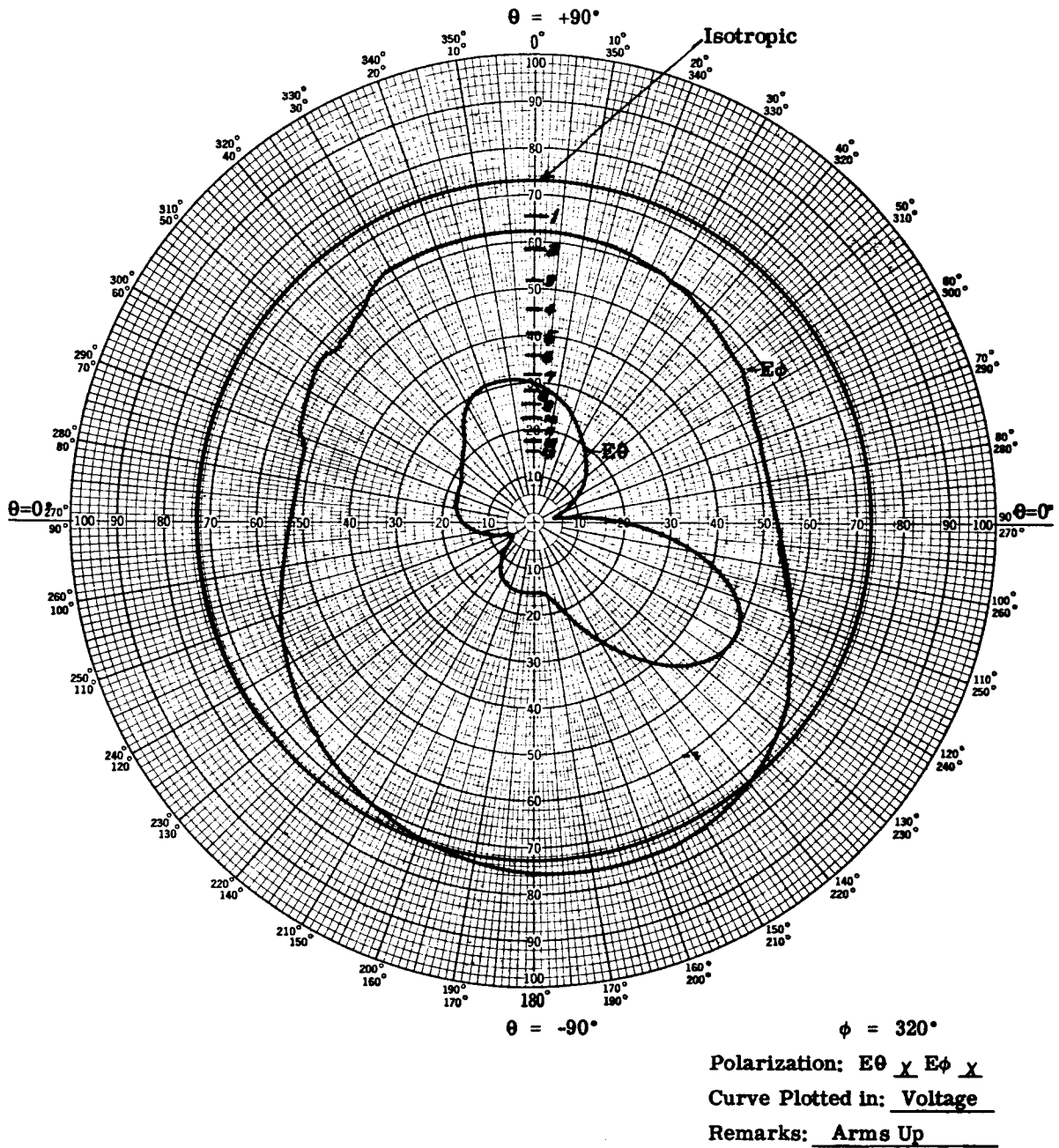
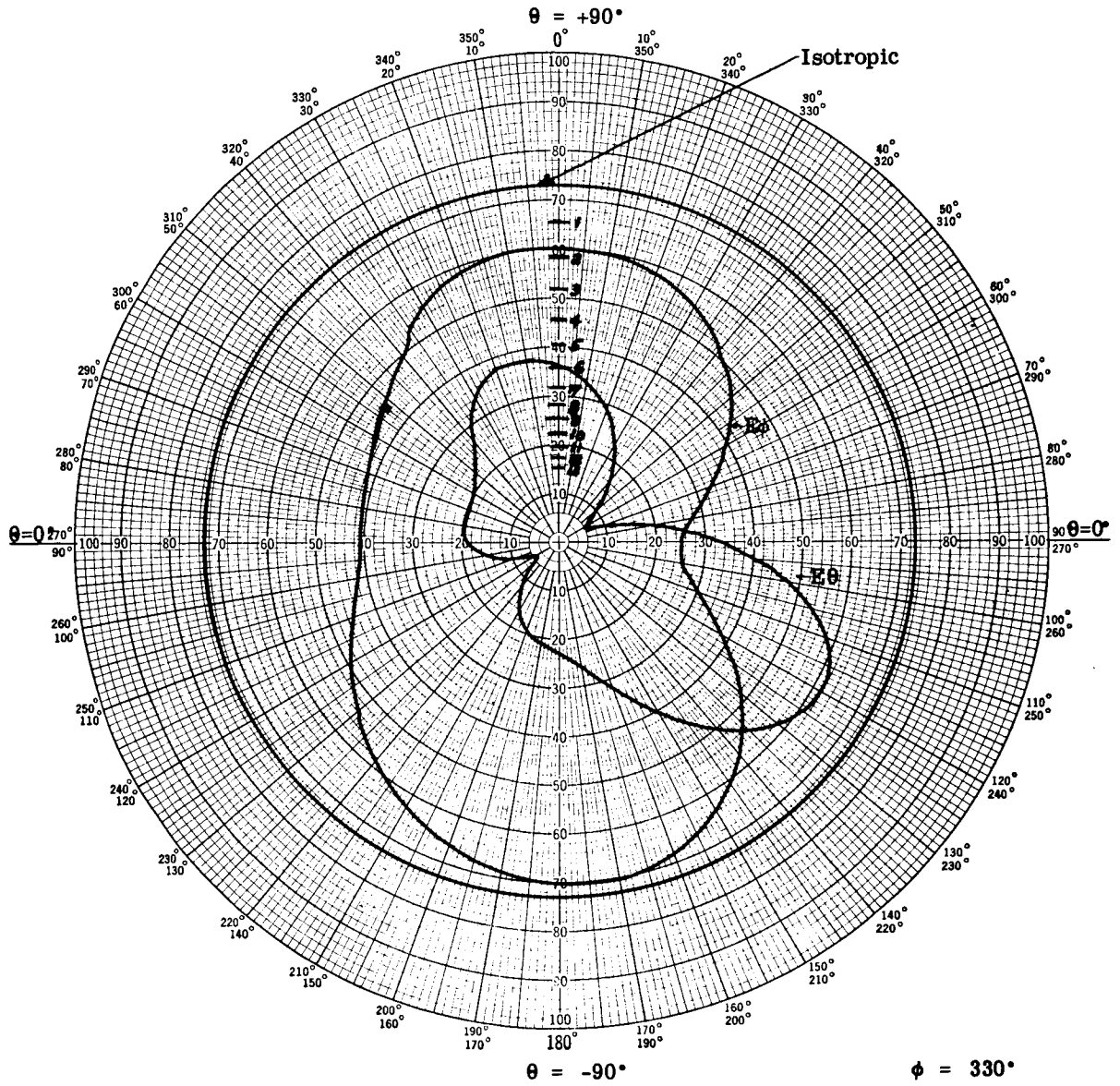


FIGURE A-15.—Transmitter frequency antenna pattern No. 15.

ORBITING SOLAR OBSERVATORY



Polarization: $E_{\theta} \propto E_{\phi} \propto$

Curve Plotted in: Voltage

Remarks: Arms Up

FIGURE A-16.—Transmitter frequency antenna pattern No. 16.

APPENDIX A—ANTENNA PATTERNS

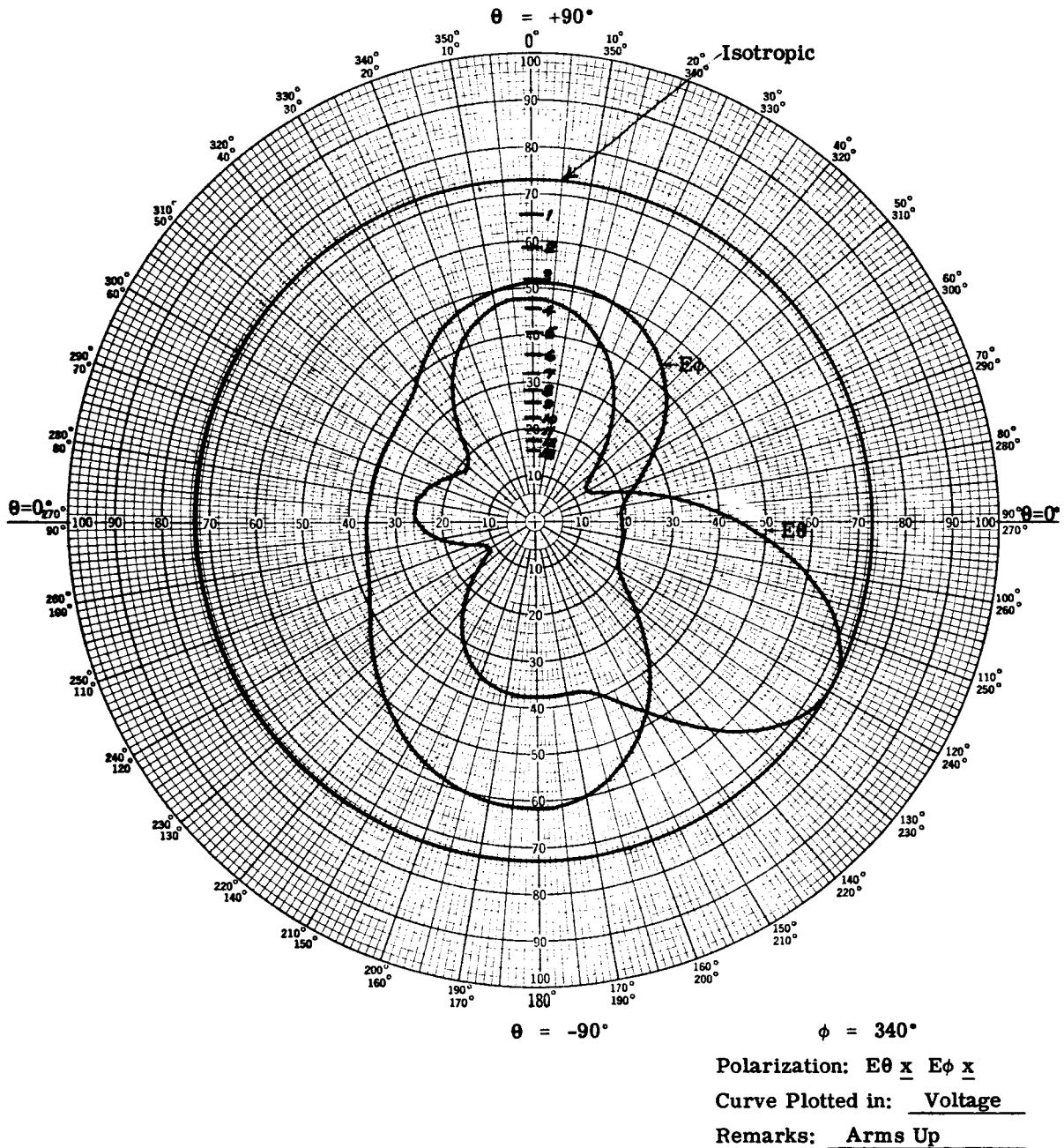
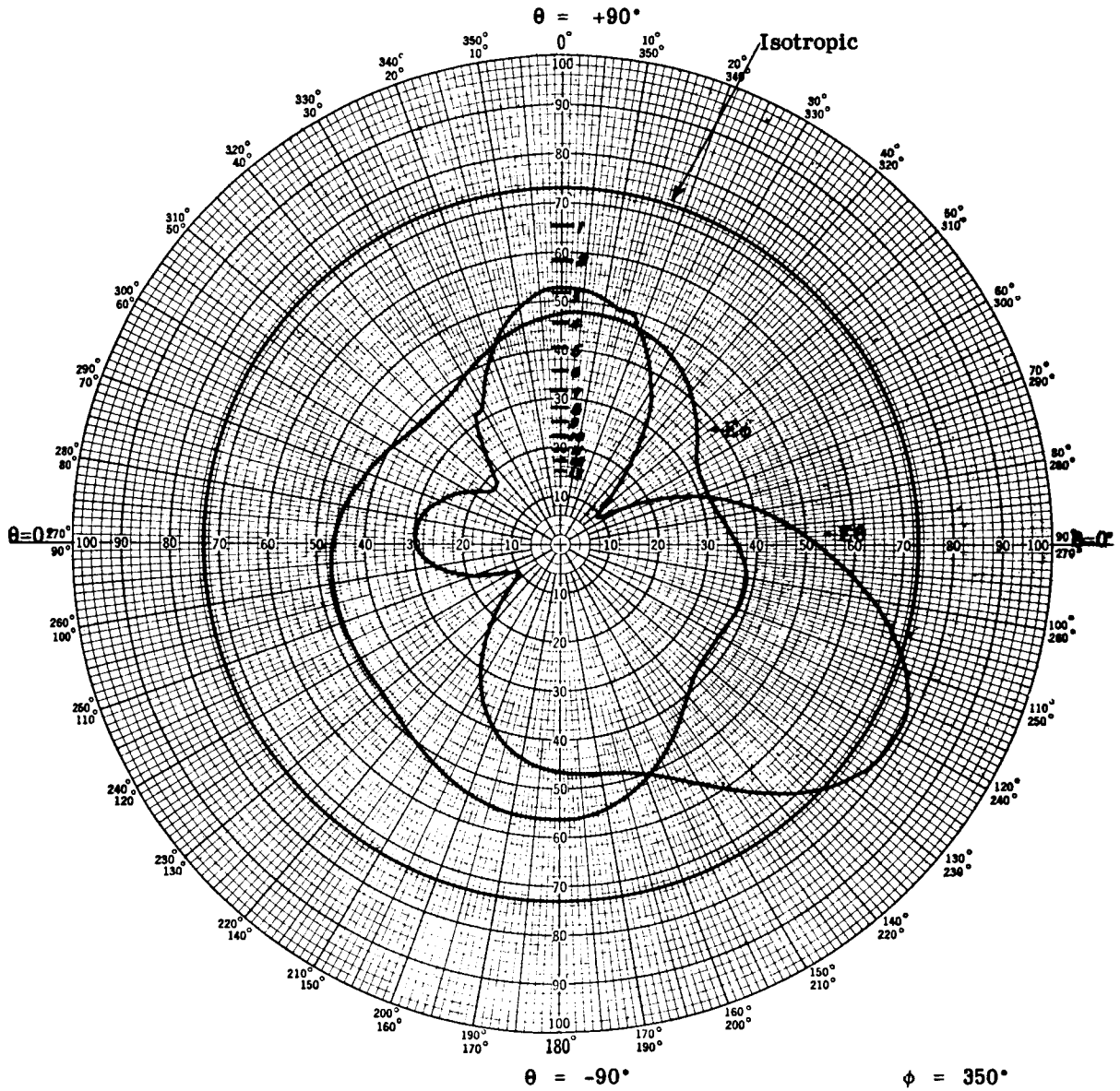


FIGURE A-17.—Transmitter frequency antenna pattern No. 17.

ORBITING SOLAR OBSERVATORY



Polarization: $E_\theta \times E_\phi$
 Curve Plotted in: Voltage
 Remarks: Arms Up

FIGURE A-18.—Transmitter frequency antenna pattern No. 18.

APPENDIX A—ANTENNA PATTERNS

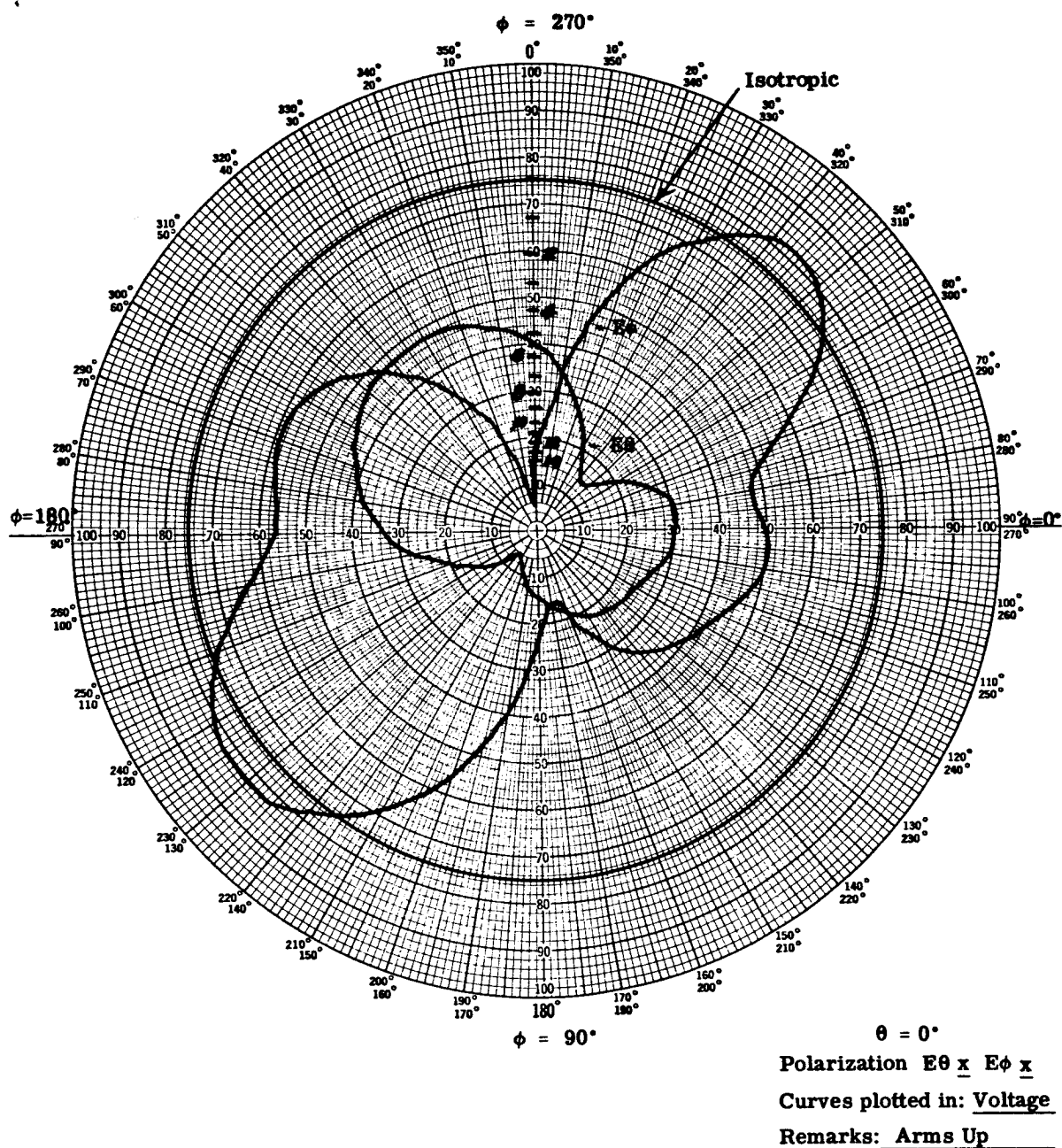
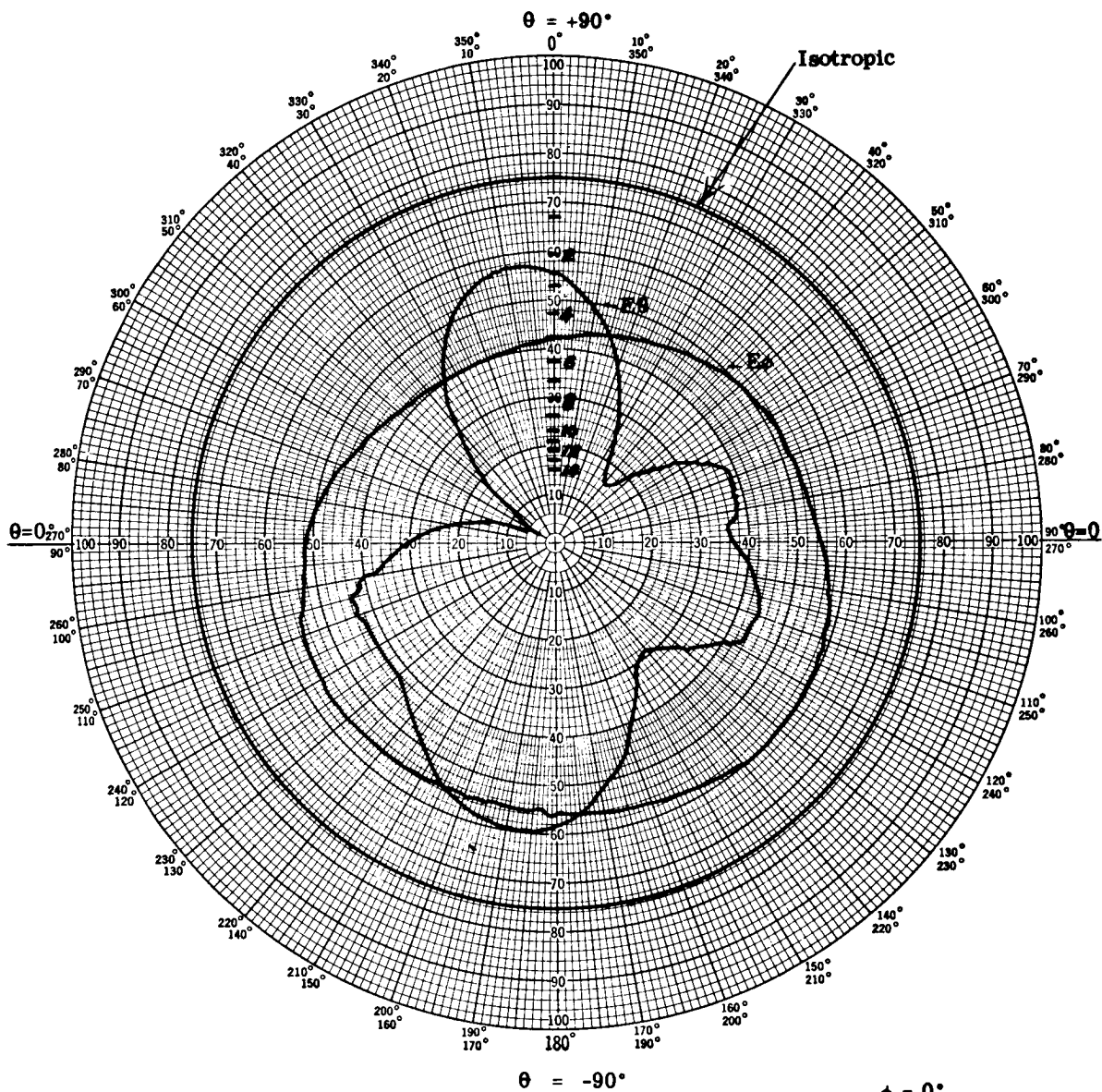


FIGURE A-19.—Receiver frequency antenna pattern No. 1.



Polarization: $E\theta \times E\phi$
 Curves Plotted in: Voltage
 Remarks: Arms up

FIGURE A-20.—Receiver frequency antenna pattern No. 2.

APPENDIX A—ANTENNA PATTERNS

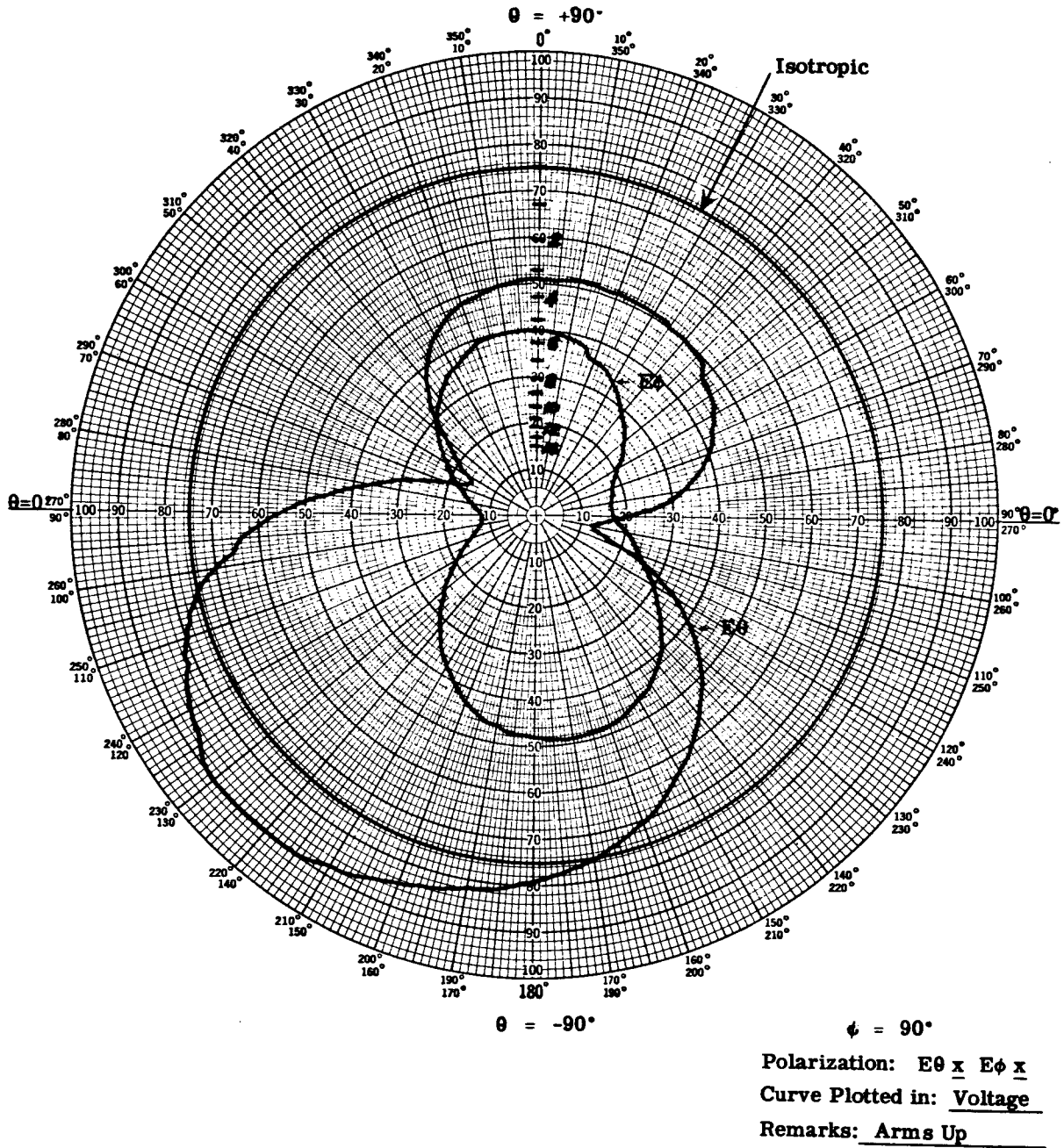


FIGURE A-21.—Receiver frequency antenna pattern No. 3.

APPENDIX B

PASSIVE TEMPERATURE CONTROL

The following curves indicate the results accomplished by the passive temperature control work.

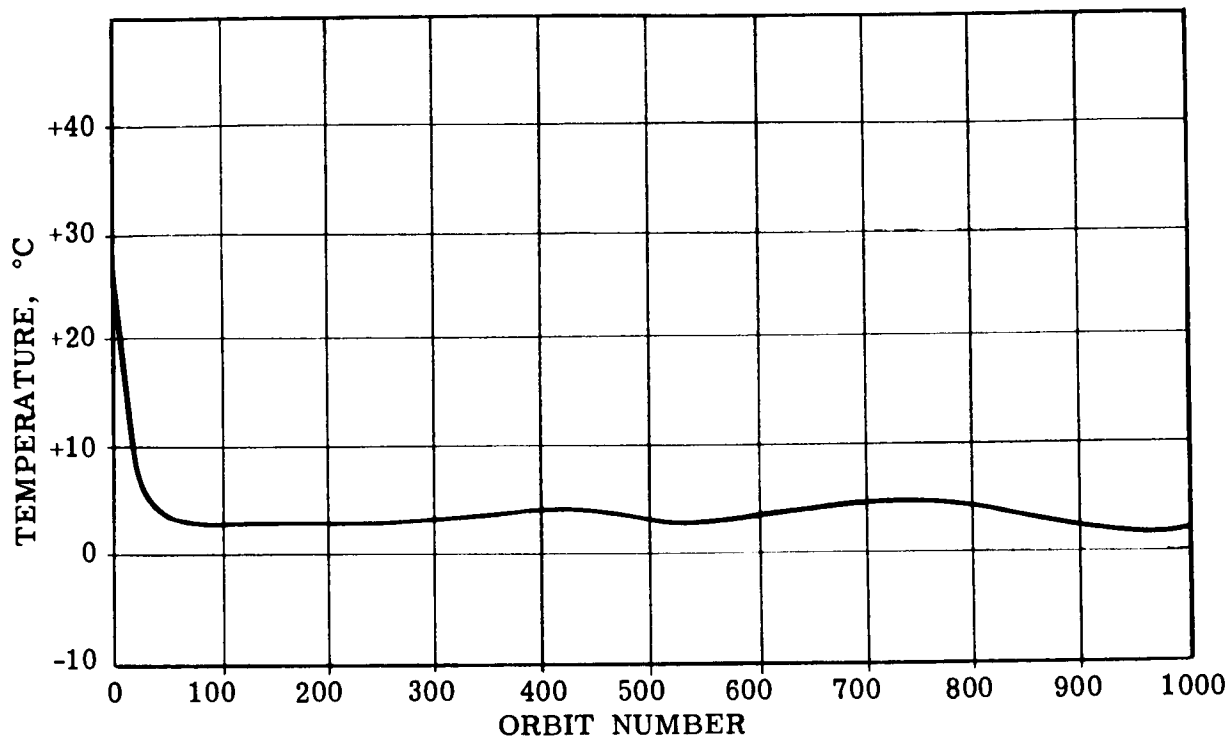


FIGURE B-1.—Battery No. 1 temperature monitor, channel 2.

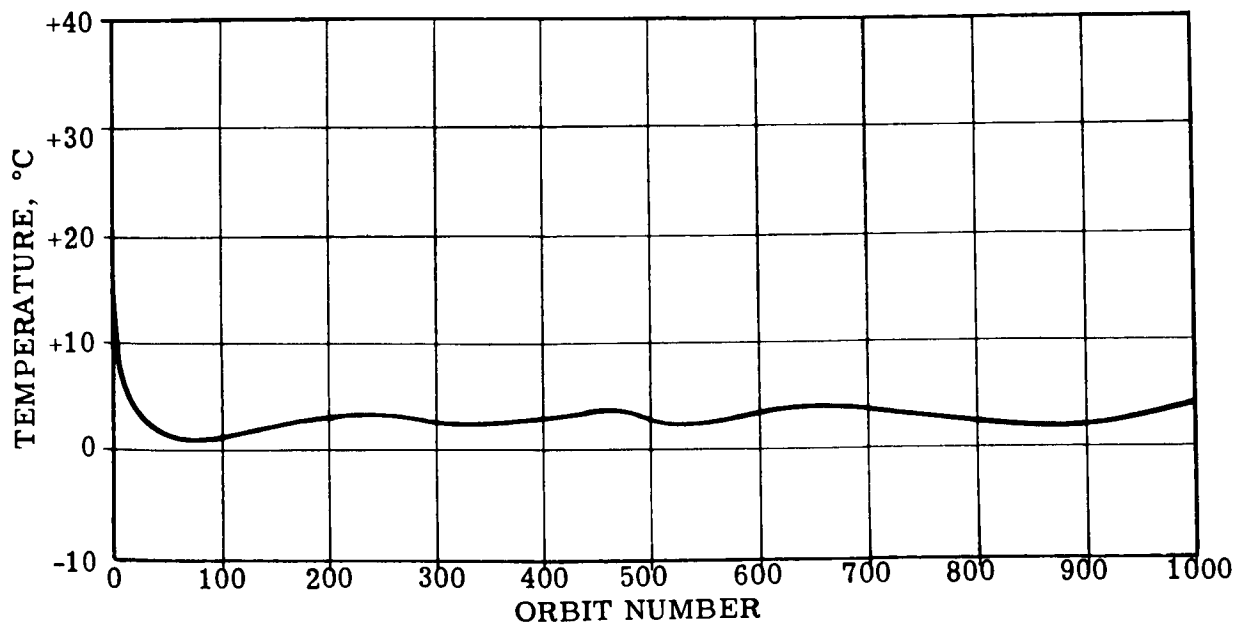


FIGURE B-2.—Battery No. 2 temperature monitor, channel 3.

APPENDIX B—PASSIVE TEMPERATURE CONTROL

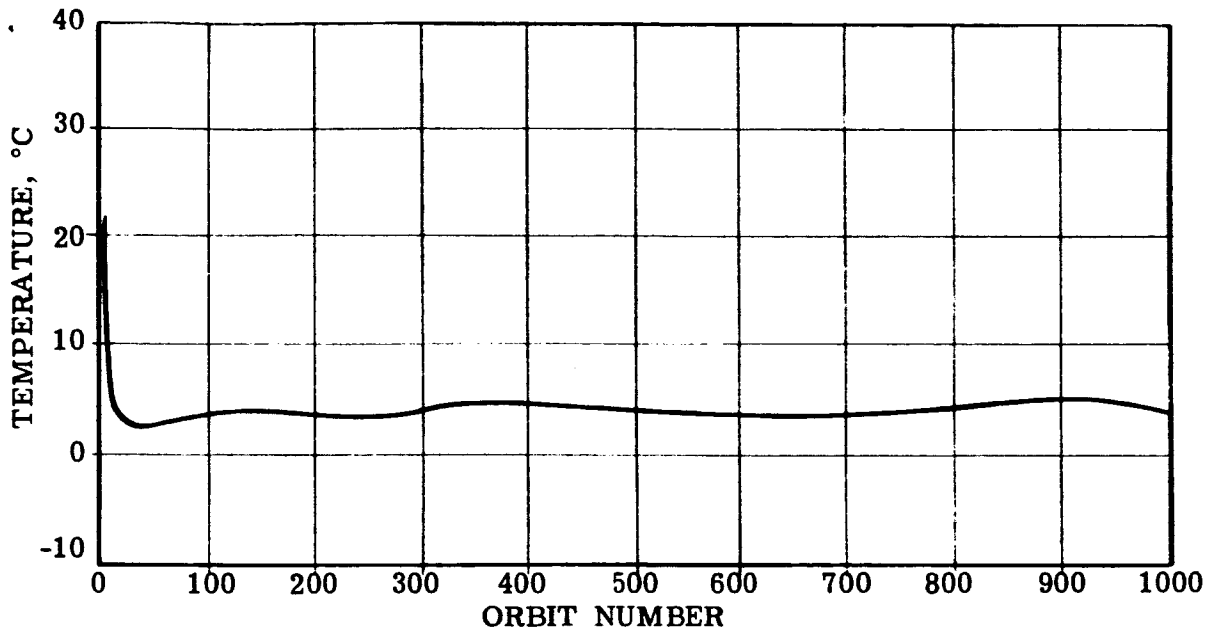


FIGURE B-3.—Battery No. 3 temperature monitor, channel 4.

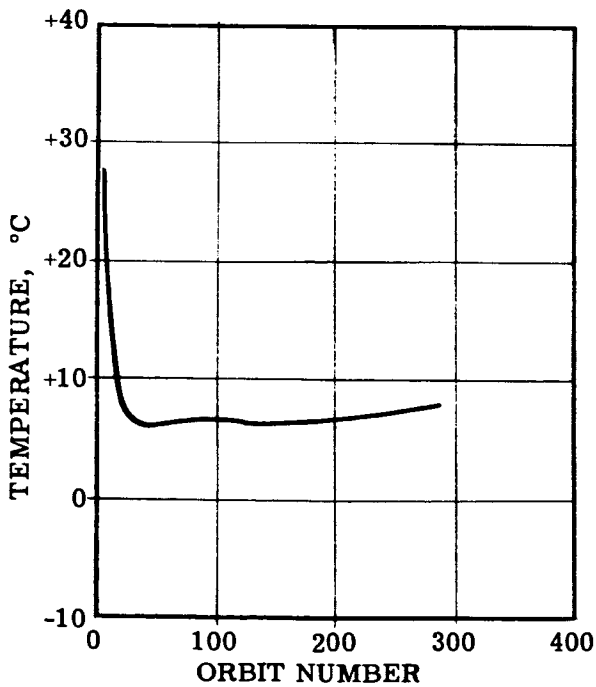


FIGURE B-4.—Battery No. 5 temperature monitor, channel 27.

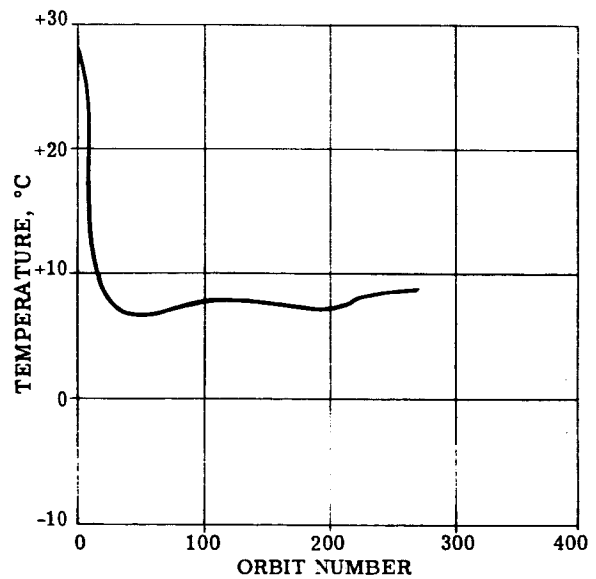


FIGURE B-5.—Battery No. 6 temperature monitor, channel 28.

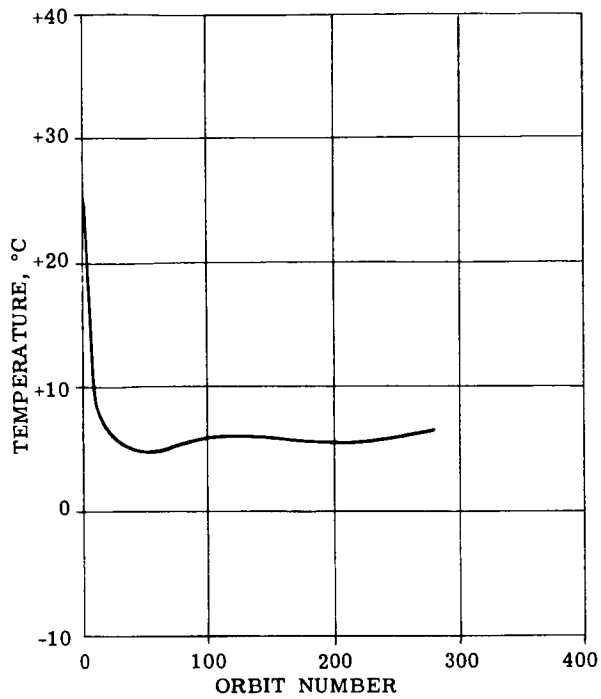


FIGURE B-6.—Battery No. 7 temperature monitor, channel 29.

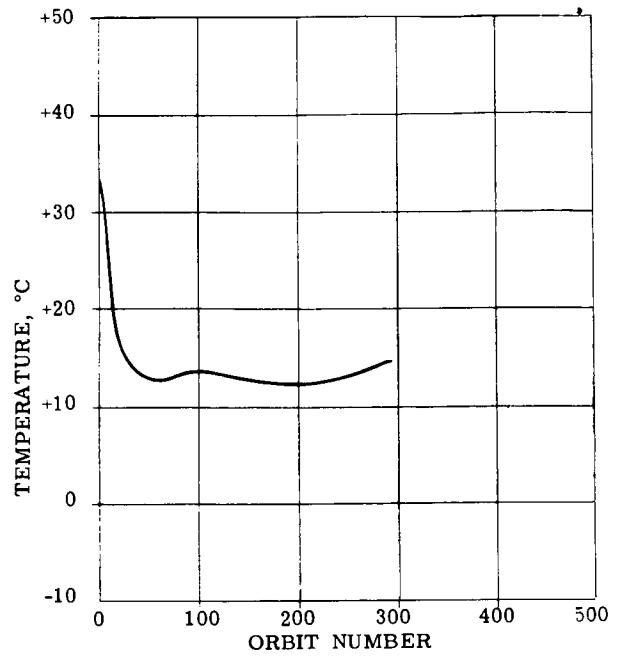


FIGURE B-8.—Bottom skin temperature monitor, channel 24.

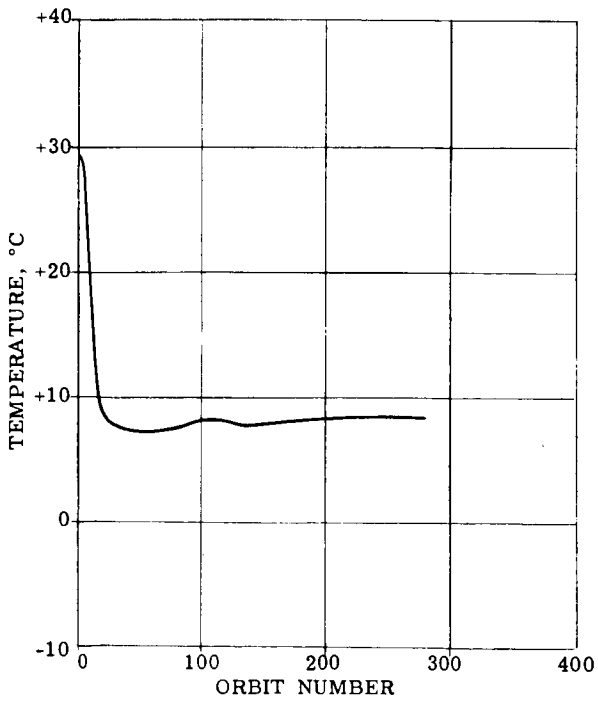


FIGURE B-7.—Top skin temperature monitor, channel 14.

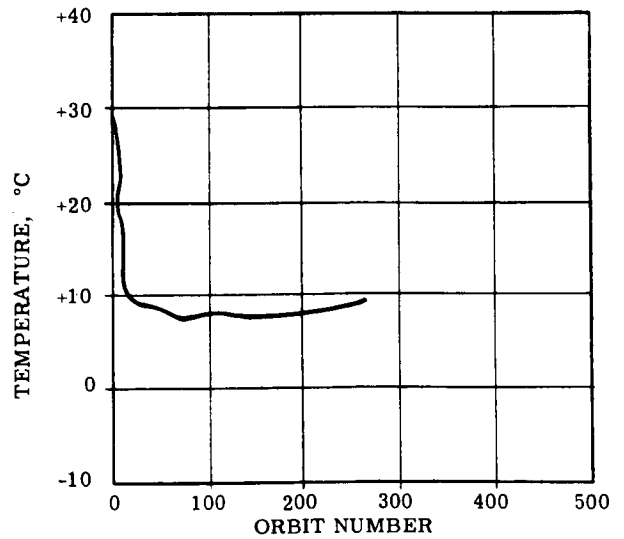


FIGURE B-9.—Rim temperature monitor, channel 15.

APPENDIX B—PASSIVE TEMPERATURE CONTROL

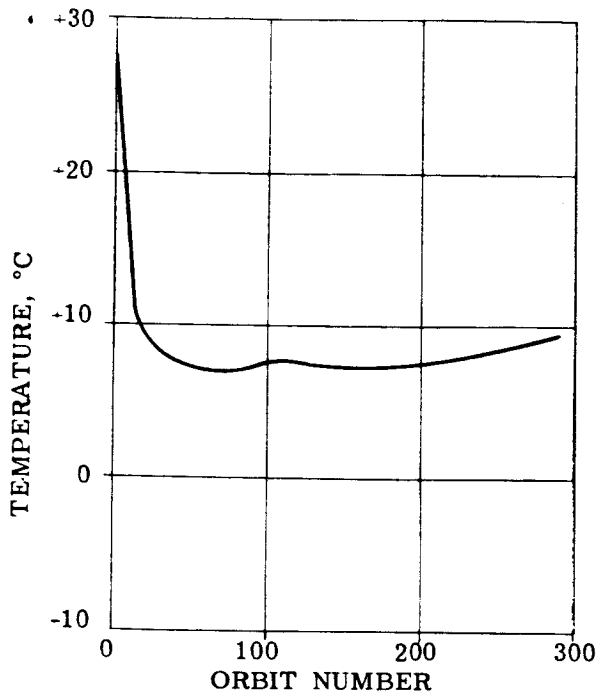


FIGURE B-10.—Hub temperature monitor, channel 26.

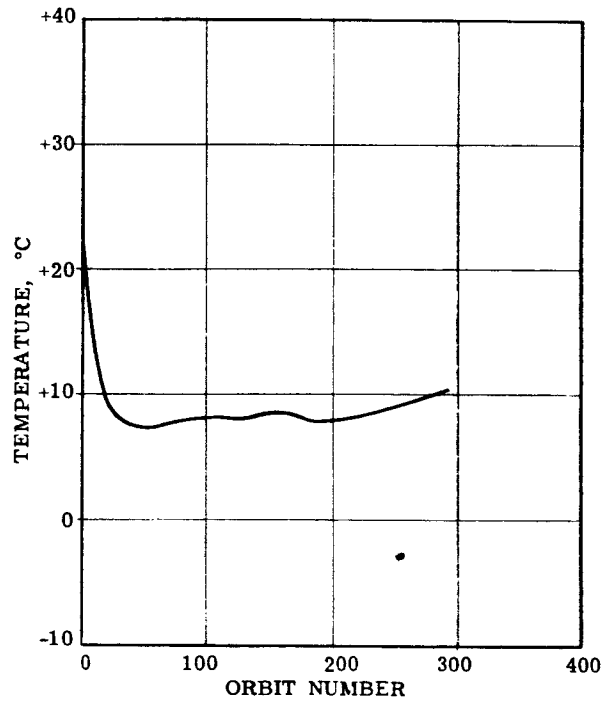


FIGURE B-11.—Tape recorder No. 1 temperature monitor, channel 10.

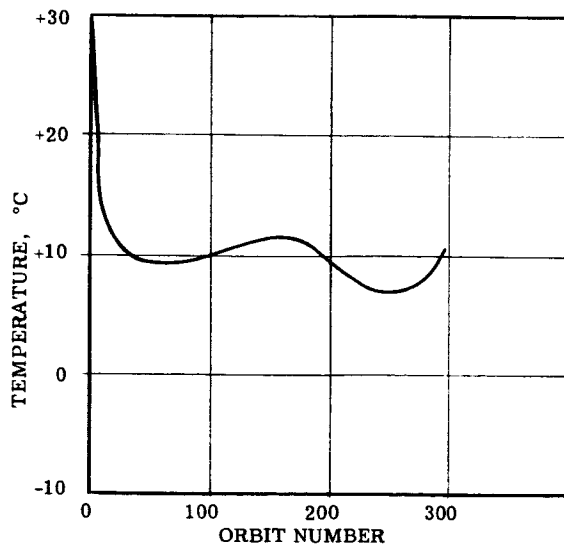


FIGURE B-12.—Transmitter temperature monitor, channel 30.

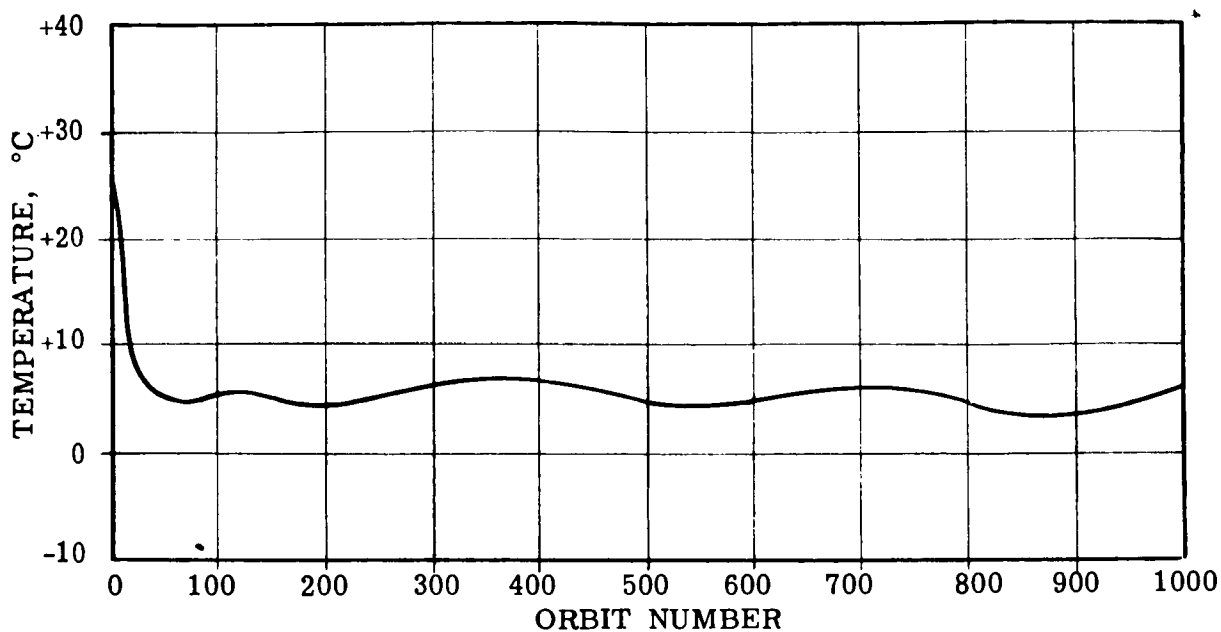


FIGURE B-13.—Azimuth casting temperature monitor, channel 47.

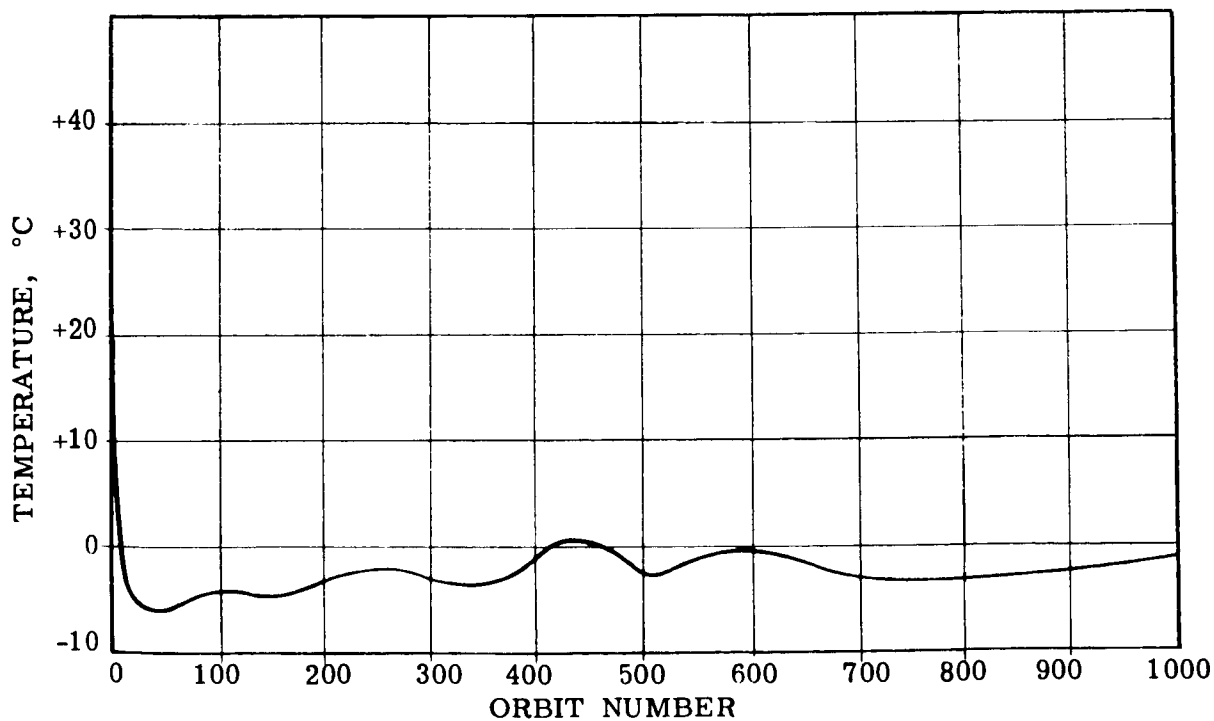


FIGURE B-14.—Power amplifier temperature monitor, channel 51.

APPENDIX B—PASSIVE TEMPERATURE CONTROL

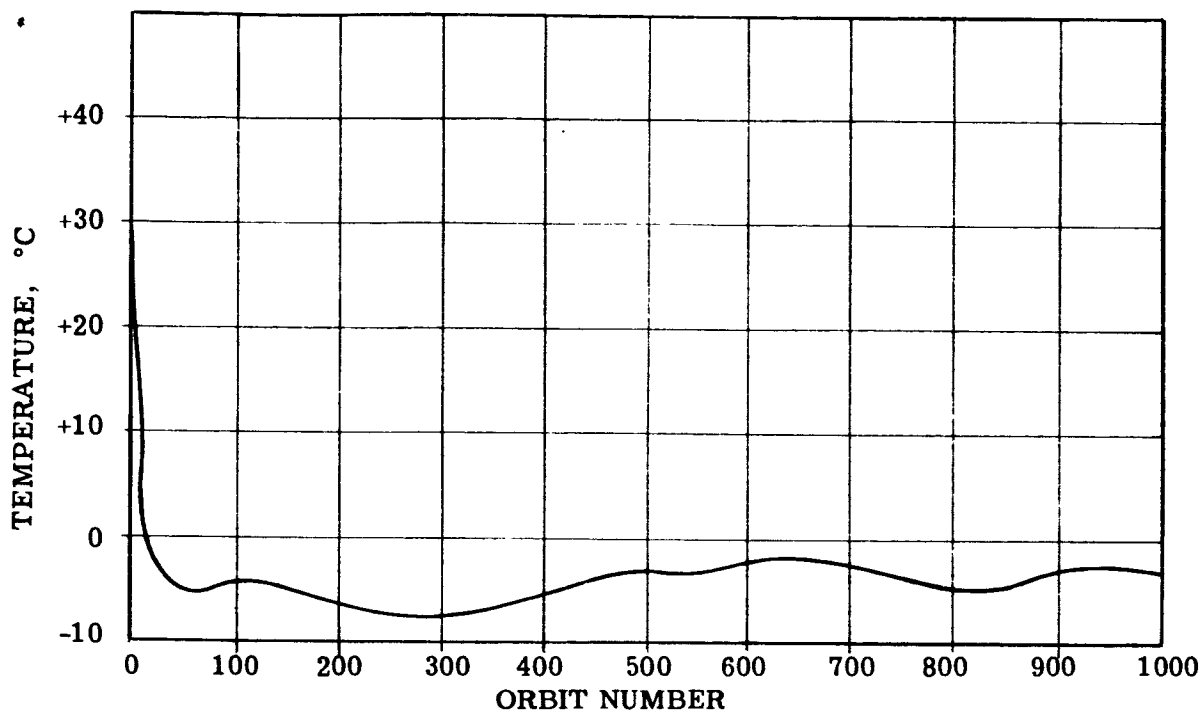


FIGURE B-15.—Linear amplifier temperature monitor, channel 50.

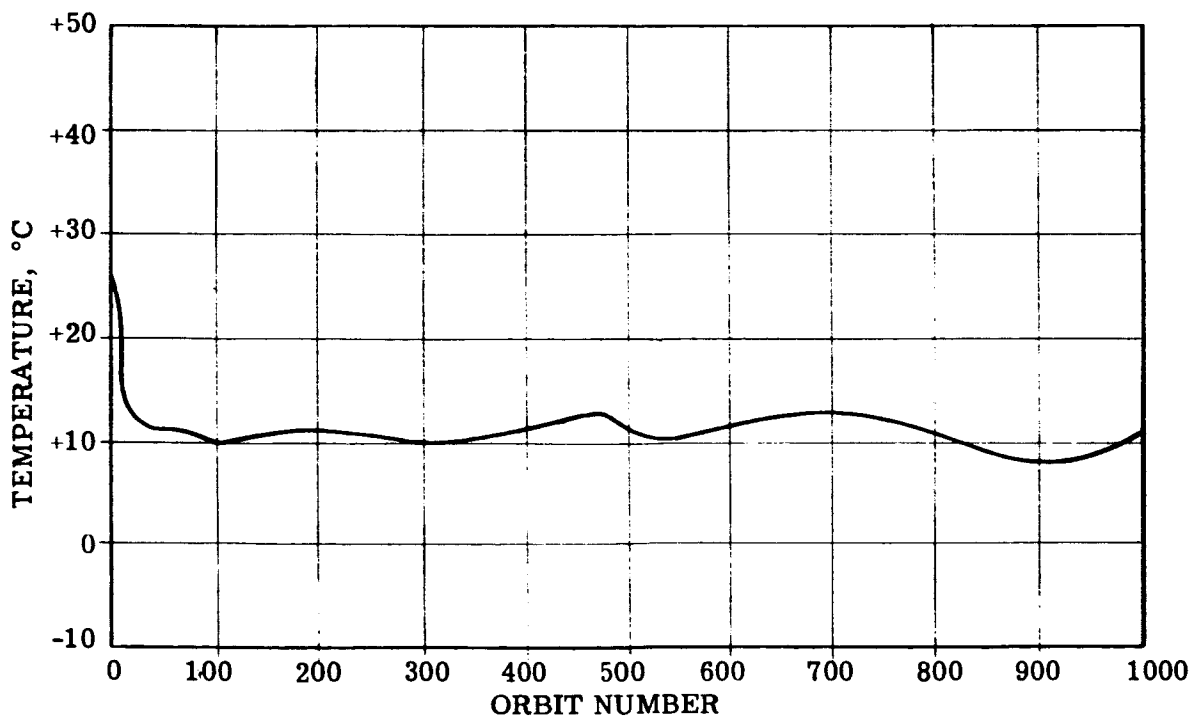


FIGURE B-16.—Goddard pointed instrument temperature monitor, channel 48.

ORBITING SOLAR OBSERVATORY

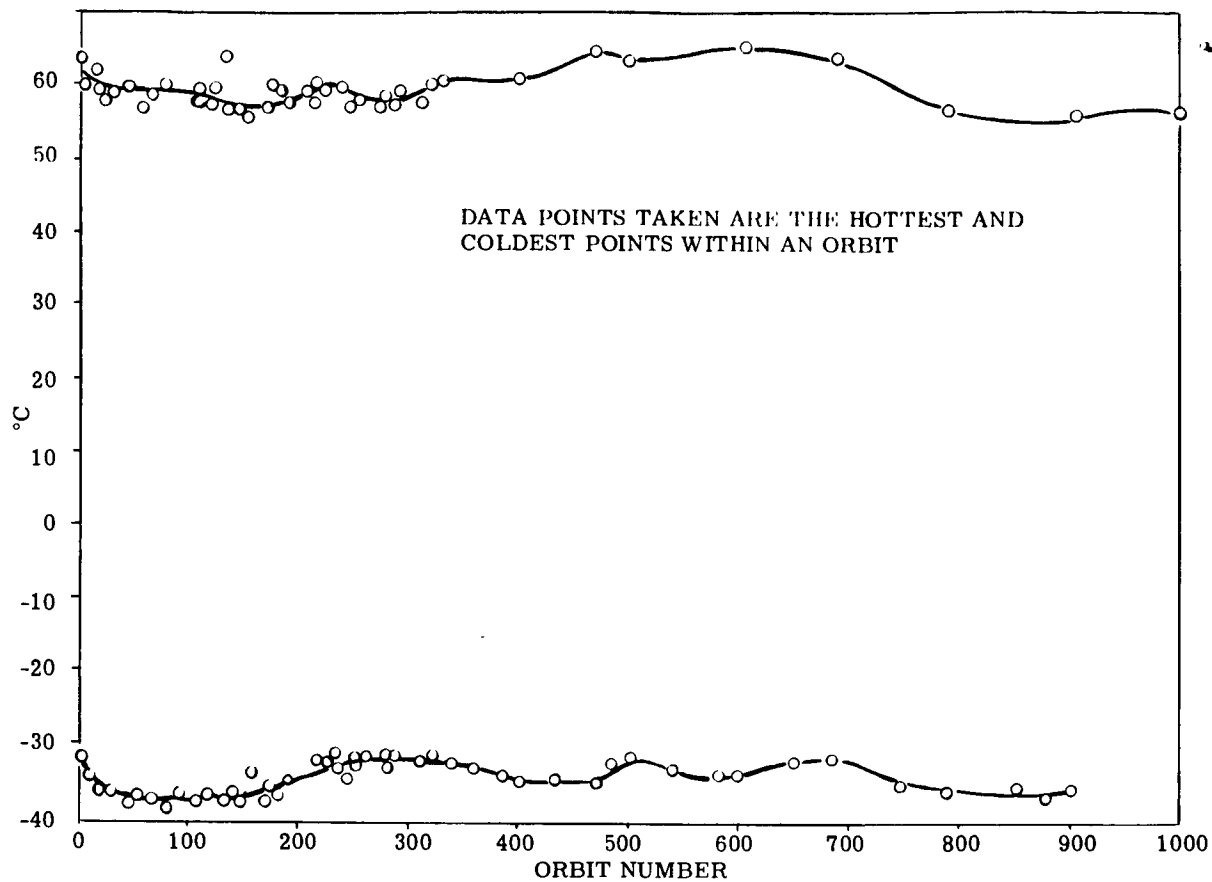


FIGURE B-17.—OSO I solar cell temperature monitor, channels 33 and 42.

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

NASA SCIENTIFIC AND TECHNICAL PUBLICATIONS

TECHNICAL REPORTS: Scientific and technical information considered important, complete, and a lasting contribution to existing knowledge.

TECHNICAL NOTES: Information less broad in scope but nevertheless of importance as a contribution to existing knowledge.

TECHNICAL MEMORANDUMS: Information receiving limited distribution because of preliminary data, security classification, or other reasons.

CONTRACTOR REPORTS: Technical information generated in connection with a NASA contract or grant and released under NASA auspices.

TECHNICAL TRANSLATIONS: Information published in a foreign language considered to merit NASA distribution in English.

SPECIAL PUBLICATIONS: Information derived from or of value to NASA activities. Publications include conference proceedings, monographs, data compilations, handbooks, sourcebooks, and special bibliographies.

TECHNOLOGY UTILIZATION PUBLICATIONS: Information on technology used by NASA that may be of particular interest in commercial and other nonaerospace applications. Publications include Tech Briefs; Technology Utilization Reports and Notes; and Technology Surveys.

Details on the availability of these publications may be obtained from:

SCIENTIFIC AND TECHNICAL INFORMATION DIVISION
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Washington, D.C. 20546