FINAL REPORT

OSO-6

orbiting solar observatory

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

BALL BROTHERS RESEARCH CORPORATION

SUBSIDIARY OF BALL CORPORATION

BOULDER, COLORADO

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NASA CR- *122*451

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PREPARED

OSO Project Staff

APPROVED

J. O. Simpson
Director,
OSO Programs
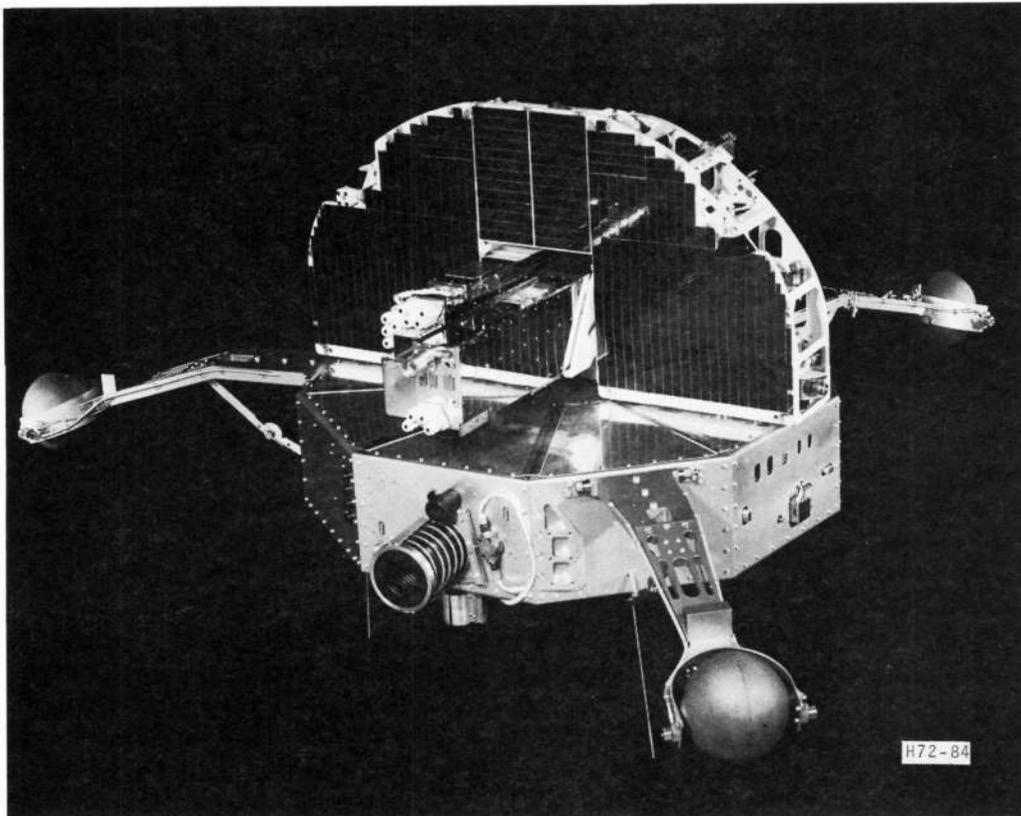
OSO-6 ORBITING SOLAR OBSERVATORY

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BALL BROTHERS RESEARCH CORPORATION
SUBSIDIARY OF BALL BROTHERS COMPANY INCORPORATED
BOULDER, COLORADO

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FOREWORD

The greatest obstacle in measuring solar and celestial radiation is the earth's atmosphere which acts as a shield and thus prevents earthbound observers from observing such phenomena. The Orbiting Solar Observatory (OSO) was conceived as a means of overcoming both this obstacle and the endurance limitations of earlier balloon-borne and rocket-borne experiments.

In 1958, Ball Brothers Research Corporation (BBRC) was awarded a feasibility-study contract by the Goddard Space Flight Center (GSFC) of the National Aeronautics and Space Administration (NASA) to investigate problems associated with designing and building such a satellite. The satellite would need to survive the rigors of launching and long-term space environment and to provide a stable platform from which scientific instruments could detect and measure cosmic phenomena.

Furthermore, it was necessary that the experiment capacity of the basic spacecraft be a flexible design to accommodate scientific experiments with differing configurations as governed by their mission objectives. Therefore, the OSO configuration differs slightly from model to model to accommodate different experiments and to reflect spacecraft design changes.

The OSO-1 observatory, launched 7 March 1962, from Cape Canaveral (now Cape Kennedy), Florida, was the outgrowth of the feasibility studies. This first OSO proved very successful and the OSO program was subsequently expanded to include a series of observatories.

The second OSO flight model (designated OSO-B) was built and delivered to the Eastern Test Range (ETR) facilities at Cape Kennedy in March, 1964. In April, accidental ignition of the third-stage motor resulted in extensive damage to the observatory. An accelerated refurbishing program was begun and OSO-B2 was built using some of the components salvaged from OSO-B1 and other available hardware.

OSO-B2 was successfully launched as OSO-2 from ETR on 3 February 1965. OSO-2 operated very successfully for nine months, at which time the mission was ended due to the depletion of its pitch-control gas.

OSO-C was delivered to ETR in July, 1965, and was launched on 25 August 1965. Due to the premature ignition of the Delta third-stage motor, however, the observatory was not placed into a successful orbit. In September of 1965, it was decided to modify the OSO-E spacecraft, then in production, to accommodate the OSO-C flight-spare experiments.

The modified OSO-E, renamed OSO-E1, was developed under an accelerated program, much like OSO-B2, and was delivered to ETR in February, 1967. This observatory was successfully launched on 8 March 1967, and was redesignated OSO-3. The OSO-3 performed nearly flawlessly for more than 3 years. The only spacecraft system problems during that period were the failure of the two tape recorders during the 3rd and 15th month of operation. OSO-3 continued to provide valuable scientific data on a realtime basis after the tape recorders failed.

OSO-D was successfully placed into orbit on 18 October 1967 and was redesignated OSO-4. It has also performed nearly perfectly except for the failure of its two tape recorders

during the fourth and sixth months of operation. At the time
of writing, OSO-4 continues to provide valuable scientific
data on a realtime basis.

OSO-F was successfully placed into orbit on 22 January 1969 and
was redesignated OSO-5. At the time of this writing, OSO-5
has had no failures and continues to provide valuable
scientific data.

OSO-G was successfully placed into orbit on 9 August 1969 and
redesignated OSO-6. This report describes this observatory,
its development history, and its orbital performance.

SUMMARY

This document is the final project report containing the description, development history, test history, and orbital performance analysis of the OSO-6 Orbiting Solar Observatory. The spacecraft contractor was Ball Brothers Research Corporation, Boulder, Colorado. The project was sponsored and directed by the Goddard Space Flight Center (GSFC) division of the National Aeronautics and Space Administration (NASA). The work was performed under contract number NAS5-9300.

The OSO-6 Orbiting Solar Observatory was the sixth flight model of a series of scientific spacecraft designed to provide a stable platform for experiments engaged in the collection of solar and celestial radiation data. The design objective was 180 days of orbital operation.

The OSO-6 contained seven primary scientific experiments furnished by the following organizations: University of New Mexico, Los Alamos Scientific Laboratory, Harvard College Observatory, Rutgers State University of New Jersey, University of Bologna, University College of London, and the Naval Research Laboratory.

This observatory was successfully launched into a near-earth orbit on 9 August 1969 from the Eastern Test Range (ETR) facilities at Cape Kennedy, Florida. The launch vehicle was a McDonnell/Douglas Astronautics Company two-stage Delta DSV-3N.

OSO-6 has telemetered an enormous amount of very useful experiment and housekeeping data to GSFC ground stations. Observatory operation during the two-year reporting period was very successful except for some experiment instrument problems.

The in-orbit performance history and the development history are presented in Sections 2 and 3, respectively, of this report. A brief summary of the first year of spacecraft operation and experiment operation is presented in the two tables below and in Section 1.

OSO-6 EXPERIMENT AND INTERFACE ONE-YEAR OPERATIONAL SUMMARY

Instrument	Operation Status	SPACECRAFT/EXPERIMENT INTERFACE											
		Telemetry				Command				Thermal			
		Failed	Poor	Adequate	Excellent	Failed	Poor	Adequate	Excellent	Failed	Poor	Adequate	Excellent
Pointed Instruments													
NRL X-Ray	Sensitivity Problem in burst detector. Data collected.				X				X				X
HCO UV	Sensitivity degraded with time. Usable data collected.				X				X				X
Wheel Instruments													
UCL X-Ray	Two Ion Chambers have failed, other nine chambers satisfactory.				X				X				X
Bologna X-Ray	No Problems				X				X				X
Rutgers Zodiacal	Occasional logic circuit problems. Data satisfactory.				X				X				X
New Mex. Neutron	Occasional problems with one detector. Data Satisfactory.				X				X				X
LASL X-Ray	No Problems.				X				X				X

OSO-6 SPACECRAFT SYSTEMS ONE-YEAR PERFORMANCE SUMMARY

System or Subsystem	Operation Status	Failed	Poor	Adequate	Excellent
Telemetry					
Dig. Mult/Encoder #1	Continuous				X
Dig. Mult/Encoder #2	Operated when selected				X
Tape Recorder #1	Operated when selected				X
Tape Recorder #2	Operated when selected				X
Transmitter #1	Continuous				X
Transmitter #2	Operated when selected				X
Command					
Receiver #1	Continuous				X
Receiver #2	Continuous				X
Wheel Decoder #1	Continuous				X
Wheel Decoder #2	Continuous				X
Sail Decoder #1	Continuous				X
Sail Decoder #2	Continuous				X
Pointing Control	Continuous				X
Pitch Control	Continuous				X
Spin Control	Continuous				X
Aspect Measuring	Continuous				X
Nutation Control	Continuous				X
Launch Sequence	Operated during orbit "0"				X
Mass Balance	----			X*	
Thermal Control	----				X
Power Supply	Continuous				X

LEGEND:

Excellent = Significantly surpassed design specifications

Adequate = Performed near design specifications

Poor = Did not meet design specifications

Failed = Stopped functioning completely

*An apparent mass unbalance condition (wobble) was observed in the pitch data received from OSO-6. The wobble amounted to about 0.2 deg P-P but did not adversely effect the operation of either the spacecraft systems or the experiment instruments.

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ILLUSTRATIONS

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TABLES

Section 1
PROJECT HIGHLIGHTS

1.1   INTRODUCTION

Brief descriptions of the OSO-6 mission objectives, develop-
ment history, in-orbit performance, spacecraft and scientific
instrument systems, and the Delta launch vehicle are presented
in this section. Detailed descriptions of these topics are
presented in other sections of this final report. Also,
material of a detailed nature, related to but not included
in this report, is appropriately referenced.

In this report, the term "observatory" refers to the entire
payload which includes the basic spacecraft and experiment
complement. The term "spacecraft" refers only to the basic
spacecraft portion and its systems excluding the scientific
experiments. A list of the abbreviations used in this
report is presented in Appendix A. A bibliography of source
material and related documents is presented in Appendix B. A
concise listing of the spacecraft systems and major components
(including their functions) is given in Appendix C.

1.2   MISSION DESCRIPTION

1.2.1   Why OSO is Needed

The sun continually emits electromagnetic radiation of greatly
varying intensities across a broad spectrum. The earth's at-
mosphere acts as a shield and absorbs the major portion of this
energy, making it difficult, and in some cases impossible, for
ground-based observers to evaluate the detailed characteristics
of solar radiation. Thus, the primary objective of the Orbit-
ing Solar Observatory is to gather such data from its vantage

point above the earth's atmosphere, and telemeter this infor-
mation to earth ground stations. OSO-1 through OSO-5 have
contributed significantly to our knowledge of solar and celes-
tial emissions. OSO-6 continued these objectives by provid-
ing the participating experimenters with additional data.

1.2.2 • Why the Cosmic Radiation Region is Studied

Although the sun's total energy output (of which over 99 per-
cent is in the visible, UV, and IR light regions[1]) varies less
than one percent, the energy of the low-frequency (radio
frequency), particulate, and the extremely high-frequency
(cosmic) radiation is known to vary enormously (see Table 1-1).
These regions of highly variable but of relatively low energy
emissions seem to have the greatest influence on earth's
geophysical nature. The low frequency RF emissions have been
extensively investigated by ground-based radio telescopes.
However, the extremely high frequency emissions consisting of
ultraviolet, X-rays, and gamma rays are almost entirely blocked
by the earth's ionosphere and atmosphere and, therefore, cannot
be observed from the ground. Furthermore, it is believed that
these extremely high frequency emissions hold important secrets
of the sun's processes.

To summarize, there are two primary reasons for studying the
sun's processes: (1) that a greater knowledge of these pro-
cesses will help in understanding the sun-earth relationship,
i.e., the effect the sun has on geophysical processes such as
weather, geomagnetic storms, and radiowave propagation; and
(2) that by understanding the sun's natural processes, we may
find the answers to other celestial processes and phenomena.

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[1]The spectrum in these regions is very much like the spectrum
of a 6000°K "black body".

Table 1-1

SUN'S EMISSIONS

Emission	Wavelength*	Energy at Earth in ergs/cm²-sec	Magnitude of Variations
RF Waves	10^4 m - 1 mm	3×10^{-7}	$\times 10^3$
Infrared	1 mm - 9000 Å	3×10^4	very small
Photographic	9000 to 3000 Å	1.4×10^6	very small
UV	3000 to 1500 Å	2×10^4	very small
Lyman-Alpha UV	1216 Å	5 to 10	$\times 3$
EUV	1500 to 300 Å	1	$\times 100$
XUV	1500 to 1 Å		
X-rays	300 Å - 1 Å	10^{-3} to 10^{-4}	$\times 10^3$
High Energy Spectrum	<1 Å, >10 Kev	uncertain	$\times 10^6$

* These bandwidths can not be precisely defined and there is
   generally considerable overlap.

1.2.3   Mission Objectives

The primary[1] objective of the OSO-6 mission was to launch and
place into orbit a spacecraft carrying solar physics experiments.
These experiments were to detect and measure electro-magnetic

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[1] A secondary objective for the launch vehicle was to place an
experimental Packaged Attitude Control (PAC) into orbit. The
purpose of the PAC mission was to flight test an earth stabilized
control system which might be used in the future for adapting the
Delta second stage for a wide variety of piggyback payloads.

radiation in selected areas of the spectrum, from the sun and other sources.

Specific mission objectives were:

- To study solar radiation in the spectral range of 300 to 1300 $\overset{\circ}{A}$ from localized regions

- To perform spectral, burst, and mapping measurements of solar X-rays within the range of 0.2 to 28 $\overset{\circ}{A}$

- To monitor photon flux in six selected spectral lines within the range of 16 to 40 $\overset{\circ}{A}$

- To measure the ellipticity, brightness, and polarization of zodiacal light

- To monitor solar X-rays, celestial X-rays, and the terrestrial X-ray albedo within the range of 20 to 200 kev

- To determine the energy spectrum and angular distribution of high-energy neutron flux within the range of 20 to 130 Mev

- To study solar resonance lines and optically thin lines of helium I, helium II, iron XI, oxygen, and nitrogen

- To study F-layer ionization and flares

In Fig. 1-1, we have superimposed the individual experiment "windows" on the sun's electromagnetic spectrum to show the areas investigated by the OSO-6 experiments. Summary descriptions of the spacecraft and its experiments are given in Figs. 1-2

through 1-9. Note that the NRL X-Ray Spectroheliograph and the HCO Ultraviolet Spectroheliograph instruments were located in the solar-oriented part of the observatory and were concerned only with solar phenomena. The other instruments (the "wheel" experiments) were located in the rotating section of the observatory. These instruments swept the celestial sphere and viewed its components (including earth and sun).

Detailed descriptions of the experiments are presented in BBRC Document TN69-04, OSO-G Experiment Payload Description, issued 16 June 1969.

1.3   SCHEDULE HIGHLIGHTS

Milestone dates for the OSO-G/OSO-6 program are listed in Table 1-2.

1.4   OBSERVATORY IN-ORBIT PERFORMANCE SUMMARY

The OSO-6 Orbiting Solar Observatory has, at the time of this writing, operated continuously for more than two years, making over 12,000 revolutions of the earth. During this time, every mission requirement has been met or exceeded, with the exception of some scientific experiment malfunctions which reduced the ability of the observatory to completely fulfill its scientific objectives.

Spacecraft systems performance is presented in some detail in Section 2 of this report. Comparison of performance data with the criteria defined in the OSO-G Model Specification, BBRC Document 26493 (See Appendix D), shows that OSO-6 greatly exceeded the 6-month minimum performance requirements during the first two years of in-orbit operation.
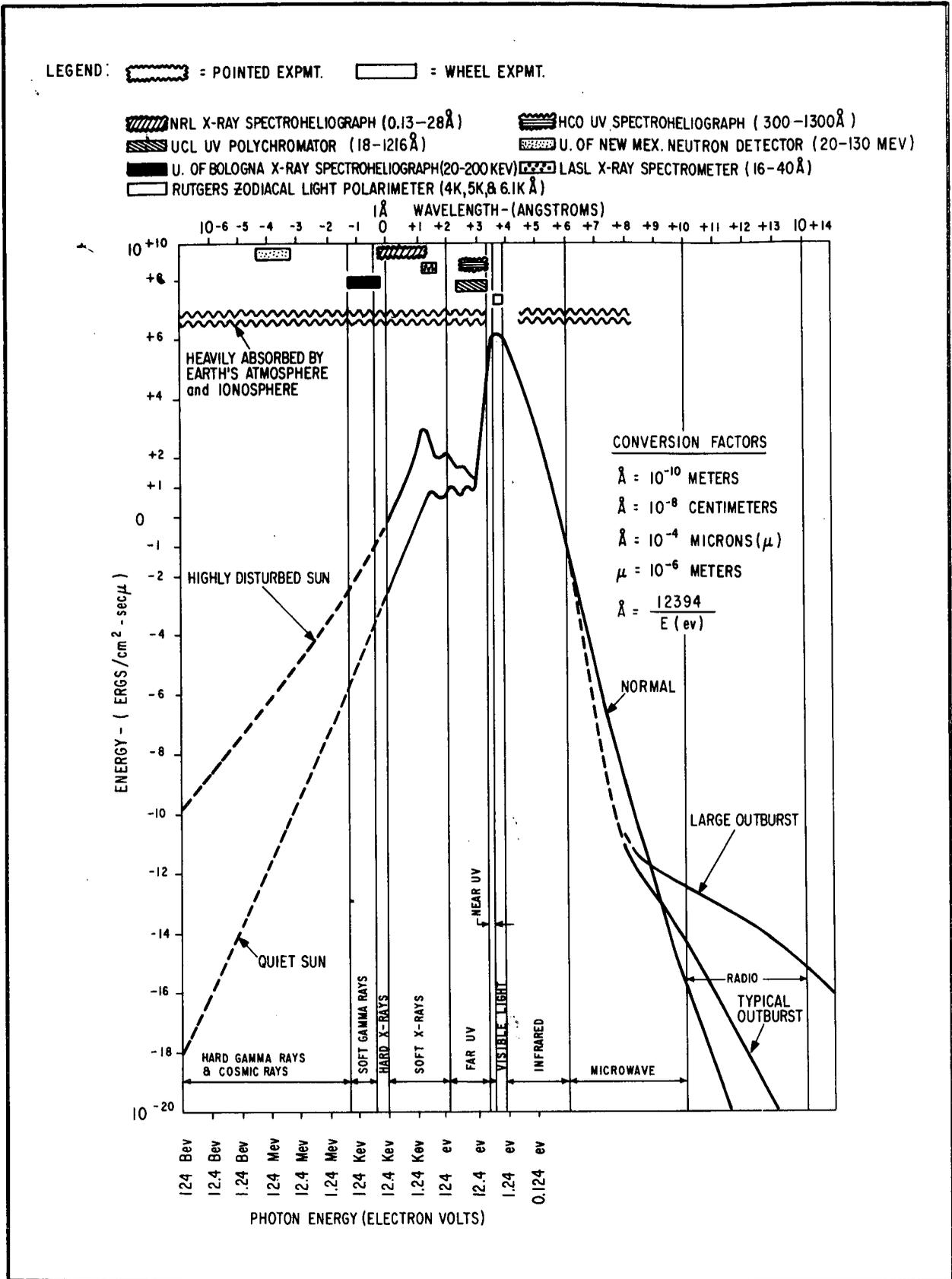
Fig. 1-1 OSO-6 Experiment Windows
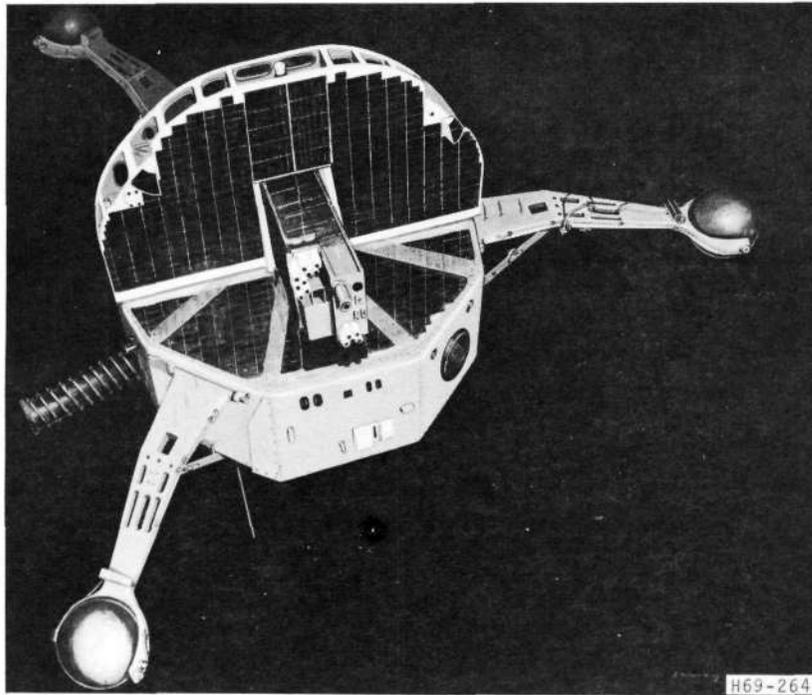
PROGRAM SUMMARY

PROJECT *ORBITING SOLAR OBSERVATORY (OSO-6)*
PROJECT DIRECTION *NASA/GSFC*
PRIME CONTRACTOR *BALL BROTHERS RESEARCH CORP.*
EXPERIMENT ORGANIZATIONS *U.S. NAVAL RESEARCH LABORATORY / UNIVERSITY COLLEGE LONDON / HARVARD COLLEGE OBSERVATORY / UNIVERSITY OF BOLOGNA / LOS ALAMOS SCIENTIFIC LABORATORY / UNIVERSITY OF NEW MEXICO / RUTGERS STATE UNIVERSITY*
MAJOR OBJECTIVES *THE OSO IS A RESEARCH SPACECRAFT DESIGNED FOR MEASUREMENT OF SOLAR PHENOMENA AND SOLAR EFFECTS. THE VARIATION IN ATTITUDE OF THE OBSERVATORY DURING ITS LIFETIME WILL ALSO PERMIT OBSERVATION OF THE ENTIRE CELESTIAL SPHERE AND PORTIONS OF THE EARTH'S SURFACE*
LIFETIME *ESTIMATED: SIX MONTHS*
LAUNCH VEHICLE *DELTA N*
CIRCULAR ORBIT *300 ± 30 NAUTICAL MILES*
INCLINATION *33 ± 3 DEGREES*
NOMINAL PERIOD *NIGHT 36 MIN - DAY 60 MIN. TOTAL 96 MIN.*
TOTAL OBSERVATORY WEIGHT *ESTIMATED 290 KG (639.9 LB.)*
POINTING ACCURACY *NOT MORE THAN ONE MINUTE OF ARC*
POINT MODE *STATIONARY POINTING AT CENTER OF SOLAR DISC*
LARGE RASTER MODE *46 BY 46 MINUTES OF ARC*
SMALL RASTER MODE *7 BY 7½ MINUTES OF ARC*
OFFSET POINT MODE *16384 POINTS WITHIN 46 BY 46 ARC MINUTE PATTERN*

H69-264

Fig. 1-2   OSO-6 Mission Description

Naval Research Laboratory- X-Ray Spectroheliograph
Principal Investigators: *Dr. T. Chubb and Mr. R.W. Kreplin*

Objectives : *Measure the solar x-ray energy and spatial distribution in the spectral region of 0.13 to 28 Å*

Description : *Consists of four individual experiments: (1) pulse height analyzer, (2) Bragg spectrometer, (3) raster experiment, and (4) burst experiment.*
The pulse height analyzer uses a sodium iodide scintillator and a proportional counter to monitor rapid energy changes in the 2 to 90 kev (0.14 to 6 Å) region. The scintillator covers the high energy region of 20 to 90 kev and the PC covers the low energy region of 2 to 20 kev.

The Bragg spectrometer uses three crystal spectrometers to provide fine spectral resolution through the soft x-ray region of 0.6 to 25 Å. The ranges of the spectrometers are: 0.6 to 4 Å, 1.4 to 8.4 Å, and 1.8 to 25 Å. The photon flux passed by each crystal is measured by a geiger counter.

The raster experiment consists of three geiger counter detectors. One determines the photon spatial distribution in the 8 to 20 Å region. The second pinpoints hot spots in the 1 to 8 Å region. The third detects and measures the lateral velocity of flares in the 8 to 20 Å region.

The burst experiment uses two geiger counter detectors to monitor the solar disc for very short duration bursts in the 18 to 25 Å region and to detect cosmic background radiation in the 2 to 8 Å region.   H72-85

Fig. 1-3   NRL X-Ray Pointed Instrument

**Harvard College Observatory –
Solar Ultraviolet Spectroheliograph**

Principal Investigator: Dr. L. Goldberg

Objectives: *Measure the absolute intensity of the solar emission lines and continua within the wavelength range of 300-1300 Å and map the spatial distribution of this intensity.*

Description: *Consists of a telescope and a normal incidence spectrometer equipped with a concave diffraction grating and a windowless photomultiplier. By ground command the spectrometer, whose bandpass is approximately 3 Å, can be scanned or set to any one of ≈ 11,000 discrete wavelength positions which are separated by 0.1 Å. The light admitted to the spectrometer originates only from a small area on the sun (equivalent to a square of ½ by ½ arc minutes) whose location depends on the direction where the instrument is pointed. Spectra from any location on the sun can be obtained by a spectrometer scan while the instrument is in the offset pointing mode. Operation in one of the raster modes and with a fixed wavelength results in spectroheliograms being generated.*

H72-87

Fig. 1-4   HCO Ultraviolet Pointed Instrument

University College, London – Ultraviolet Polychromator

*Principal Investigators: Prof. R.L.F. Boyd
and Dr. B. Woodgate*

Objective: *Measure specific emission lines in the ultraviolet region of 18 to 1,216 Å.*

Description: *Consists of a grating spectrometer and eleven channel electron multipliers (CEM's). Light enters and impinges on a diffraction grating and then passes through nine slits placed on a Rowland circle at positions corresponding to nine of the emission lines that are to be observed. The light intensity passing through the slits is then measured by nine CEM's. The other two CEM's are used to measure scattered light behind the spectrometer and background noise from high energy electrons.*

H72-86

Fig. 1-5   UCL Ultraviolet Wheel Instrument

Rutgers State University - Zodiacal Light Polarimeter

Principal Investigator: Dr. A.L. Rouy

Objectives: *Measure Zodiacal light intensity, the degree and axis of polarization and the amount and nature of any elliptically polarized light at wavelengths of 4,000, 5,000 and 6,100 Å.*

Description: *Consists of a collimating telescope, quartz rotators, retardation plates, polarizing prism, monochromating filters and photomultiplier. Zodiacal light (which is polarized) is first collected and collimated by the telescope with a look-angle of less than two degrees. The telescope is protected from stray and scattered light by a sun shield that extends out from the rim panel. The light passes through an optical system in which the polarization axis of the light is rotated in increments of 45°. The light then passes through a Rouy prism and the resulting intensities are measured by the photomultiplier at the various angles. Additional measurements are made when the light in one axis is retarded in phase by the retardation plate. This data is then used to calculate the degree and angle of polarization of the zodiacal light.*
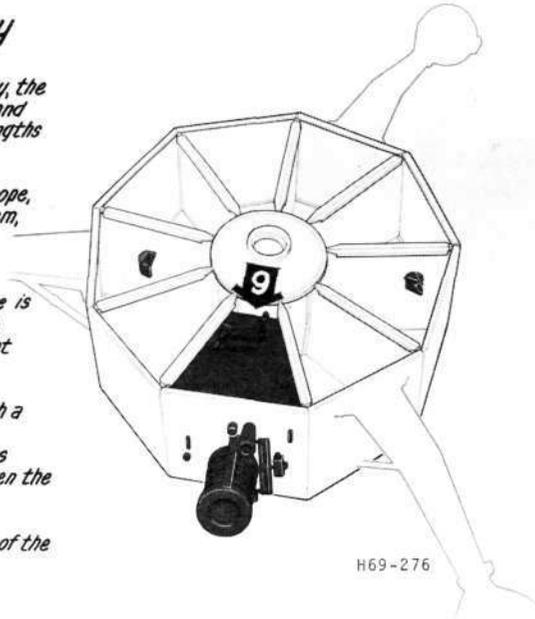
H69-276

Fig. 1-6　Rutgers Zodiacal Light Wheel Instrument
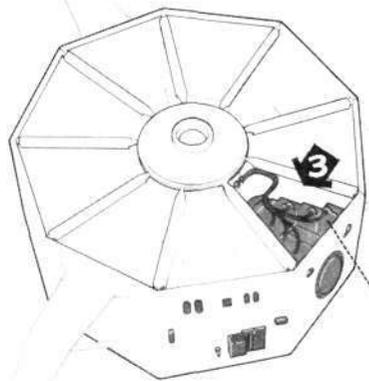
University of Bologna Solar X-Ray Spectroheliograph

Principal Investigator: Prof. D. Brini

Objectives: *Measure the solar hard x-ray energy, celestial spatial distribution, and earth albedo in the spectral region of 20 to 200 kev.*

Description: *Consists of a sodium iodide scintillator detector and a photomultiplier. The scintillator converts entering x-ray protons into light pulses. The photomultiplier and logic circuits then convert these light pulses into electrical pulses in four ranges corresponding to photon energy ranges of 20 to 40, 40 to 70, 70 to 110 and 110 to 200 kev.*

H69-259

Fig. 1-7　Bologna X-Ray Wheel Instrument

University of New Mexico - Neutron Detector

Principal Investigator: Dr. C.P. Leavitt

Objectives: Measure the flux, direction and energy of neutrons in the 20 to 130 mev region that originate in the earth's atmosphere and detect and measure neutrons from solar sources.

Description: Consists of a "Proton recoil telescope" which is composed of four telescope scintillators surrounded by a guard counter and which utilizes a total of eight photomultipliers. Proton recoils with energies between 20 and 130 mev are detected and the pulse heights in all the telescope scintillators are recorded for each event. In the absence of neutron events, charged particle events are recorded for calibration purposes. The effective directional resolution is about 45 degrees. The azimuthal look angle with respect to the solar vector is determined for each event by the use of either the sun pulse or the magnetometer pulse.

H72-88

Fig. 1-8 UNM Neutron Wheel Instrument

Los Alamos Scientific Laboratory X-Ray Spectrometer

Principal Investigator: Dr. H.V. Argo

Objective: Measure specific emission lines in the X-ray region of 16 to 40 Å.

Description: Consists of six fixed-crystal spectrometers which measure the intensity of 6 specific emission lines between 16 and 40 Å. Three mica crystals are used for short wavelengths and three lead stearate crystals are used for longer wavelengths. Each diffracting crystal is installed in a Johan-type spectrometer and spectral resolution is obtained by using the Bragg principle. Photon counts from each spectrometer are stored and read out about every 2.5 seconds.

H69-258

Fig. 1-9 LASL X-Ray Wheel Instrument

Table 1-2

OSO-6 MILESTONE DATES

Date	Milestone Event
7 Oct 65	Initial Experimenters' Meeting
31 Oct 66	Begin Design Review and Production Activities
26 Jun 68	Wheel Section Delivered from Production to Integration and Test Facility
26 Jul 68	Sail Section Delivered from Production to Integration and Test Facility
5 Oct 68	Mate Sail Section to Wheel Section
29 Nov 68	Spacecraft 90-percent Acceptance
23 Mar 69	Final 10-percent Spacecraft Acceptance
9 Apr 69	Experiment Instrument Integration Starts
24 May 69	Observatory Vibration Testing Starts
4 Jun 69	Observatory Thermal-Vacuum Testing Starts
17 Jul 69	Observatory Acceptance/Shipment to ETR
9 Aug 69	Launch

Throughout the observatory lifetime, hundreds of outbursts of solar activity were monitored by OSO-6's scientific instruments. A great amount of information on solar and nonsolar gamma radiation, ultraviolet radiation, and X-ray radiation sources and intensities and charged-particle spectra has been obtained. This data included the solar eclipse of 7 March 1970.

Rigorous treatment of experiment performance, which involves experimenter proprietary information, is left to the prerogative of the particular scientists involved.

1.5   LAUNCH VEHICLE DESCRIPTION AND LAUNCH SUMMARY

1.5.1   Launch Vehicle Description

The OSO-6 spacecraft was launched from Cape Kennedy, Florida, by a launch vehicle designated Delta-72.

Delta-72 was a two-stage DSV-3N launch vehicle (Fig. 1-10) for which the McDonnell Douglas Astronautics Company (MDAC) is prime contractor. The first stage (S/N 20012) was a modified (long tank), liquid propellant Thor booster powered by an MB-3 Block III Rocketdyne engine system rated at 172,000 pounds of thrust at sea level. This stage used RJ-1 as the fuel and liquid oxygen (LOX) as the oxidizer. It burned to propellant depletion in about 220 seconds. Pitch and yaw control was provided by gimbaling the main engine in response to signals from an inertial reference package. Two small vernier engines provided roll control.

The first stage was thrust-augmented by three strapped-on Thiokol TX354-5 solid propellant motors (S/Ns 93, 143, and 162) each rated at about 52,000 pounds of thrust and having a nominal burn time of 37 seconds. These motors burned to completion and were jettisoned 70 seconds after liftoff. This combination of the main engine and solids plus two vernier engines provided a liftoff thrust of about 330,000 pounds.

The second stage (S/N 20239) was an Aerojet General AJ10-118E liquid propellant engine system rated at 7,800 pounds average thrust in a vacuum. This engine used unsymmetrical dimethyl-hydrazine (UDMH) as the fuel and inhibited red nitric acid (IRFNA) as the oxidizer. It's burn time was controlled from the ground and was about 600 seconds. Attitude control during the powered portion of the second-stage flight was accomplished in the same manner as the first stage except that the thrust for roll correction was provided by cold-gas thrusters. The second-stage guidance compartment housed the flight control
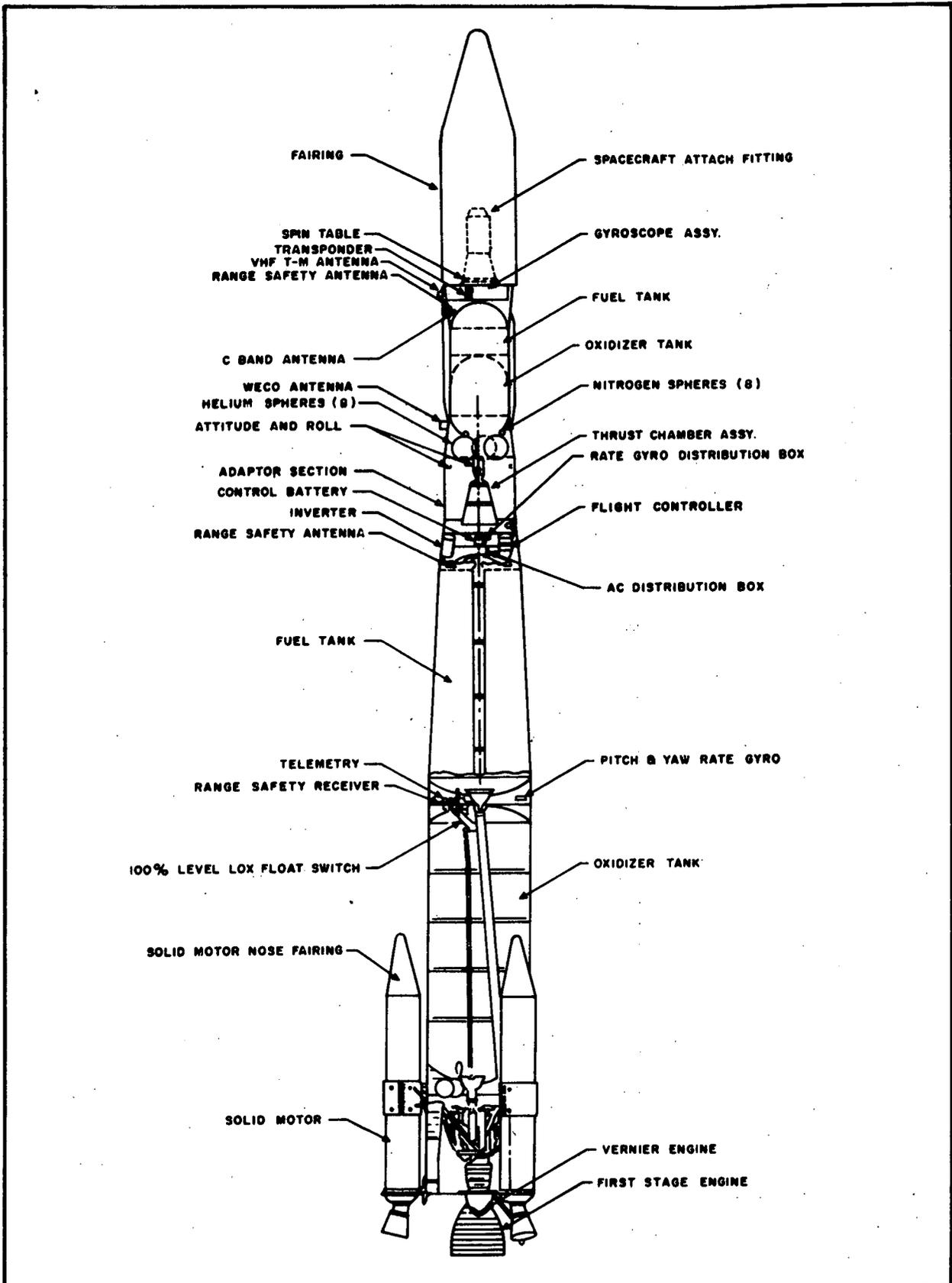
Fig. 1-10 Delta-N Launch Vehicle

FAIRING

SPACECRAFT ATTACH FITTING

SPIN TABLE
TRANSPONDER
VHF T-M ANTENNA
RANGE SAFETY ANTENNA

GYROSCOPE ASSY.

FUEL TANK

OXIDIZER TANK

C BAND ANTENNA

WECO ANTENNA
HELIUM SPHERES (8)

NITROGEN SPHERES (8)

ATTITUDE AND ROLL

THRUST CHAMBER ASSY.

RATE GYRO DISTRIBUTION BOX

ADAPTOR SECTION
CONTROL BATTERY
INVERTER
RANGE SAFETY ANTENNA

FLIGHT CONTROLLER

AC DISTRIBUTION BOX

FUEL TANK

TELEMETRY
RANGE SAFETY RECEIVER

PITCH & YAW RATE GYRO

100% LEVEL LOX FLOAT SWITCH

OXIDIZER TANK

SOLID MOTOR NOSE FAIRING

SOLID MOTOR

VERNIER ENGINE

FIRST STAGE ENGINE

system, the radio guidance system, the velocity cut-off system instrumentation, the radio destruct system, and tracking and power systems.

The OSO was attached to the second stage by an interstage structure which included a spin table (Fig. 1-11). The purpose of the spin table was to spin-stabilize the OSO (at about 100 rpm) just before it was separated from the launch vehicle. Spinup was provided by several small rockets located around the perimeter of the spin table. Separation was accomplished by the vehicle timer which fired bolt-cutting squibs which released the retaining bands that attached OSO to the adapter. A small velocity increment was imparted to OSO by a helical spring that was compressed between the OSO attach fitting and the third-stage adapter. At separation OSO's separation switches closed and power was applied to the OSO launch sequence timers (see Paragraph 1.6.6).

The first and second stages were guided by two separate pre-programmed autopilot systems. The Western Electric Company (WECO) provided a ground-based radio guidance system to make any necessary trajectory changes during first and second stage powered flight. The communications system of the launch vehicle monitored functional and environmental conditions of the vehicle. The data was transmitted to the ETR ground stations through VHF FM/FM telemetry equipment. Tracking was accomplished through a C-band transponder which responds to interrogation from ground radar sites.

1.5.2   Launch Summary

The Delta-72 vehicle and OSO-6 spacecraft were launched from Pad 17A, Cape Kennedy Air Force Station, Florida, at the scheduled time of 0352:00.1 EDT on 9 August 1969 (Fig. 1-12).
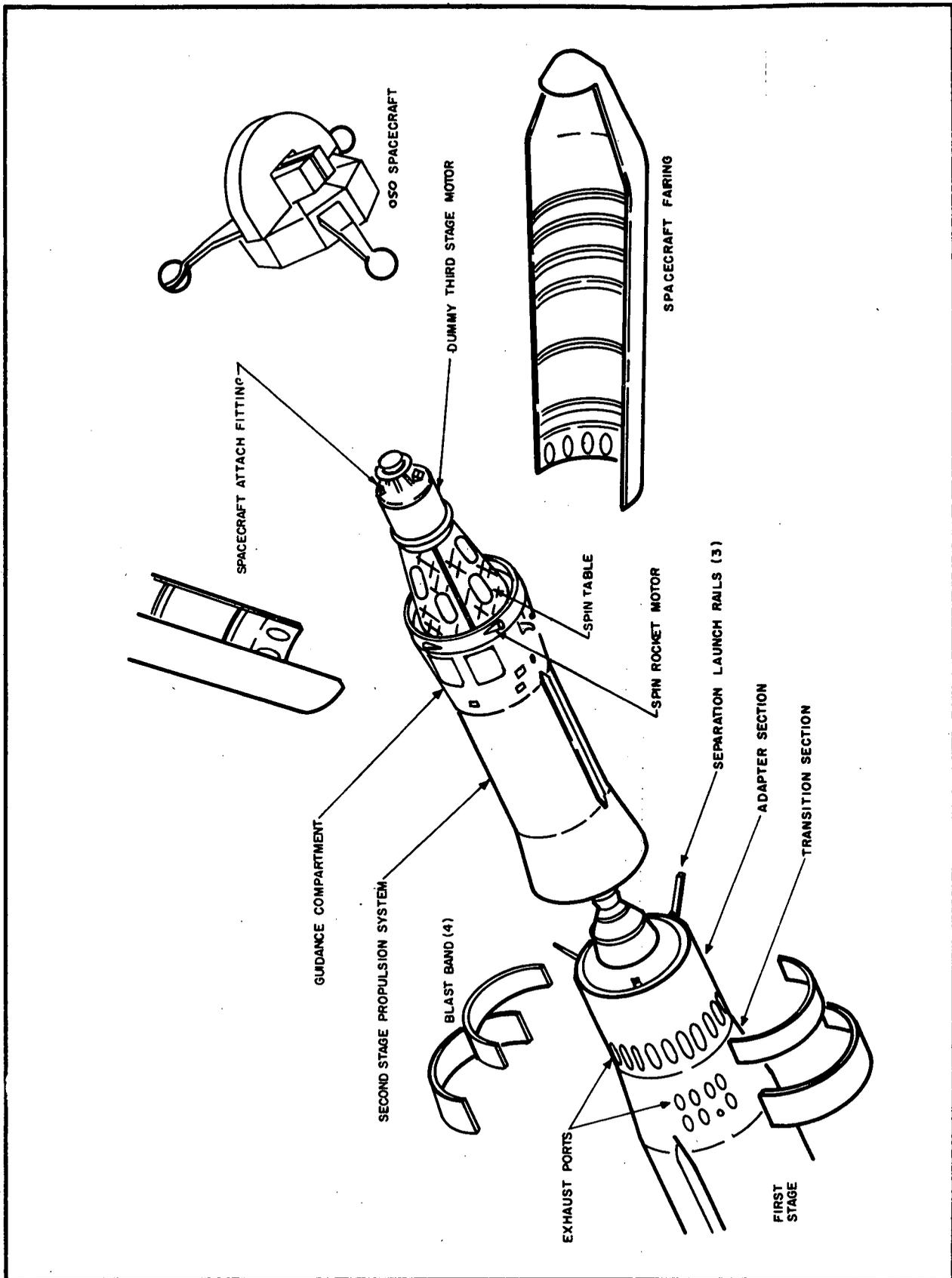
Fig. 1-11   Delta-N Separation Mechanisms

There were no major problems or unscheduled holds during the final F-0 Day countdown. The weather on F-0 Day was very good and caused no problems. There was no major pad damage.

Guidance, control, electrical, and propulsion systems on both vehicle stages were generally nominal. Telemetry performance and data acquisition were excellent. Radar and optics support were good except for the loss of one camera.

Launch Vehicle Operation. The Delta-72 vehicle was launched on a flight azimuth of 108 degrees true. Initial plots of the early portion of the flight showed the vehicle to be right and high of the nominal predicted. (The position plots were about 2 nm higher than nominal XZ until MECO.) The balance of the flight was nominal or near nominal. The angle between the OSO wheel plane and the sun was 2.7 degrees at orbit injection.

The Instantaneous Impact Chart plots were slightly right of nominal until WECO guidance brought them back to nominal at about T+150 seconds. These plots remained nominal for the rest of the flight.

MECO occurred as predicted and SECO appeared to have occurred as expected. The rest of the flight was on time. The major flight events and their nominal and actual times of occurrence are shown in Table 1-3.

H72-89

Fig. 1-12   OSO-6 Launch

Table 1-3

SIGNIFICANT LAUNCH EVENTS

Event	Times After Liftoff (sec)	
	Nominal	Actual
Liftoff	0352:00 EDT	0352:00.1 EDT
Main engine cutoff	220.8	217.6
Stage-II ignition and Stage-I/II separation	224.8	221.6
Fairing jettison	229.8	226.6
Second-stage engine cutoff	605.6	591.6
Terminal pitch maneuver started	631.88	Unknown
Pitch maneuver ended	718.83	Unknown
Spin Table spinup	738.8	735.6
OSO separation	740.8	737.6

OSO Spacecraft. Spacecraft functions occurred normally during powered flight. At T+735.6 seconds the spacecraft was spun up to 97.87 rpm. Realtime data from Antigua verified spacecraft separation, initiation of the spacecraft timer, and deployment of the spacecraft arms. Realtime data from Ascension Island indicated the spacecraft was despun, the sun was acquired, nutation damper was uncaged, pointed experiments elevation unlocked, and the auto pitch and spin limiters were enabled.

The elevation drive motor current was higher than expected when in the pitch up position due to slight loading by the flex cables. This anomaly did not degrade spacecraft operation.

The early OSO-6 orbital parameters from Tracking and Data System were:

	Nominal	Actual
Apogee (kilometers)	552.6	560.3 (302.4 nm)
Perigee (kilometers)	540.0	494.6 (266.9 nm)
Inclination (degrees)	32.9	32.964
Period (minutes)	95.68	95.1
Eccentricity	0.00019	0.004754

The orbital path over the earth's surface for the first few orbits is shown in Fig. 1-13.

1.6   SPACECRAFT DESCRIPTION

The role of the basic spacecraft was to accurately point the pointed instruments at the center of the sun, provide a stable platform for the "wheel experiments", provide inertial aspect and time correlations for the experiments, provide operational control by ground command, and telemeter the experiment data and spacecraft performance (housekeeping) data to earth ground stations. To do this, the spacecraft used ten primary systems:

    (1)   Structures and Thermal Control
    (2)   Pitch and Roll Control
    (3)   Spin Control
    (4)   Pointing Control
    (5)   Nutation Control
    (6)   Launch Sequence Control
    (7)   Command
    (8)   Telemetry
    (9)   Power
  (10)   Aspect

In addition to these ten spacecraft systems, there were seven primary scientific experiments; the composite of these systems and experiments (including the basic spacecraft structure) made up the complete observatory.

Fig. 1-13 OSO Subsatellite Plot

1.6.1   Spacecraft Structures and Thermal Control

The basic spacecraft consisted of two main structures: the rotating wheel structure, and the solar-oriented sail structure (Fig. 1-14).

<u>Wheel Structure</u>. The wheel consisted of a cylinder divided into nine wedge-shaped compartments arranged around a central hub. Four of these compartments housed the wheel electronics (Fig. 1-15) and the remaining five housed the wheel experiments. Attached to the perimeter of three compartments were three extendible arms that supported the spin-control gas bottles and reaction jets. During the launch phase, these arms were released from the stowed state and moved upward into the plane of the wheel, thus increasing the effective diameter of the wheel. This was done to increase the wheel spin-moment-of-inertia, and thereby increase its gyroscopic rigidity while also providing a longer lever arm for the spin-control jets to act upon. The three arms were latched in the stowed state by squib-actuated latching pins and support brackets; these brackets also served as part of the antenna array. A hydraulic arm damper system was used to prevent damage to the arms when they were released during the launch phase.

The wheel and sail were connected by an azimuth shaft (Fig. 1-16) which extended from the base of the azimuth shaft assembly through the wheel to the base of the azimuth casting which supported the sail. This shaft was held in position by two low-friction azimuth bearings; one at the top and one at the bottom. An azimuth drive motor was located near the uppermost bearing with its stator attached to the azimuth shaft housing and its rotor attached to the azimuth shaft. The azimuth drive motor, in conjunction with the pointing control system, pointed the sail (and, therefore, the pointed experiments) in azimuth during orbit-day by driving the sail at an opposite

SOLAR CELL ARRAY
MOUNTING SURFACE

SAIL STRUCTURE

AZIMUTH CASTING

ELEVATION CASTING

POINTED INSTRUMENTS

AZIMUTH SHAFT

CONTROL SYSTEM
EYE BLOCKS

WHEEL STRUCTURE

TOP THERMAL SHIELD

CENTRAL HUB

COMPARTMENT
SIDE WALL

COMPARTMENT DECK

BOTTOM THERMAL
SHIELD

ARM DAMPER

ARM

TYPICAL EXPERIMENT
INSTRUMENT

COMPARTMENT
RIM PANEL

ANTENNA

SUPPORT BRACKET

SPIN GAS BOTTLE

Fig. 1-14 Main Features of OSO Structure

MAGNETOMETER PROBE
SPIN CONTROL ASSEMBLY
MAGNETOMETER ELECTRONICS
SQUIB BATTERIES
PCM JUNCTION BOX ASSEMBLY
COMMAND DECODERS AND RECEIVERS
DIGITAL MULTIPLEXER & ENCODER
ANALOG SUBCOMMUTATOR
DIGITAL SUBMULTIPLEXER POWER SUPPLY
ANTENNA MATCHBOX
MAIN BATTERIES
DIGITAL SUBMULTIPLEXER
DIPLEXER
HYBRID CIRCULATOR
SPIN CONTROL & ASPECT SENSORS
ANTENNA
TRANSMITTER
LAUNCH SEQUENCE TIMERS
TAPE RECORDERS
COAXIAL SWITCH
ASPECT MEASURING ELECTRONICS
TRANSMITTER

H69-267

Fig. 1-15   Wheel Electronics Systems

BEARING ASSEMBLY
AZIMUTH DRIVE MOTOR
PITCH GAS CONTAINER
AZIMUTH SHAFT
"O" RINGS
BEARING ASSEMBLY
SLIP RINGS
ATTACH FITTING
SEPARATION SWITCH
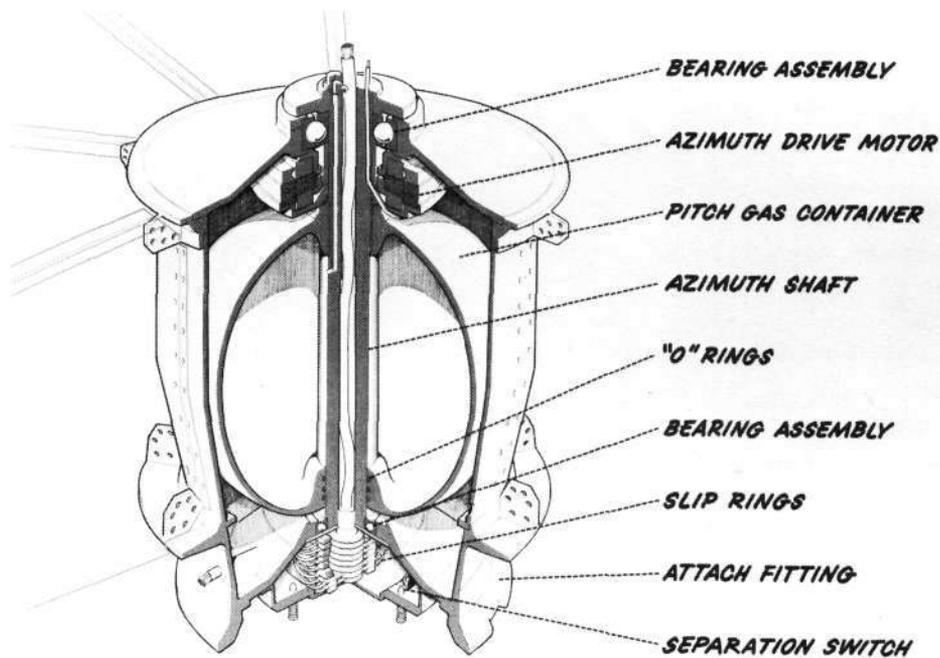
Fig. 1-16   Hub and Azimuth Shaft Assembly

but equal rate with respect to the wheel. Mounted at the base of the azimuth shaft, was a slip ring assembly that transferred power, telemetry signals, and control signals between the sail and the wheel and the separation switches that indicated when separation occurred. Mounted between the two azimuth bearings was a high-pressure nitrogen gas container that stored the gas supply for the pitch control system contained in the sail.

Sail Structure. The sail consisted of a semicircular framework attached atop the azimuth shaft. Solar cells covered the entire sun-facing surface except that part occupied by the pointed experiments. Behind the solar cell array were the electronic, mechanical, and pneumatic components necessary for the operation and orientation of the sail and pointed experiments (Figs. 1-17 and 1-18). The two pointed experiment packages were mounted side-by-side in the elevation casting, which in turn was mounted in the azimuth frame. The elevation casting was supported by two elevation stub shafts and two elevation bearings. An elevation drive motor, connected to one of the stub shafts, provided, in conjunction with the pointing control system, elevation pointing control of the pointed experiments. Both the left-hand and right-hand experiments carried very sensitive solar sensors (fine eyes) that provided "fine" azimuth and elevation pointing error and readout signals for the pointing control system. Also mounted on the sail were other azimuth solar sensors (coarse eyes) that provided "coarse" drive signals for initially despinning the sail each orbit-morning.

Pitch control of the observatory was provided by a nitrogen-gas pneumatic system (Fig. 1-19) contained within the sail and controlled by the pitch control system and a system of magnetic control. The pitch gas was stored in the pitch gas bottle contained in the azimuth shaft assembly and the precession jets were located on either side of the sail.
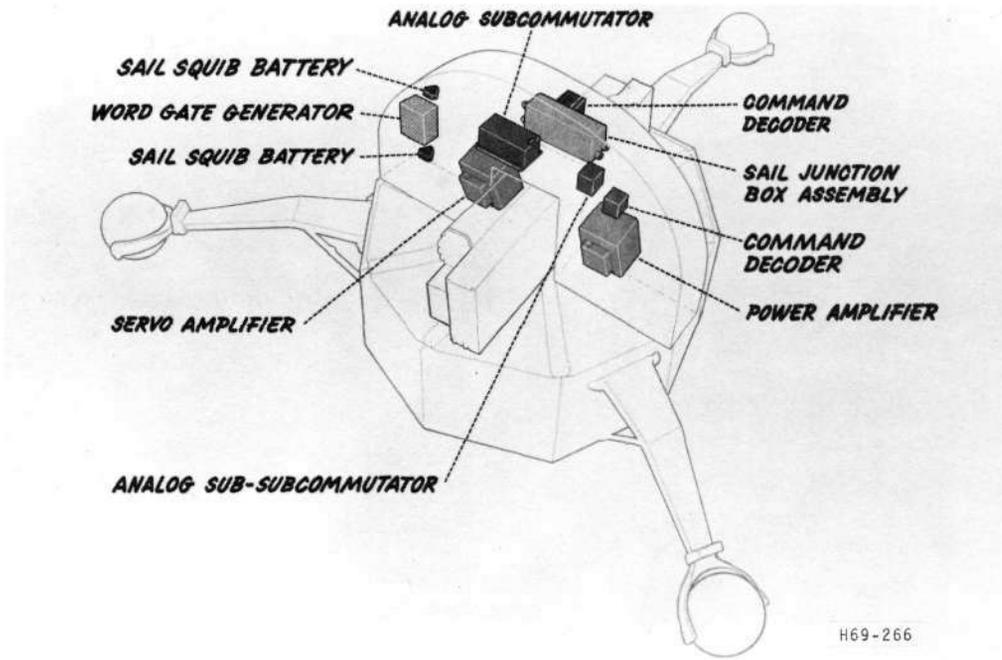
Fig. 1-17   Sail Electronics
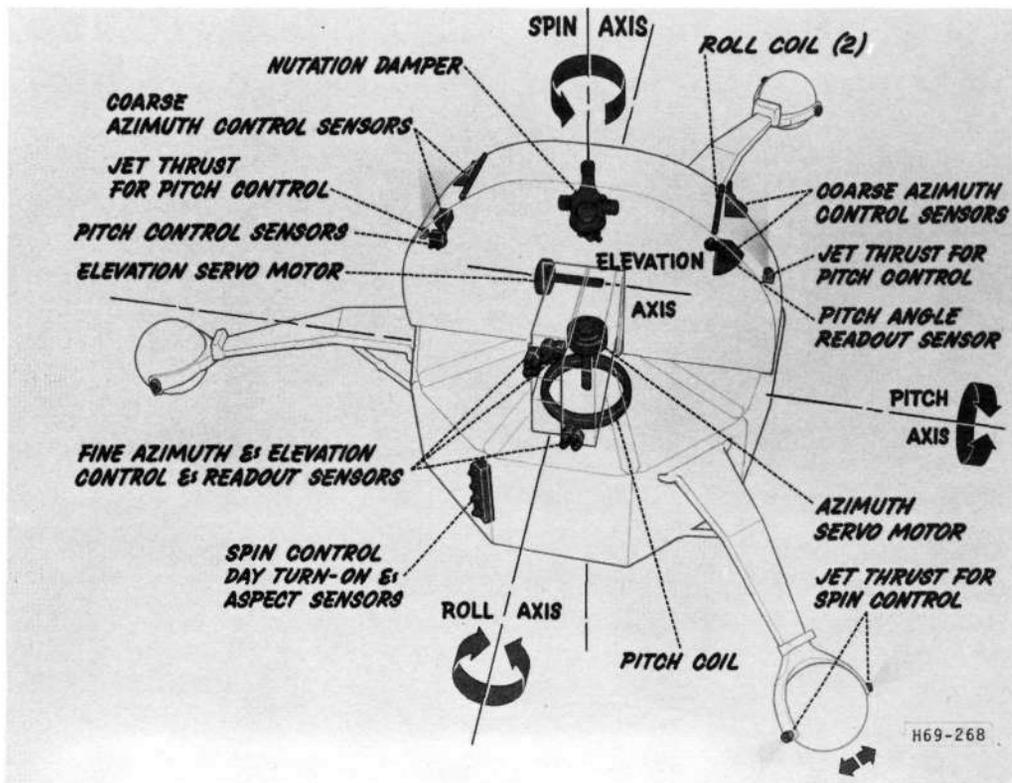
Fig. 1-18   Control Systems
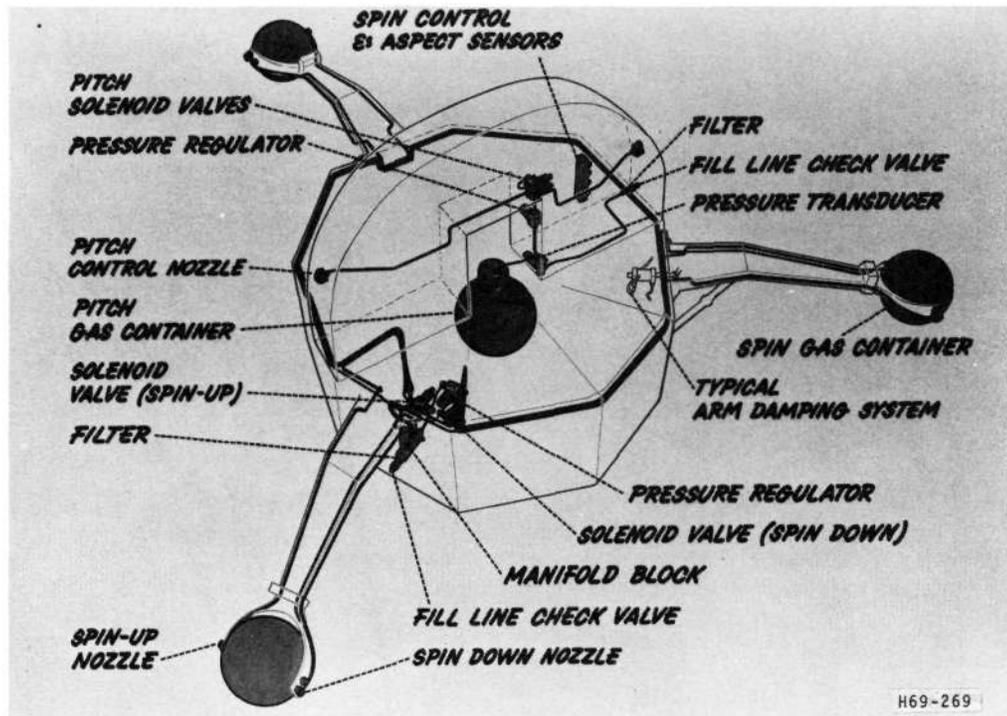
Fig. 1-19 Gas Control Systems

A nutation damper was mounted behind the solar cell array and
served to damp any nutational motion the spacecraft may have
acquired during its operation.

Spacecraft Dimensions and Mass Properties. The OSO-6 observatory
had an overall diameter of about 44 inches and an overall height
of about 38 inches including the attach fitting (Fig. 1-20). In
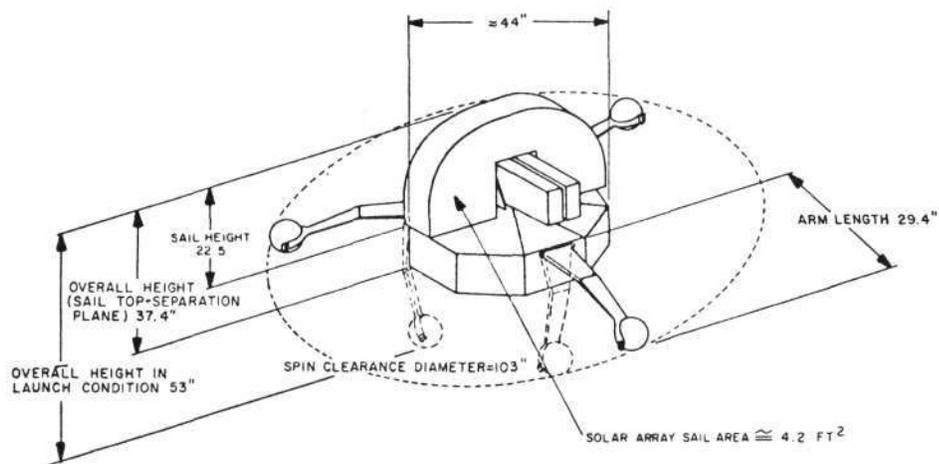
Fig. 1-20 OSO Overall Dimensions

the launch condition, the arms were folded, thereby creating a total payload length of about 54 inches. Table 1-4 lists some of the more important mass properties of OSO-6.

Table 1-4

OSO-6 IMPORTANT MASS PROPERTIES

Property	Value
Total observatory launch weight	639.91 lb.
(Total experiment weight)	(218.76 lb.) (34.3 percent of total)
Spin MOI (slug-ft^2)	
Wheel only	22.1 (launch); 29.9 (orbit)
Wheel and sail	27.5 (launch); 35.3 (orbit)
Transverse MOI (slug-ft^2)	
X-axis	23.6 (launch); 25.0 (orbit)
Y-axis	24.8 (launch); 23.9 (orbit)
Mean (X & Y)	24.5 (orbit)

Thermal Control. For OSO to accomplish its mission, it is necessary to keep its temperature within specified limits in those sections of the spacecraft that house temperature-sensitive experiment instruments. Therefore, a thermal analysis was performed for the OSO-6 mission with an assumed experiment and spacecraft equipment complement. This analysis considered the thermal characteristics and temperature requirements of each instrument and included such factors as: solar radiation, albedo, earth radiation, electrical heating, radiation to space, conduction and radiation between parts of the spacecraft, and the energy stored by these parts. This analysis also considered typical orbits from the hottest to the coldest during a year, and predicted the temperature changes during each orbit and the orbit mean temperature. From this analysis, we determined the

type of temperature control schemes needed to satisfy the temperature control requirements.

As a result, simple, passive temperature control schemes were used throughout. These schemes consisted of using special paints and surface finishes to control the heat gains and losses and by providing the proper heat conduction paths within the structure.

A more detailed discussion of the structure and thermal control systems is given in Section 11. Summary descriptions of structural components are given in Appendix C.

1.6.2   Pitch and Roll Control

Pitch Control. Because of external torques acting on the spacecraft and the motion of the earth about the sun, the orientation of the spin axis had to be continuously corrected in pitch to maintain the spin axis within ±3 degrees of normal to the solar vector during the life of the spacecraft. These corrections were made by two independent subsystems: (1) a pneumatic pitch control subsystem, and (2) a magnetic bias coil (pitch-roll coil). The latter was used as the primary control system and the gas system was used only when fast pitch maneuvers were required.

The pneumatic pitch control system consisted, basically, of pitch-angle solar detectors, electronic control circuits, and nitrogen gas thrusters. It has two modes of operation: (1) the automatic pitch mode and (2) the manual pitch mode.

In the automatic pitch mode, this system automatically maintained the pitch attitude by sensing the pitch angle and releasing high pressure N_2 gas through either the pitch-up or pitch-down control nozzles (thrusters) when the ±3 degree

limits were exceeded. These nozzles were positioned so their impulse caused the observatory to precess in the proper direction (up or down) to correct the pitch error.

In the event of an automatic mode failure, there were circuits that would automatically select the manual pitch mode. The automatic and manual modes could also be selected by ground station command. In the manual control mode, bursts of pitch gas could be released in response to ground commands to change pitch attitude as desired.

The magnetic bias coil consisted simply of a coil of wire placed around the central hub of the spacecraft (Fig. 1-18). When activated by ground commands, this coil would produce a magnetic field that interacted with the earth's magnetic field and caused the spacecraft to slowly pitch in the desired direction. This coil was used to conserve pitch gas and thus greatly increased the operational life of the spacecraft.

The pitch attitude angle was continuously monitored during the day portion of the orbit by a pitch readout solar sensor and telemetered to receiving ground stations.

Roll Control. Roll control was strictly by magnetic means and was provided by the pitch-roll coil (discussed above) and two roll-spin coils mounted in the sail and aligned parallel with the roll axis (Fig. 1-18). These two sets of coils were controlled by ground commands to maneuver the spin axis into the desired roll attitude. The roll attitude was determined by the aspect measuring system.

A more detailed discussion of the pitch and roll control systems is presented in Section 7 of this report. Summary descriptions of this system and its components are given in Appendix C.

1.6.3   Spin Control

It was necessary to keep the wheel spin rate between certain limits for the following reasons: (1) so that the gyroscopic stability of the observatory could be maintained, (2) so that the tuned nutation damper would be most effective (nutation frequency is dependent upon spin rate), and (3) to facilitate the acquisition of wheel experiment data. For these reasons, a pneumatic spin control system was used to automatically keep the wheel spin rate between 0.44 and 0.68 revolutions per second.

The spin control system consisted basically of sun-detecting spin control eyes mounted on the perimeter of the wheel, electronic control circuits contained in the spin control box, and a nitrogen gas pneumatic system. The spin control eyes detected the sun once per wheel revolution, and if the wheel spin rate was outside the range of 0.44 to 0.68 rps, a burst of high pressure N_2 gas would be released from the appropriate spin-up or the spin-down nozzles located at the ends of the three arms.

In the event of a system malfunction in which the spin rate would go too far outside the normal range, the spin control system contained circuits that would automatically switch the system to the manual control mode. In the manual mode, either despin or spin-up bursts could be released by ground command. The selection of either mode of operation could also be accomplished by ground command when desired.

Data from which the spin rate could be computed was continuously telemetered to receiving ground stations. A more detailed description of the spin control system is presented in Section 7. Summary descriptions of this system and its components are given in Appendix C.

1.6.4   Pointing Control

The purpose of the pointing control system was to provide automatic solar acquisition each orbit-morning, and to point the sail and the pointed experiments at desired positions on the sun during the orbit-day.

The pointing control system consisted of three pointing control servo loops: (1) the azimuth-coarse, (2) the azimuth-fine, and (3) the elevation-fine servo loops. These servo loops consisted basically of three separate control sensor assemblies (control eyes), electronic control circuits contained in the servo amplifier and power amplifier, and the azimuth and elevation servo motors (Fig. 1-17 and 1-18).

The OSO-6 pointing control system could be operated in any of four modes: (1) normal center-point mode, (2) large raster mode, (3) offset point mode, and (4) small raster mode (Fig. 1-21).

POINT MODE
(1) NORMAL ORIENTATION TO THE CENTER OF THE APPARENT SOLAR DISK WITHIN ±1 ARC MINUTE
(2) USED BY BOTH POINTED EXPERIMENTS ON SELECTED WAVELENGTHS AT CERTAIN TIMES

LARGE RASTER MODE
(1) ACTUATED BY GROUND COMMAND
(2) 64 LINES ; 0.72 ARC MINUTE APART
(3) LINE SWEEP RATE: 46 ARC MINUTES PER 7.68 SECONDS
(4) TOTAL RASTER TIME: 491.52 SECONDS
(5) USED BY BOTH POINTED EXPERIMENTS ON SELECTED WAVELENGTHS AT CERTAIN TIMES

SMALL RASTER MODE
(1) ACTUATED BY GROUND COMMAND
(2) 16 LINES; 7 ARC MINUTES WIDE; 7½ ARC MINUTES HIGH
(3) LINE SWEEP RATE: 7 ARC MINUTES PER 1.92 SECONDS
(4) TOTAL RASTER TIME: 30.72 SECONDS
(5) CAN BE CENTERED ON ANY OFFSET POINT POSITION
(6) USED BY BOTH POINTED EXPERIMENTS ON SELECTED WAVELENGTHS

OFFSET POINT MODE
(1) ACTUATED BY GROUND COMMAND
(2) ANY POINT CAN BE SELECTED WITHIN A 128 BY 128 POINT GRID (16384 POINTS) THAT COVERS A SQUARE 46 BY 46 ARC MINUTES
(3) USED BY BOTH POINTED EXPERIMENTS ON SELECTED WAVELENGTHS AT CERTAIN TIMES

Fig. 1-21 Pointing Modes

In the normal center-point mode, the optical axes of the pointed experiments were pointed at the center of the sun with an accuracy better than ±1 arc minute. In the large raster mode, the pointed experiments were caused to scan the sun with a 64-line (46 x 46 arc minute) raster pattern centered on the sun. In the offset point mode, the pointed experiments could be pointed at any one of 16,384 points (128 x 128 grid) within the large raster pattern. In the small raster mode, the pointed experiments were caused to raster about any of the offset points with a raster pattern 7 arc minutes in azimuth and 7.5 arc minutes in elevation.

The different modes of operation were selected by command and their use depended on the desires of the experimenters and the GSFC/OSO project personnel. The azimuth and elevation error signals and the point and raster azimuth and elevation readout sensor outputs were telemetered to receiving ground stations.

A more detailed description of the pointing control system is presented in Section 7. Summary descriptions of this system and its components are given in Appendix C.

1.6.5   Nutation Control

Nutation of the observatory from precessional impulses was damped by a "passive" nutation damper located near the top of the sail (Fig. 1-18). This damper consisted of a pendulum bob supported by a wire and emersed in a bath of silicone oil. The bob was tuned to the natural nutation frequency of the observatory when it rotated at 0.5 rps. When the observatory spin axis nutated, the motion of the bob in the oil dissipated the nutation energy in the form of heat.

A more detailed description of this system is presented in Section 7.

1.6.6   Launch Sequence Control

The launch sequence system controlled the unlatching of mechanical assemblies stowed or locked during launch, and supplied activation signals for electrical control functions needed during orbit injection. The system was fully redundant, containing two of each of the following:

- Separation switches

- Electromechanical launch sequence timers

- Squib-firing battery packs in the wheel

- Squib-firing battery packs in the sail

- Squibs in each latch block, and

- Relays and wiring as required.

The timers, which controlled the spacecraft launch sequence events, were started by the separation of the spacecraft from the spin table. The separation signal was provided by the separation switches which were located on the bottom of the OSO attach fitting (Fig. 1-16). The timers provided pulses to control relays which executed the spacecraft functions shown in Fig. 1-22.

This system is described in more detail in Section 7. Summary descriptions of this system and its components are presented in Appendix C.

Ball
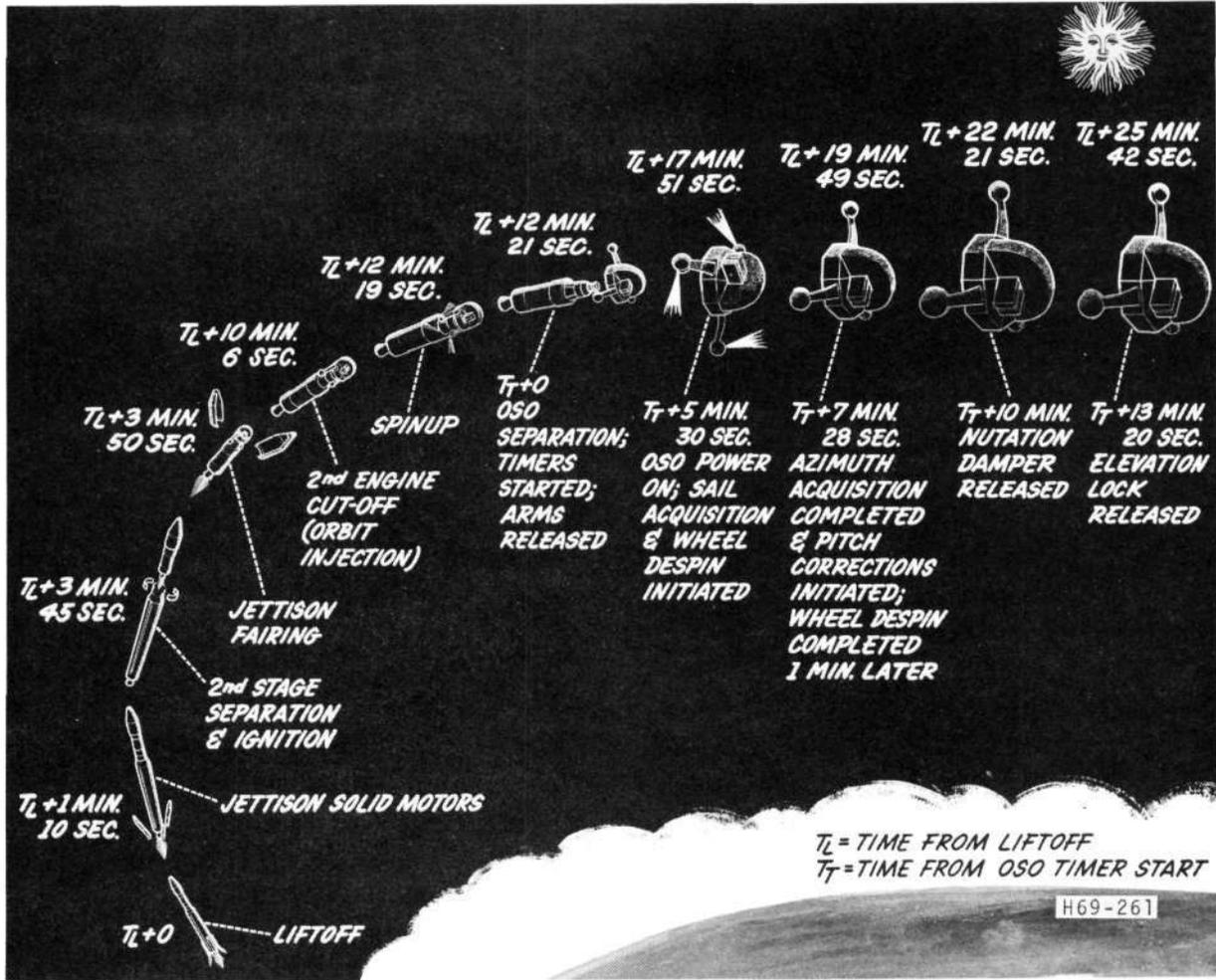
Fig. 1-22   OSO-6 Launch Sequence

1.6.7   Aspect Determination

A knowledge of the orientation ("look" direction) of an experiment
when it records a cosmic event assists in the interpretation of
the event.   This orientation is called "experiment aspect" and was
calculated using data from OSO's aspect determination system.

The system consisted of a solar sensor, a fluxgate magnetometer
and null detector, the pitch-angle monitor, and associated
electronic circuits.   The telemetered information after processing
yielded:

(1) The angle between the observatory spin axis and the solar direction

(2) The roll angle of the observatory spin axis (about the solar direction) relative to ecliptic north

(3) The angular position of the wheel in its plane of rotation at the time of a detected cosmic event

There were other aspect parameters that were calibrated prior to launch. Examples of these are the tilt of an experiment's look direction relative to the wheel plane, and the angular separation between an experiment's look direction and the azimuth reference axis.

In addition to the spacecraft aspect measuring equipment, some wheel instruments contained their own aspect circuitry. This circuitry usually measured the elapsed time (and hence the amount of wheel rotation) between the passing of an azimuth solar sensor past the sun and the occurrence of a cosmic event. Then, by knowing the roll and pitch angles, the origin of the event could be determined.

The aspect system is described in more detail in Section 5. Summary descriptions of this system and its components are presented in Appendix C.

1.6.8   Command System

The command system provided the means by which the spacecraft and experiments could be remotely controlled by an RF command link between the spacecraft and ground stations. The system

consisted basically of two redundant command receivers, two redundant command decoders in the wheel, two redundant sail decoders, and various control relays and control circuits that provided functional control of the spacecraft and experiments.

The command system used a digital-tone format whereby the address and instruction commands consisted of binary-coded tone bursts. The OSO-6 command system had a complement of 206 commands: 67 for spacecraft functions, 108 for experiment functions, and 31 spares. The command frequency was 149.52 Mhz.

The command receivers detected and amplified the coded RF command signals. Each receiver provided input to an individual wheel decoder, and ground station commands controlled selection of which receiver provided input to each of the two sail decoders. The command decoders accepted the receiver output and, if properly addressed, decoded and executed the instruction commands.

A more detailed description of the command system is presented in Section 10 of this report. Summary descriptions of this system and its components are given in Appendix C.

1.6.9   Telemetry System

The spacecraft and experiment data was sampled, encoded, and telemetered to receiving ground stations by the data handling and RF subsystems. The data consisted of experiment and space- craft data that originated from spacecraft and experiment transducers, detectors, and monitor circuits. This data was presented to the data handling subsystem in either analog or digital form. Spacecraft and experiment performance data was in analog form; experiment prime data was in digital form. The capacity of the system was 184 8-bit data channels, seven

sync words, and 5 spares. The bit rate was 800 bps and the transmitting frequency was 136.71 Mhz.

The telemetry system processed the spacecraft and experiment data by performing five basic operations. These operations were:

    (1)   Commutation (sampling) and analog-to-digital conversion (encoding) of the spacecraft and experiment performance data

    (2)   Multiplexing (time sharing) of the digitized performance data, and experiment prime digital data

    (3)   Conversion of the multiplexed digital data into biphase format (Manchester coding)

    (4)   Simultaneous data storage and real-time transmitting of data to receiving ground stations

    (5)   Transmitting stored data to receiving ground stations in fast playback mode upon command

A more detailed description of the telemetry system is presented in Section 9 of this report. Summary descriptions of this system and its components are given in Appendix C.

1.6.10   Power System

The spacecraft power system converted solar energy into electrical energy, stored this energy in a rechargeable battery, and supplied it to the spacecraft and experiments, upon demand, throughout each orbit. The power system consisted of a solar

cell array, a storage battery assembly, and a power distribution system.

The silicon solar cell array converted the solar energy into electrical energy during orbit-day when the plane of the array was maintained nearly normal to the solar direction. Part of the array output was used to recharge the storage batteries and the balance was consumed by the spacecraft and experiment electrical systems during orbit day. The nickel-cadmium storage battery assembly stored electrical energy for launch and orbit-night consumption, and provided a power reservoir to meet peak power demands. Electrical power was distributed throughout the spacecraft by control relays and associated wiring.

The solar cell array could produce about 40 watts of power during orbit day, which exceeded the power demand by a margin of about ten percent.

A more detailed description of the power system is presented in Section 8 of this report. Summary descriptions of this system and its components are given in Appendix C.

1.6.11   Major Improvements to OSO-6 Systems

The more significant system improvements are briefly described in the following paragraphs:

Pointing Control. Previous OSO's had the capability to point at the center of the sun and to scan the sun with a 40 x 40 arc minute pattern. In addition to the normal sun-centered pointing mode, the OSO-6 was capable of:

(1) Generating a 46 x 46 arc minute raster centered on and covering the entire sun and some of its atmosphere.

(2) Providing offset pointing to any position in a 128 x 128 (16,384) point grid pattern centered on the sun. The overall size of this pattern was 46 x 46 arc minutes.

(3) Generating a small raster of 7 x 7.5 arc minutes that covered about 1/40 of the large raster area. This raster could be centered on any one of the offset points mentioned above.

These improvements meant that the linear region of the control eyes and readout eyes had to be extended and the gain of the pointing control servos had to be changed.

Sail Junction Box. The OSO-6 J-box used plug-in terminal boards and double-density connectors for interfacing with the sail harness and the flex cables. This arrangement made it easy to remove the boards for rework or replacement. In addition, smaller and lighter relays were used on OSO-6.

Flex Cables. The OSO-5 flex cables consisted of printed copper paths thermally bonded between layers of mylar with 37 pin connectors soldered and potted on the ends. The low melting point of mylar caused these cables to be susceptible to heat damage during soldering operations. The flat pointed wire was susceptible to damage by handling during routine spacecraft operations.

The OSO-6 required additional circuits across the elevation axis. In order to reliably provide the additional circuits and maintain flexibility across the elevation axis, OSO-6 flex

cables consisted of shielded and unshielded stranded wires embedded in a flat ribbon of silicone rubber material with 0.070 in. centers and "twist pin" connectors. The resulting cable was more flexible and durable than mylar cables and did not have the melting problem during solder operations.

Roll Control. Because of the requirement of the Rutgers experiment to keep the wheel plane parallel with the ecliptic plane for a minimum of 27 days, a roll coil system was incorporated into OSO-6 to control the roll attitude of the spacecraft.

The system consisted of two electromagnets and associated command and power circuitry. These magnets were located in the sail and aligned parallel to the roll axis. Each magnet was capable of generating a 2500 gauss cm^3 ($\pm$10%) magnetic dipole moment in the $\pm$x direction with an applied voltage of 19 volts. Therefore, both coils could generate a total magnetic dipole moment of $\pm$5000 gauss cm^3. This magnetic moment was capable of rolling the spacecraft by about 1.5 degrees/day.

Transmitter Spectrum. Previous measurements of the PSK 600A transmitter spectrum indicated that the spectrum was not in conformance with NASA Aerospace Data Systems Standards X560-63-2. In order to minimize this nonconformance, the technique of pre-modulation filtering was used. This technique limited the bandwidth of the information signal and consequently the bandwidth of the transmitted signal.

The pre-modulation filter was an active filter with a cut-off frequency of approximately 35 KHz. Functionally, the filter module consisted of a shaping circuit followed by a five pole active filter. The supply voltages for these circuits were derived from zener-diode supplies. An EMI filter was added to minimize the amount of conducted 136 MHz on the spacecraft power lines.

Receiver Sensitivity. In an effort to decrease the sensitivity of the OSO-6 command system to any given RF environment, the sensitivity of the receivers was reduced. The measured command system threshold using these modified receivers was about -103 dbm. This was about a 10 db increase in system threshold when compared to the -110 to -130 dbm measured on previous command systems.

Word Gate Generator. The word gate generator (WGG) was developed and installed in OSO-6 sail in order to minimize the number of slip ring circuits required. The WGG generated selected main frame and digital sub-frame word gates. That is, any 16 of the 32 main frame gates and any 12 of the possible 48 DSM gates could be duplicated in the WGG. The WGG characteristics were essentially identical to those of the wheel gate sources.

HCO/OSO-D Experiment Protection. In order to fly the flight spare HCO/OSO-4 instrument on OSO-6, an adapter was needed to redistribute the HCO interface wiring. This adapter was called the "K-Box". In addition to matching the OSO-4 and OSO-6 interfaces, the K-Box provided filtering originally contained in the HCO Filter Choke Module and contained the relays and wiring required to operate a overload protection circuit.

This circuit monitored the HCO high voltage power supply current. If a high voltage malfunction was indicated by a change in current, the OP circuit automatically actuated the HCO off command to disable the high voltage power supply.

Tape Recorder Operation. On OSO-5 both recorders were turned on by the same command. Separate commands were provided to turn the OSO-5 tape recorders off. On OSO-6, the recorders had all separate on and off commands.

Command Decoders. Previous OSO's used three decoders (two in the wheel and one in the sail). OSO-6 used four decoders (two each in the wheel and sail) to increase the command capability and to increase the reliability by providing redundant commands for critical functions.

Az. and El. PWM-Driver Oscillators. The OSO-6 azimuth and elevation drive oscillators (part of the pointing control servo) were separated in frequency by (nominally) 500 Hz to eliminate low-frequency "beating" and the resulting bus noise.

Section 2
OSO-6 FLIGHT PERFORMANCE

2.1 INTRODUCTION

During the first two years of orbital operation, OSO-6, making over 10,000 orbits around the earth, met every spacecraft mission requirement.

An impressive amount of high-quality data has been gathered from OSO-6's scientific payload. Analysis of this data continues at the facilities of the experimenters. While several of the experiments suffered instrument malfunctions, all experiments fulfilled a major part of their mission objectives.

The performance of the OSO-6 spacecraft is summarized in some detail in this section. Performance of experiments is not covered to any significant extent since analysis of such performance is the prerogative of the experimenter using his proprietary information.

2.2 PERFORMANCE IN THE LAUNCH PHASE

The OSO-6 was placed into a nearly circular orbit. Initial apogee and perigee were about 560 and 495 kilometers, respectively. The inclination of the orbit to the terrestrial equator is about 32.96 degrees. Performance of the launch vehicle was excellent, indicated by the closeness of the predicted and actual orbit parameters. The launch vehicle is described and its performance summarized in Para. 1.5 of this report.

Performance of OSO during the launch and early-orbit phases was likewise excellent in every respect. The launch-sequence timers performed their functions as programmed; solar acquisition, wheel despin, and transition to the orbit configuration occurred without difficulty.

Milestone vehicle and payload events are summarized in Fig. 1-22 and Table 1-3 of this report. Spacecraft spinup and separation were properly sequenced by the launch vehicle control circuits. Spin rate of the OSO just prior to spacecraft separation was about 98 rpm. At separation and spacecraft arm erection, spacecraft spin rate dropped to about 73 rpm due to the change in moment-of-inertia caused by arm deployment. Automatic despin to 0.618 rps took about 63 seconds, exhausting about 300 psi of spin-gas pressure.

The transition from the launch state to the orbit state was properly sequenced by the launch-sequence timers. No pitch correction was required, since the pitch angle at orbit injection was +2.7°. Azimuth and elevation solar acquisition occurred normally, and pointing accuracy was within specified tolerance. Battery charging started at solar acquisition. The nutation damper dissipated nutations caused by vehicle separation and initial elevation acquisition.

2.3 POINTING CONTROL SYSTEM IN ORBIT

The observatory pointed instruments were directed to within 0.3 arc minutes of the center of the sun (when in the sun-centered point mode) during the entire life of the observatory.

Figure 2-1 was derived from a computer plot of every readout of sail analog channels 1 and 3 obtained during a typical tape-recorder playback. These channels report elevation and azimuth pointing position. This type of plot is called an orbit profile plot. The computer software for deriving such a plot is discussed in Section 4. As part of the post-launch operational analysis effort, such plots were made and studied periodically. They were generally made within a few days after the pass had occurred, from data supplied by the NASA/GSFC OSO Control Center.
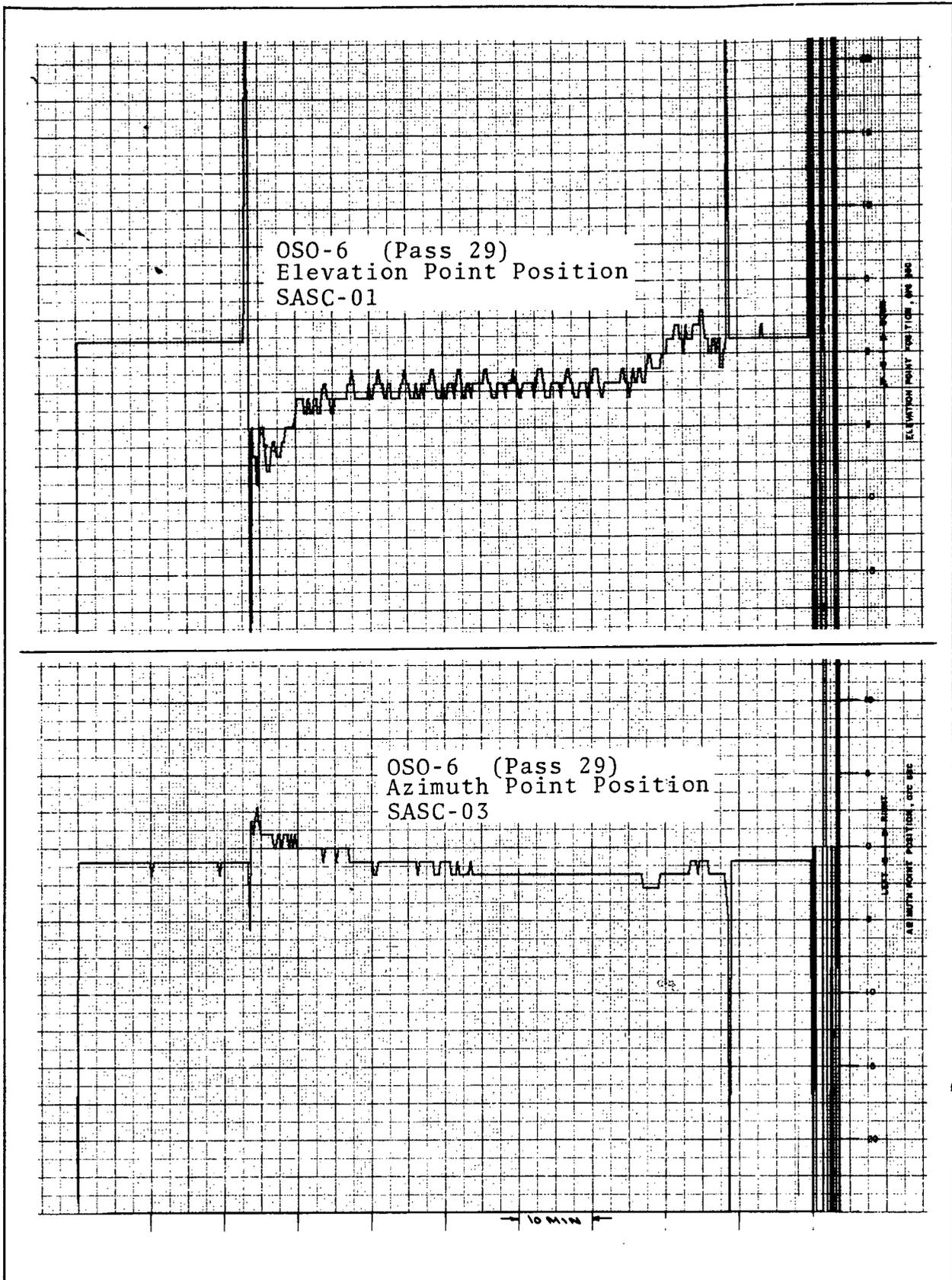
Fig. 2-1   OSO-6 Typical Orbit Profile--Sun-Centered Point

The discrete magnitudes of pointing error shown in the plots are a result of the analog-to-digital conversion resolution of the telemetry equipment. The interval between levels is equal to one telemetry bit. To reduce the effect of invalid samples on the curve, the plot program is designed to average transient samples with previous samples. This smooths the trace somewhat, producing the steps seen in the high-transient areas of the plots.

As can be seen, the "jitter" present in the pointing error was about ±2 (peak) seconds of arc. The rms value is evidently much smaller, but it cannot be evaluated from the digital data.

Offset Modes. The servo operated well within the design specifications in the offset pointing modes throughout the two-year design life of the observatory. Large raster, small raster, and offset pointing modes were selected by ground command as scheduled by the NASA-GSFC OSO control personnel. Azimuth and elevation position of the pointed instruments in each of the offset modes are shown in Figs. 2-2 through 2-4. These are typical orbit-profile plots, showing the indicated analog channel machine-plotted as a function of time for one full playback of the OSO on-board tape recorders.

Fig. 2-2 shows single-channel plots of elevation and azimuth position in the large raster mode from the playback in Pass 2585. The shift in position which appears from one orbit day to the next is a result of the loss of raster synchronization with the analog subframe through the orbit night. The sampling point in the raster is randomly established at the beginning of each orbit day. The azimuth scan plot, because of synchronization of the raster with the telemetry frame, is a plot of a single point in the scan, and, therefore, appears as a constant position.

OSO-6 (Pass 2585)
Elevation Raster Position
SASC-02

OSO-6 (Pass 2585)
Azimuth Raster Position
SASC-07
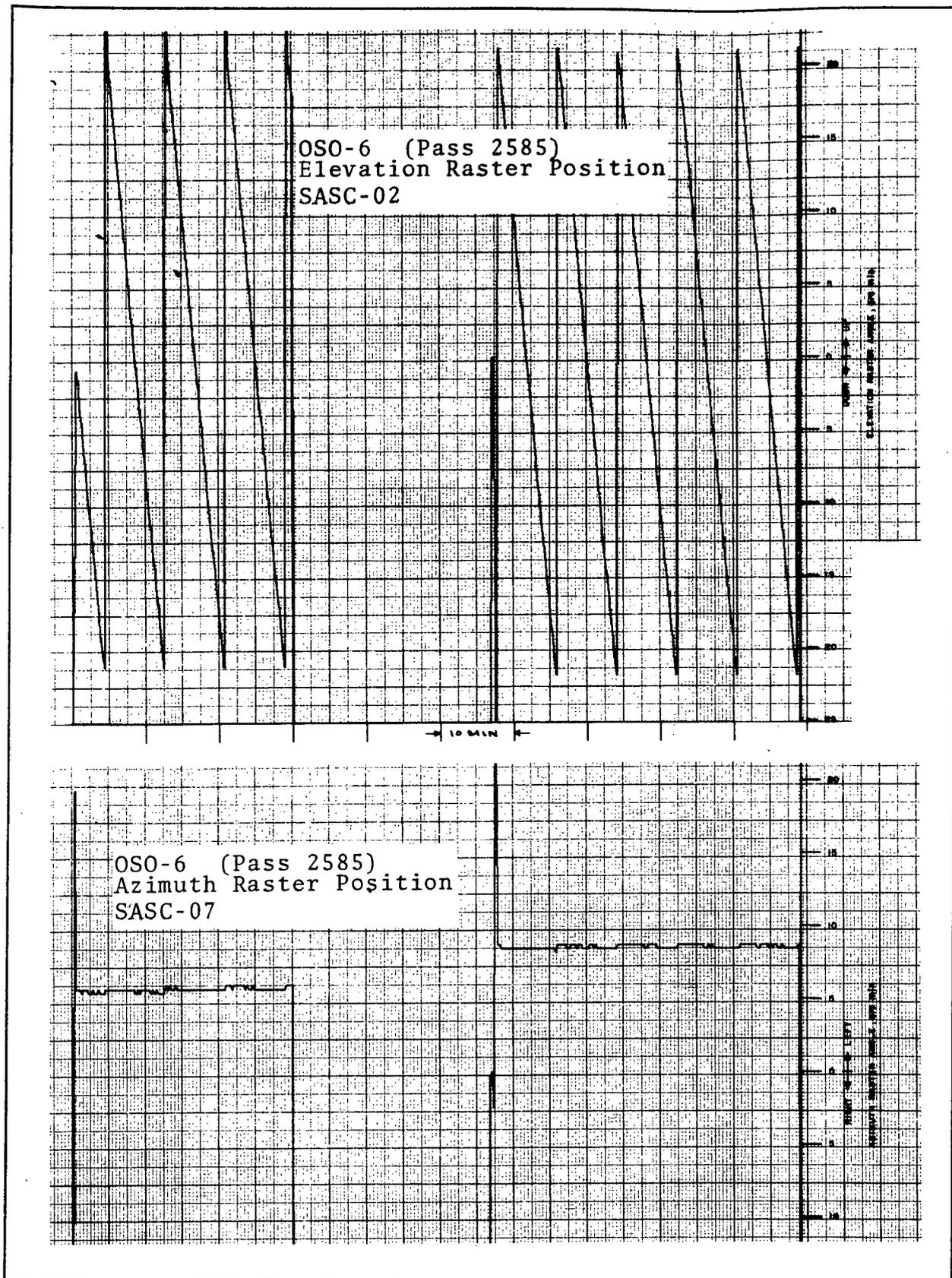
Fig. 2-2 OSO-6 Typical Orbit Profile--Large Raster
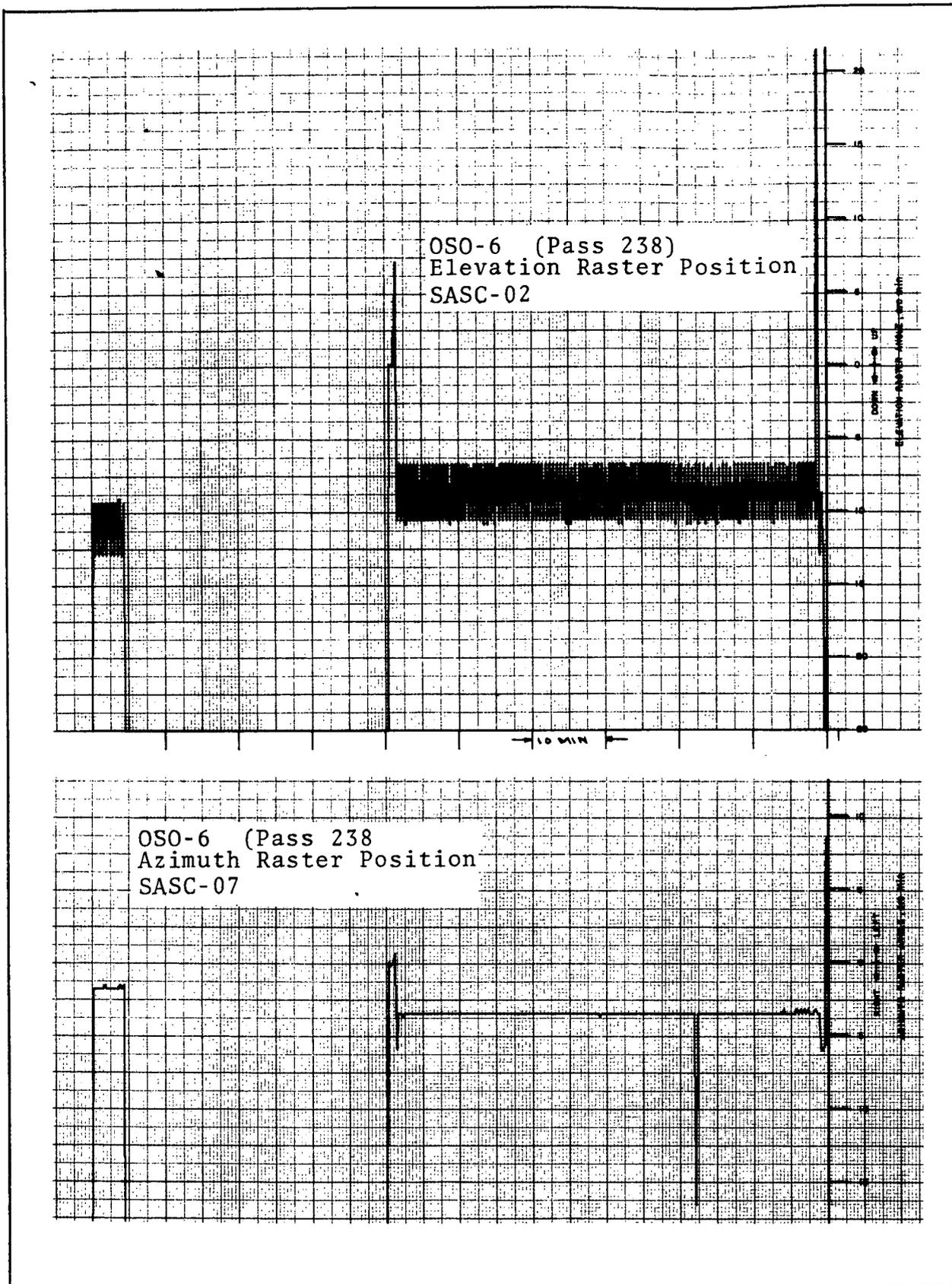
Fig. 2-3 OSO-6 Typical Orbit Profile--Small Raster
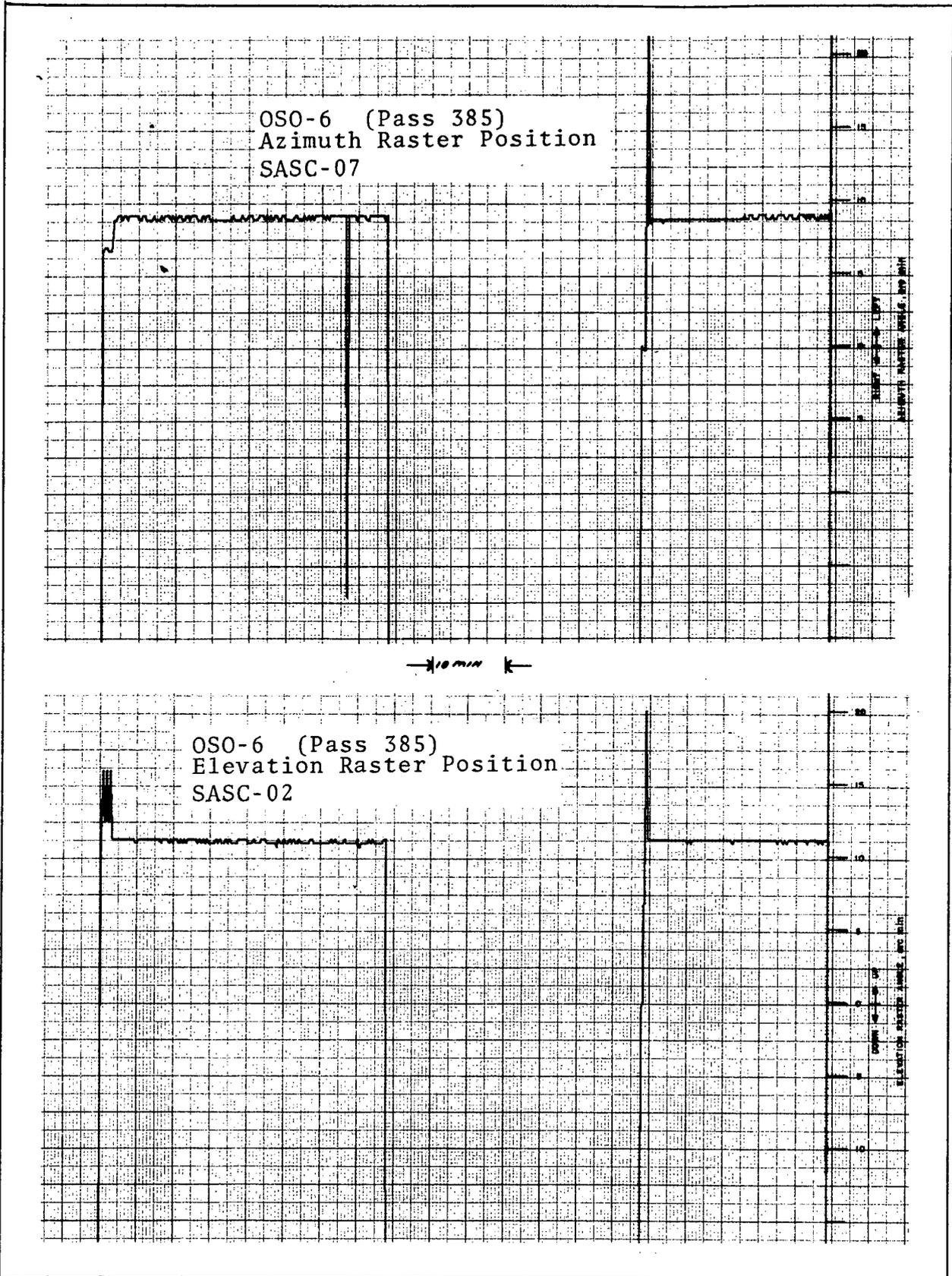
Fig. 2-4   OSO-6 Typical Orbit Profile--Offset Point

Figure 2-3 shows single-channel plots of elevation and azimuth position in the small raster mode from the playback in Pass 238. The shift in position noted through orbit night is the combined effect of telemetry frame resynchronization, and relocation of the small raster in the offset grid in behalf of the Harvard experimenter.

Figure 2-4 shows the elevation and azimuth position in the offset-point mode. The large transients in these plots are due to loss of pointing position at orbit dusk, the transient character of position during solar acquisition at orbit dawn, and telemetry anomalies associated with the splice in the continuous-loop tape of the on-board tape recorders.

Long Term Plots. Performance of the azimuth and elevation servos in the fine point mode is shown in Fig. 2-5. This is a manual plot of the position monitors and motor torque monitors covering one orbit-noon sample per day (when operating in the sun-centered point mode) for the first 100 days. These plots were continued over the two-year period and are summarized as follows:

(1) Elevation Fine Position (SASC-01)

This monitor indicated a gradual downward drift to about -10 arc seconds, with the greatest drift taking place in the first year.

(2) Elevation Raster Position (SASC-02)

This monitor seemed to show a slight upward drift of the instruments over the two-year period, but the drift was too slight to be conclusively identified by the analog monitor.
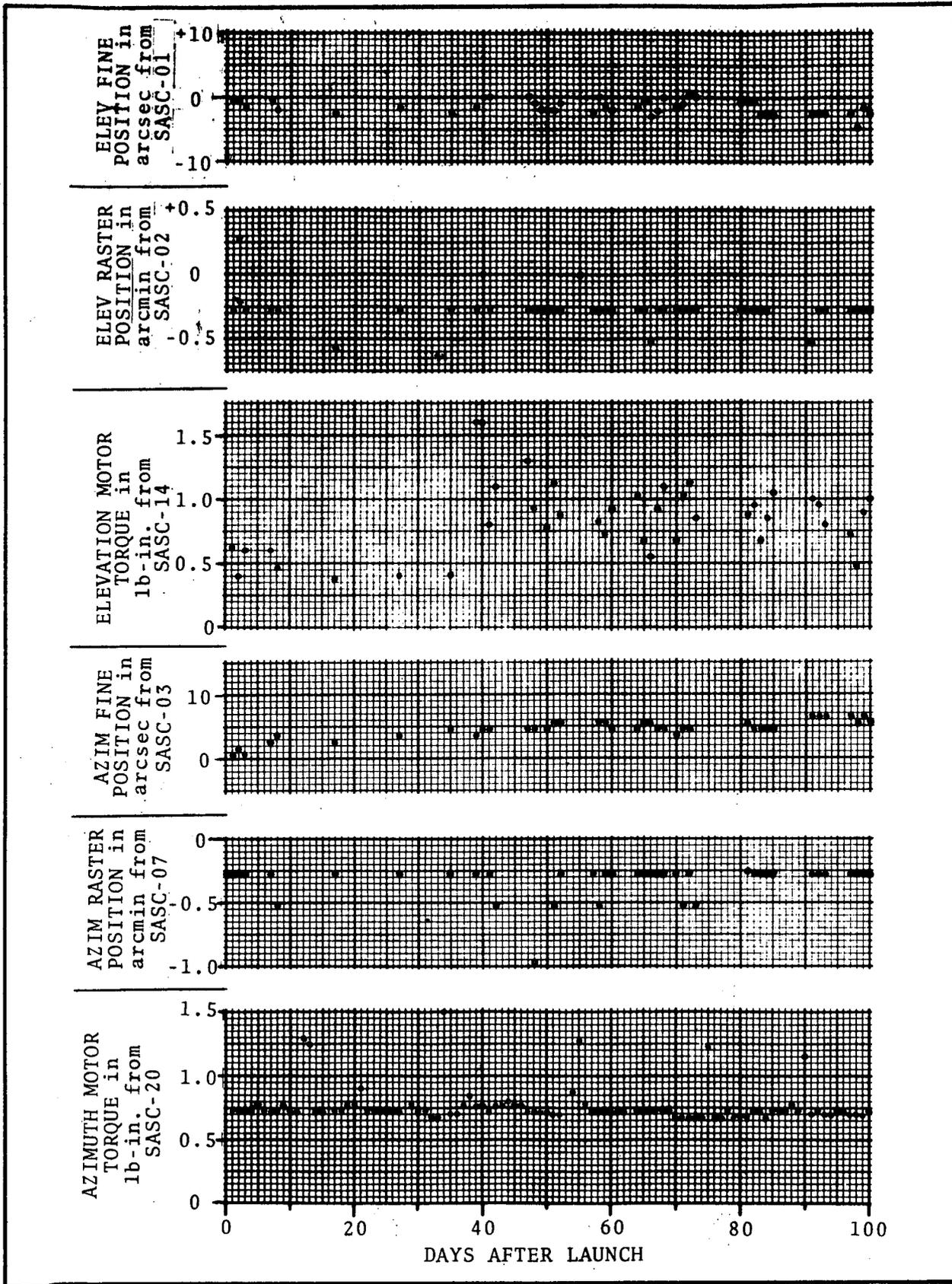
Fig. 2-5   Point Mode Performance -- First 100 Days

(3) Elevation Motor Torque (SASC-14)

No drift trend could be identified from the
monitor. Apparently, elevation bearing performance
did not degrade during the two-year life of
the observatory. Samples of the readout seemed
randomly distributed between 0.5 and 1.5 lb.-in.,
with an average torque about 0.8 lb.-in. Note
the jump in torque at about Orbit 38 on Fig. 2-5.
Comparison with Fig. 2-6 reveals that this jump
is related to OSO pitch angle. Indeed, the
torque produced by the flex cables from the
sail to the PIA caused elevation motor current
to increase with increasing pitch angle. While
the increase in motor current has to be considered
an undesirable effect, there was another effect
which compensated for it to some extent. The
torque applied by the flex prints held the PIA
against one side of the servo dead zone, thereby
decreasing elevation jitter. This is analogous
to the azimuth bearing friction's favorable
effect on jitter. Prior to these observations,
we had considered installing a spring in the
elevation gimbal for this purpose. Observing
OSO-6 in flight made the desirability of such a
spring obvious, and it was therefore incorporated
in subsequent OSO designs. Because of this
phenomenon, however, the OSO-6 pitch angle had to
be controlled within smaller boundaries than
planned, to limit the effect of flex-print torque
on elevation motor current.

(4) Azimuth Fine Position (SASC-03)

After a leftward drift of about 3 arc seconds
early in the flight, position settled at about

7 arc seconds left, and remained fairly constant thereafter.

(5) Azimuth Raster Position (SASC-07)

This monitor seemed to indicate a slight drift to the right over the two-year period. Owing to the limited resolution of the analog-encoded data, and the extremely small drift, the actual drift rate could not be evaluated.

(6) Azimuth Motor Torque (SASC-20)

The motor torque monitor gave no evidence at all of any degradation in performance of the azimuth bearings. The average value over the two-year period stayed at about 0.75 lb.-in.

2.4   ATTITUDE AND SPIN CONTROLS AND MONITORS

The attitude and spin controls and monitors were operated successfully as needed throughout the mission. Fig. 2-6 shows a plot of attitude, spin rate, and roll and pitch coil status during the first 560 days of the flight.

Spin Pneumatic. The pneumatic spin control exhausted about 300 psi of its nitrogen gas supply in the initial despin to 0.618 rps during the launch sequence. In Orbit 10, manual spin bursts were used to lower the spin rate to 0.521 rps. Thereafter, manual spin up or down bursts were used as needed to maintain the desired spin rates, including several changes made in order to observe the effect on the scientific experiment payload.

The use of the pneumatic spin control for spin rate was rarely needed except during times when the roll control coils were

FOLDOUT FRAME 2

FOLDOUT FRAME 1
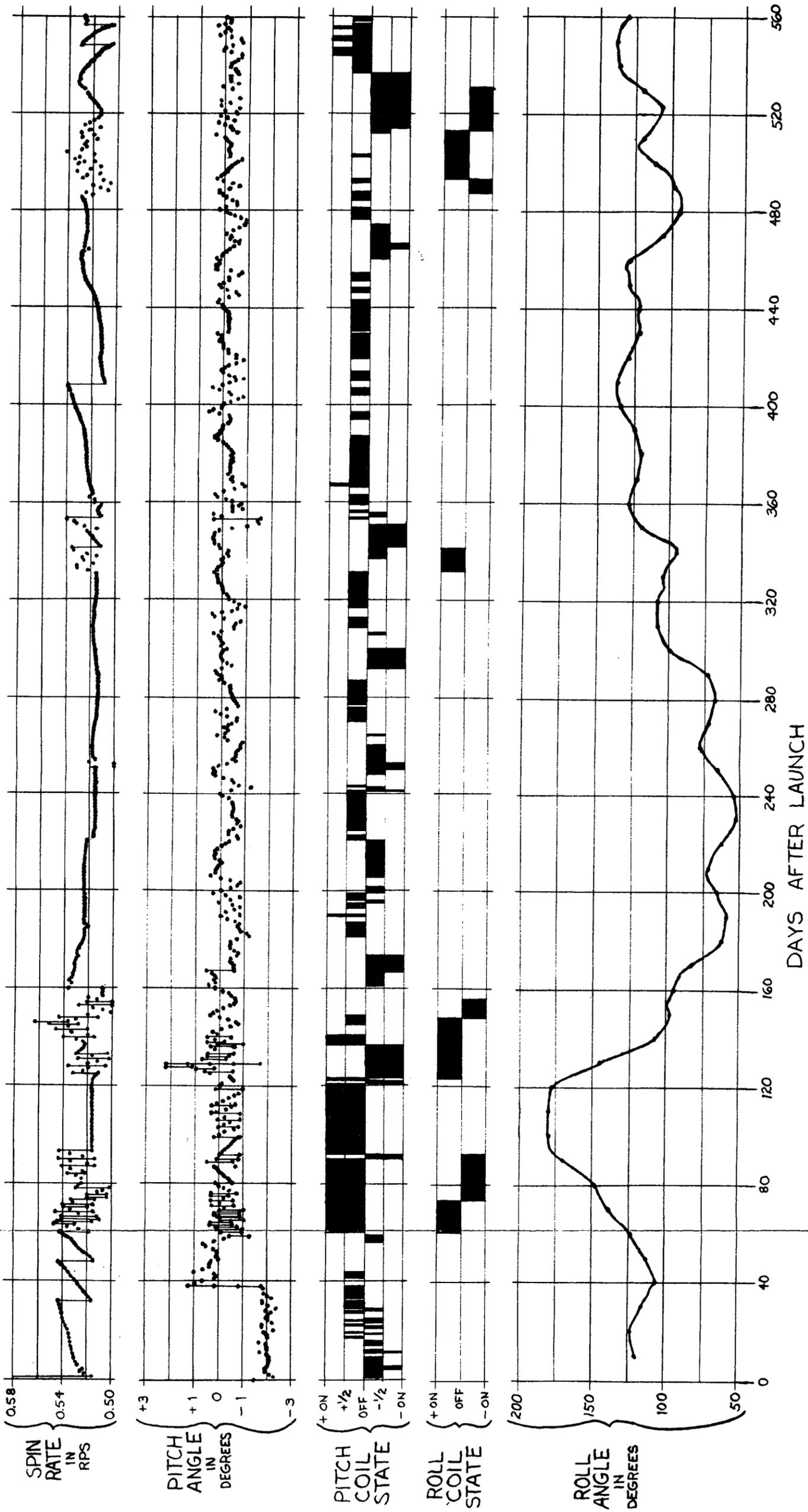
Fig. 2-6   OSO-6 Attitude History

2-12

2-12-1

energized. A glance at Fig. 2-6 shows a very strong roll coil/ spin rate interaction. At the end of the 560-day period shown in Fig. 2-6, the pneumatic spin control system was still charged to about 1600 psi. Storage pressure at launch was about 2970 psi. During the flight each spin up or down burst exhausted about 20 psi from the system and incremented the spin rate by about 0.025 rps.

Pitch Pneumatic. The pitch attitude of OSO-6 was maintained well within the specified +4-degree limit throughout the life of the observatory. No pitch correction was required in the launch phase since the observatory was placed into orbit with a pitch attitude of +2.7 degrees. It was immediately noticed that the elevation torque motor current was high, however, and after analysis of preflight test data it was concluded that the flex prints were applying a torque to the pointed instruments. Reduction of the pitch angle to -2° during the first two orbits, using the manual pitch pneumatic control, cut torque motor current to its proper level.

Pitch angle history for the first 560 days of the mission is presented in Fig. 2-6. At the end of the 560-day period, pitch gas pressure had diminished from its prelaunch value of about 2950 psi to about 2080 psi. Use of the pitch magnetic bias coil greatly reduced pitch gas consumption.

Magnetic Bias Coils. The magnetic pitch control coil and roll control coils were used successfully throughout the orbit life of OSO-6. Proper switching of the pitch coil power greatly reduced the consumption of the pneumatic system's gas supply, as in earlier OSO flights. Historical data, covering the first 560 days of the flight, are presented in Fig. 2-6.

Use of the roll coils for roll attitude control was complicated somewhat by their effect on the spacecraft spin rate. The roll coils were used early in the flight to establish a 180-degree roll orientation. This orientation was then maintained for about a month, then roll angle was allowed to decrease, giving the rotating instruments an opportunity to scan the celestial sphere.

There were no problems associated with command control of the magnetic attitude control coils except for several instances of coil status being changed by spurious commands. These changes were corrected by subsequent commands, executed in a timely fashion.

Aspect Monitors. As shown by the attitude history in Fig. 2-6, the OSO-6 attitude monitor system operated successfully throughout the mission. Pitch orientation was read out from the pitch-angle sensor through the sail analog subcommutator. The SORE (Spin Orientation and Rate Electronics) system provided data from which spin rate, roll orientation, and instantaneous look direction of the rotating instruments were calculated. Operation of the various attitude controls were based on the information provided by the SORE system.

Wheel spin rate was also successfully calculated from data supplied by the spin rate monitor through the wheel analog subcommutator.

2.5   TELEMETRY SYSTEM IN ORBIT

The OSO-6 telemetry system operated properly throughout the orbit life of the observatory. The OSO-6 government-furnished tape recorders have continued to supply excellent playback data well beyond two years. Real time and playback data were of excellent quality and there was no loss of data caused by malfunction of either the DME or tape recorders.

Receiving ground stations reported no difficulty in receiving or locking-in the OSO-6 transmitted signal.

Proper sampling of all data points, and their timing in the telemetry frame and subframes, were maintained throughout the mission. Data were time-correlated with spacecraft aspect and instrument look direction by using information supplied by the SORE and the telemetry frame counter in the digital subframe. A typical orbit profile of frame count is shown in Fig. 2-7.
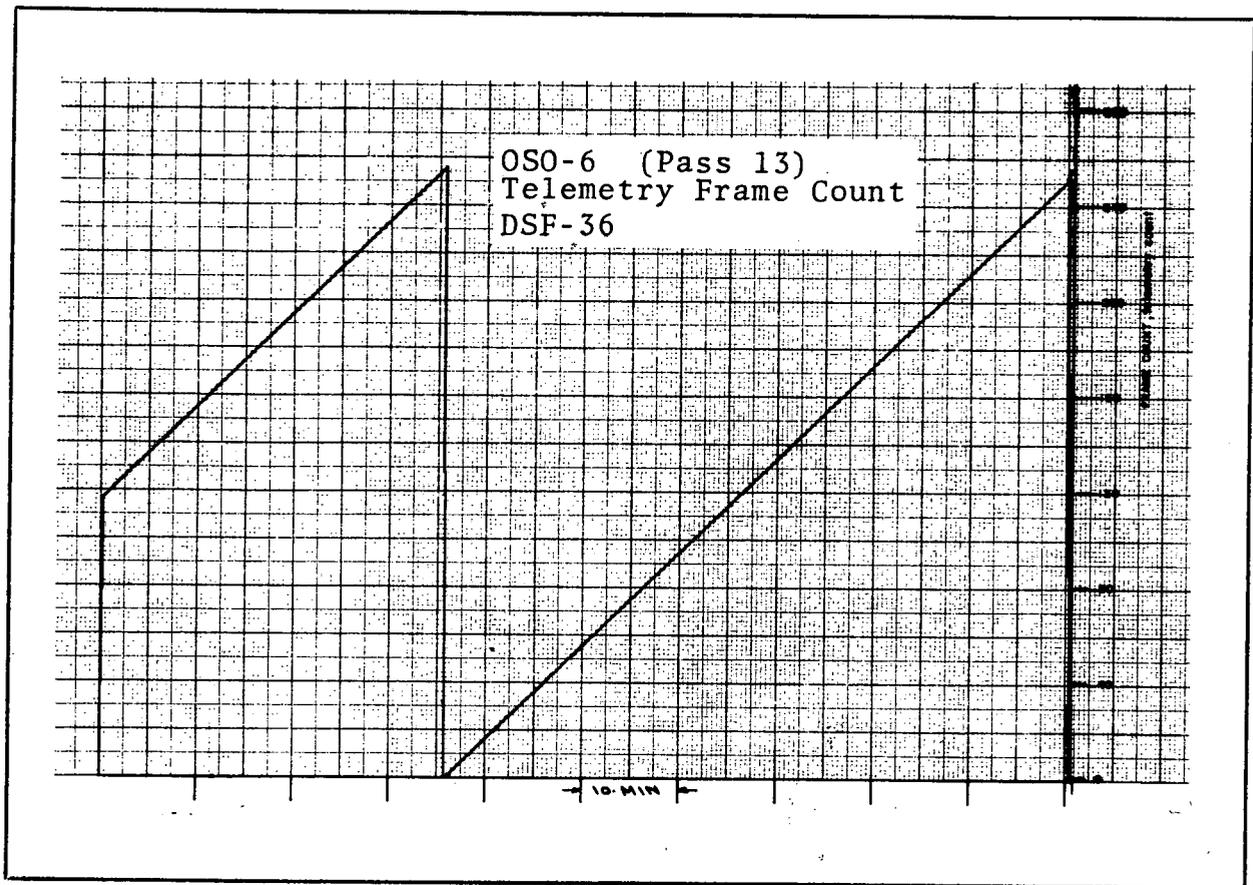
Fig. 2-7 OSO-6 Typical Orbit Profile--Frame Count

2.6   COMMAND SYSTEM IN ORBIT

The OSO-6 command system operated well throughout the first two years in orbit. Transmission of commands to the spacecraft resulted in proper execution of the planned functions.

There were numerous command anomalies reported on OSO-6 during the two year period as follows:

- During the first 6 months, 36 anomalies involving 78 command functions were reported.

- During the second 6 months, 25 anomalies involving 33 command functions were reported.

- During the third 6 months, 16 anomalies involving 20 command functions were reported.

- During the fourth 6 months, 136 anomalies involving 264 command functions were reported

All of the anomalies reported were of the type where a command function was executed which had not been sent by an OSO ground-control facility.

The command functions most often executed in the reported anomalies were:

- 46, Rutgers Manual Sequence (10100011) 69 times;

- 529, NRL Power Off (01101001) 82 times;

- 43, Rutgers Reverse Mode (10011001) 20 times;

- 465, Elevation Vernier Offset Bit (11011000) 20 times; and

- 535, HCO LV On (01111000) 17 times.

There are no apparent similarities in the binary codes for these commands which would indicate a specific design weakness in the decoder matrix.

The command anomaly data available indicate that the reported anomalies occurred during times of high radio frequency interference (RFI). Indeed, testing on OSO-type decoders had shown these decoders to be prone to generate spurious commands under conditions of high RFI. This observation has lead to refinements of the command system on OSO's after OSO-6.

The geographical distribution of command anomalies during the two-year reporting period leads to some speculation about the source of the troublesome RFI. Early in the flight the anomalies seemed to occur over the Gulf of Mexico/Caribbean Sea area. Then they seemed most frequent near the eastern coast of China and Southeast Asia. There was then a brief period when many anomalies occurred near New Guinea, after which the China/Southeast Asia area became prominent again.

2.7   POWER SYSTEM IN ORBIT

After two years of in-orbit operation, the OSO-6 power system has shown little, if any, sign of degradation in performance.

Solar array current during the two-year period ranged from about 1.98 amperes to 2.10 amperes, in phase with the solar constant. Long-term trends, if any existed, were not apparent in the orbit-noon data. The temperatures reached by the solar-cell array varied in phase with the length of the orbit-day, as determined by the precession of the orbit.

Solar-cell array degradation generally results from radiation effects, micrometeoritic bombardment, increases in temperature caused by degradation of heat-dumping paths, fractures of solar cells, connecting strips or cover slips, and delamination of cover slips. Apparently no significant loss of output resulted from any of these factors, proving the value of design measures used to minimize array degradation.

The performance of the main storage battery did not change noticeably during OSO-6's first two years of orbit life. The voltage drop at day-to-night transition and during acquisition remained essentially constant, indicating that battery internal resistance did not increase appreciably. The fall in voltage during orbit night likewise remained fairly constant, indicating that no important changes took place in battery capacity.

In the first few days after launch, the main battery showed a tendency to overcharge. The experiment instruments were not energized on a full-time basis at the time. Subsequent application of experiment loads and timely use of the dummy load prevented overcharging throughout the remainder of the two-year period.

2.8   STRUCTURES AND TEMPERATURE IN ORBIT

Throughout the orbit life of OSO-6, the spacecraft structure performed well in meeting the mission requirements. Pointed experiment settling during the launch and subsequent warpage in flight was well within specification. Performance of the azimuth and elevation bearings and the arm-support mechanisms was excellent.

Observatory temperatures were very close to the preflight predictions so no in-depth study was made. Observatory hub temperature varied cyclically between the predicted limits in phase with the length of the orbit day. All temperatures, as indicated by the on-board monitors, varied similarly within acceptable limits.

Both the annual period of the solar constant and the 50-day period
of OSO orbit precession were evident in the long-term orbit-noon
temperature plots. The solar-cell panel over the pointed instruments
showed the greatest temperature excursion, ranging between -24
and +62 degrees centigrade in a single orbit. Orbit noon data
showed a variation of about 5 degrees caused by the solar constant,
and up to 11 degrees caused by orbit precession. Compensation
for these cyclic variations in temperature showed that flight
data revealed no significant degradation in OSO-6's passive
thermal control.

The spacecraft hub temperature varied between about 8 and 18 degrees
centigrade, varying with the above-mentioned cyclic phenomena.
Variation of hub temperature in any single orbit was less than
about three degrees, well within the five-degree specified limit.

2.9   EXPERIMENT INTERFACES IN ORBIT

Throughout the OSO-6 orbital lifetime there were no verifiable
failures in the spacecraft-experiment interfaces. The
instruments occasionally received spurious commands, as did the
spacecraft. The distribution of spurious commands between the
instruments and the spacecraft functions leads to the conclusion
that the OSO-6 command decoders, as earlier OSO decoders, were
susceptible to RFI, rather than that there were any problems
with instrument-spacecraft interfaces.

The OSO-6 instrument initial-turnon history is presented in
Table 2-1.

Table 2-1
OSO-6 EXPERIMENT INITIAL TURN-ON HISTORY

Date	Orbit	Time (EDT)	Experiment	Command
9 Aug. 69	001	09:39:00	Rutgers	Squib Arm
"	"	09:39:01	Rutgers	Squib Fire
"	"	10:09:00	UCL	Squib Arm
"	"	10:09:00	UCL	Squib Fire
10 Aug. 69	010	00:17:30	HCO	LV On
"	"	00:18:38	HCO	LV Off
"	014	06:31:00	UCL	LV On
"	"	06:31:15	UCL	LV Off
"	"	06:32:00	Rutgers	LV On
"	"	06:32:05	Rutgers	LV Off
"	015	08:12:35	LASL	LV On
"	"	08:12:43	LASL	LV Off
12 Aug. 69	044	06:01:30	Bologna	LV On
"	"	06:01:35	Bologna	LV Off
"	"	06:01:45	NRL	LV On
"	"	06:03:45	NRL	LV Off
"	054	21:53:30	NRL	PM HV On
"	"	21:53:35	NRL	PM HV Off
"	"	21:53:50	NRL	Prop. Ctr. HV On
"	"	21:54:10	NRL	Bragg HV On
"	"	21:54:15	NRL	Bragg HV Off
"	"	21:54:30	NRL	Burst HV On
"	"	21:54:50	NRL	Rstr. HV On
"	"	21:54:55	NRL	Burst & Rstr HV Off
"	"	21:55:10	NRL	PM HV Off
"	"	21:56:48	HCO	HV On
"	"	21:57:08	HCO	HV Off
13 Aug. 69	056	01:12:45	UCL	HV #1 On
"	"	01:13:16	UCL	Pwr Off

Table 2-1 (continuation)

Date	Orbit	Time (EDT)	Experiment	Command
13 Aug. 69	056	01:13:40	UCL	HV #2 On
"	"	01:14:05	UCL	Pwr Off
"	"	01:14:30	UCL	HV #3 On
"	"	01:14:55	UCL	Pwr Off
"	057	01:55:46	Rutgers	HV On (Cont.)
"	070	23:17:00	UNM	Pwr On (Cont.)
"	"	23:19:30	HCO	HV On (Cont.)
"	"	23:20:30	NRL	All HV On
"	071	23:54:00	NRL	All HV Off
14 Aug. 69	072	02:39:00	UCL	All HV On (Cont.)
"	"	02:40:00	NRL	All HV On (Cont.)
"	073	03:51:34	LASL	LV On
"	"	03:51:39	LASL	HV #1 On
"	"	03:51:44	LASL	HV #1 Off
"	"	03:52:14	LASL	HV #2 On
"	"	03:52:19	LASL	HV #2 Off
"	074	05:30:11	LASL	All HV On (Cont.)
"	"	05:31:21	Bologna	HV On
"	"	05:36:51	Bologna	HV Off
"	075	06:54:20	Bologna	All Pwr On (Cont.)

LEGEND: Cont. = Experiment Turned-On for Continuous Operation
LV = Low Voltage
HV = High Voltage
PM = Photomultiplier
Prop. Ctr. = Proportional Counter
Rstr. = Raster

2.10 IN-ORBIT ONE-YEAR PERFORMANCE SUMMARY

Spacecraft performance during OSO-6's first year in orbit is summarized in Table 2-2, along with specification items which apply to in-orbit performance parameters. As can be seen by the Table, OSO-6 met all of its mission requirements.

Table 2-2

SUMMARY OF SPACECRAFT SYSTEMS PERFORMANCE
VS. MODEL SPECIFICATION FOR FIRST YEAR OF OPERATION

Spacecraft System	Model Specification	Actual Performance
In-Orbit Lifetime	6 months minimum in a 300 +50 nmi orbit inclined at 33 +3 deg. to the equator	All systems were operating satisfactorily at the end of one year.
Pointing Control	The pointed instruments shall be oriented in the fine control mode within 90 seconds after entering sunlight; while in the sun-centered or offset point modes, the azimuth and elevation pointing error shall be less than +1 min.; jitter shall be <5 sec. in el. and <3 sec. in az.; while in the raster modes the raster size shall be 46 +2 min. for large and 7 x 7.5 +0.5 min. for small.	Azimuth Alignment. The alignment error between the right-hand (reference) instrument and the left-hand instrument increased from 0.03 arc minutes left at launch to about 0.11 arc minutes left by August, 1970. Elevation Alignment. The elevation alignment error between the RH and LH instruments increased from about 0.03 arc minutes down at launch to about 18 arc minutes down by June, 1970. Raster Size. The large raster size at "mean" sun intensity was 45.2 arc min. in az. and 44.8 arc min. in el.; the small raster varied between 6.9 +0.2 arc min. for el. and 6.8 +0.2 arc min. for az. Azimuth Motor Torque. The azimuth motor torque was between 0.60 and 0.80 in.-lb. throughout the first year of orbit life. Acquisition Time. Solar acquisition consistently occurred in about 60 seconds after day-power turn-on throughout the first year of orbit life. Target Eye Degradation. The target eye readout degraded at 0.12 percent per month rate. At the end of the first year, the target eye had degraded about 1.44 percent.
Telemetry	Experiment and spacecraft data shall be continuously time multiplexed at a rate of 100 8-bit words per second; this data shall be continuously transmitted in realtime except during the playback period; during realtime transmission the data shall be continuously stored by tape recorders; the tape recorders shall have a capacity of 98 to 101 minutes; the data format shall remain as assigned; the realtime data rate	Tape Recorders. Both recorders operated properly during first year of orbit life; the pbk. rate was 14.4 K bps +10 bps and the pbk. time was nominal. Transmitters. Both transmitters operated success-fully throughout the first year of orbit life. The signal strength (with diversity combining) was nominal and varied between -85 and -116 dbm. Data Handling Subsystem. Both data handling sub-systems operated successfully throughout the first year of orbit life.

Table 2-2 (continuation)

Spacecraft System	Model Specification	Actual Performance
	shall be 800 +0.4 bps; the playback rate shall be 14400 +288 bps; the playback period shall be 5.64 +0.083 minutes; the xmtr. carrier frequency shall remain within 5 kHz of assigned; the carrier shall be PSK modulated with a deviation of 57 +5 deg. for both realtime and playback transmission.	Receivers. Both of the command receivers operated successfully throughout the first year of orbit life.
Command	Any one command sent to the spacecraft results in the execution of a planned function.	Decoders. All four command decoders operated successfully throughout the first year of orbit life. The only problems noted were a few spurious commands that were accepted. The specific causes were not determined.
Power	19 +3 vdc shall be provided to the spacecraft during both day and night; the output capacity shall be 1.87 amp-hr./orbit minimum; the peak load capacity shall be 8 amps for 5 seconds duration; undervoltage protection shall be provided at 16.2 +0.2 vdc with power being automatically reapplied at 19 +0.2 vdc.	Main Battery. The spacecraft main battery operated successfully throughout the first year of orbit life. The battery noontime voltage varied between 18.2 and 20.78 for the "hottest" and "coldest" orbits, respectively. The "dummy load" was used to keep the battery voltage typically between 18 and 20 volts during each orbit. Solar Array. The solar array power output remained above the specified value and no degradation was noticed. Undervoltage Protection. An undervoltage condition did not occur during the first year of orbit operation and, therefore, this feature was not checked.
Pitch and Roll Control	Automatic pitch control to within +4 deg.; pitch gas pressure of 3000 (+100, -200) psi at launch. No roll requirement.	Pneumatic Pitch Control. The pneumatic pitch control system operated successfully throughout the first year of orbit life. A total of 48 manual pneumatic pitch corrections were performed in the first year with an average gas consumption of about 14 psi/degree of pitch motion. The usefulness of the magnetic coils was apparent in that only 600 of the original 2800 psi of pitch gas was consumed in the

Table 2-2 (continuation)

Spacecraft System	Model Specification	Actual Performance
		first year. There were no automatic pitch corrections made during the first year and therefore data is not available for this mode of operation. Pitch and Roll Coils. The pitch and roll coils were operated successfully throughout the orbit life and accounted for the greatly reduced consumption rate of pitch gas. Also, the roll coils were used several times to change the roll attitude.
Spin Control	Automatic spin rate control to within 0.44 to 0.68 rps; the spin rate change per correction shall be 0.028 $\pm$0.01 rps; switch to manual thresholds shall be 0.39 $\pm$0.012 and 0.7 $\pm$0.015 rps.	The pneumatic spin control system operated successfully throughout the first year of orbit life. About 300 of the original 2800 psi of spin gas was used to initially despin the wheel from the third-stage spin rate of 1.63 rps (97.9 rpm) to 0.613 rps. During the first year, six automatic spin-up, one manual spin-up, and 21 manual spin-down maneuvers occurred. The average amount of spin gas used per burst was about 14 psi. The spin gas supply at the end of one year was about 2050 psi.
Launch Sequence	The spacecraft shall be placed in the normal orbit mode of operation within 1400 seconds after third-stage spinup.	The launch sequence system operated successfully with all launch sequence functions occurring at or near the correct time.
Aspect Determination	The roll angle shall be determined within $\pm$3 deg.	The aspect system met the following mission objectives: (1) Provided the information necessary for determining the observatory roll attitude to 1 deg. accuracy. (2) Provided information for deriving the wheel spin rate. (3) Provided information for deriving the wheel azimuth position as a function of time. For the first year of in-orbit operation, the OSO-6 spin axis was maintained between 60 and 180 degrees (referenced to ecliptic north).

Table 2-2 (continuation)

Spacecraft System	Model Specification	Actual Performance
Thermal Control	The internal temperature of the wheel compartments shall not deviate more than 5°C per orbit and the mean temperature shall be 15 ±5°C.	The thermal control schemes maintained the wheel in-orbit temperatures very close to those desired.
Mass Balance	As required to meet other design requirements.	About 0.2 degrees of wobble was detected; the cause has not been determined and it did not adversely affect the operation of other spacecraft systems.
Magnetic Balance	As required to meet other design requirements.	The prelaunch magnetic balance adjustments made on OSO-6 produced very nearly the desired effects.

Section 3
OSO-6 DEVELOPMENT AND SUPPORT HISTORY

3.1 · INTRODUCTION

The technical effort and program planning behind the OSO-G
Orbiting Solar Observatory (OSO-6) are presented in this section.
The various aspects of the project are discussed, together with
details of their implementation. These include the planning,
test programs, development, support activities, and post-launch
follow-up efforts.

Fabrication, assembly, integration, and testing of OSO-6 was
undertaken by Ball Brothers Research Corporation (BBRC) under
Contract NAS5-9300 issued by NASA/GSFC. A prototype/qualification
model was not included in the project.

3.2 MANAGEMENT

The OSO-6 project was conducted at the BBRC Boulder facilities
using essentially the same organization structure, policies, and
procedures as other OSO projects. Technical direction was
assigned to the existing OSO project staff. Experienced members
of the OSO-5 launch team and key members of other BBRC organiza-
tions assisted the Program Manager in completing the OSO-6
project. Work was accomplished in compliance with the applicable
portions of NASA Quality Assurance Publication NPC-200-2, and
NASA "Reliability" Publication NPC-250-1. The spacecraft's
progress was monitored by the project staff with the aid of PERT
Status and Cost Control Reports.

Established BBRC policies and procedures regarding work authoriza-
tion, security of classified data, receiving, shipping, control
of nonconforming supplies, project stores control, property
control, measurement control, engineering record and release,
etc., were used in the OSO-6 program.

3.3   SCHEDULE

Milestone events are listed in Table 1-2 of this report. More detailed breakdown of the OSO-G development and support history is presented in the paragraphs which follow.

3.4   DESIGN REVIEW AND HARDWARE SELECTION

Before hardware fabrication began on the OSO-6 project, a critical review of designs as reflected in documents and drawings from earlier OSO's was conducted. The purpose of this review was to determine the adequacy of this documentation to meet the intent of NASA Document NPC-200-2, Quality Program Provisions for Space System Contractors, and to ensure that the designs were compatible with specific mission requirements.

The major considerations and purposes in the design review before component fabrication and procurement were:

(1)   To develop the basic design definition and documentation for each component, subassembly, and assembly

(2)   To ensure that current design definitions incorporated necessary changes identified in the documentation of previous OSO projects, including failure reports, liaison engineering actions, and engineering orders

(3)   To ensure that all component, subsystem, and system requirements were covered and that the electrical, thermal, and mechanical interfaces were compatible

(4) To estimate the relative criticality of
equipment package input-output circuitry
and its vulnerability to failure

(5) To define possible failure modes and assure
that assemblies were interfaced on a
noninterference basis

(6) To provide drawings in a form suitable
for OSO-G spacecraft systems analysis work

(7) To consider weight, balance, materials, thermal,
vibration, and stress properties

(8) To verify that hardware adjustment methods,
test methods, installation methods, etc.
did not have adverse effects on observatory
operational integrity

(9) To verify that design documentation was of
sufficient quality to allow proper fabrication,
testing, and quality control of each article,
and

(10) To compile a record of all information
developed during the review and the complete
revised documentation for design approval
and release.

The design review so described was a joint effort of personnel
from the Systems Engineering, Quality Assurance and Reliability,
Materials, Production, Design, and Integration and Test depart-
ments. After the design review of a particular subsystem was
completed, it was processed through drawing release so that
production work could proceed during review of other subsystems.

Table 3-1 presents a summary identification of the hardware
flown on OSO-6 as a result of the design review and subsequent
spacecraft development. Items are identified by drawing numbers,
serial numbers, specifications, and procedures as appropriate.
Functional changes from prior OSO configurations are discussed
in later sections covering descriptions of spacecraft systems.

3.5   COMPONENT FABRICATION AND TEST

Component procurement, fabrication, and test was carried out in
accordance with a prescribed reliability program (Reliability
Program Plan for the OSO Spacecraft). Long-lead-time items
were started as early as possible. Procurement of OSO-6
hardware started immediately after design release.

A parts program, providing the best available types and highest
quality parts, governed procurement of components. Reliability
considerations were based on previous part history, NASA and
major space program preferred-parts selection, and audits of
currently available types and qualification data. Specifications
were tightened to screen out all but the best units that can be
made with a given process.

Specification-control drawings (SCD) identified part requirements,
source, type and manufacturer, acceptance or rejection criteria,
and tied specific parts into assembly drawings and test
specifications.

Tests included parameter measurement at room temperature, thermal
cycling, power aging and retest, x-ray, high-temperature storage,
and other tests as appropriate. Special fixtures constructed
for use in earlier OSO programs were used to expedite component
qualification and acceptance.

Table 3-1

OSO-6 SPACECRAFT SUBASSEMBLY IDENTIFICATION

Assembly/Subassembly	Serial No.	Drawing No.	Documentation/Notes
Wheel Structure	2	19291-501	MS26493, TS26494
Tube Assy.	2	15481-1	
Arm Installation	45359-1, -4, -7	15495-1	
Rib Assy.	45261	19215-1	
Deck Assy.	37358	19522-1,-501, 503,505	
Purge System	45275	19555-1	
Az Bearing, Lower	122	20324-1	TS18788, TP18789, TN66-82
Az Bearing, Upper	136	20323-1	TS18788, TP18789, TN66-82
Latch Assy, Squib	276, 277, 279	15551-1	(Arm Latches)
Sail Structure	1889A	19302-507	
Latch Assy, Squib	268 (Elev)	15551-1	
Latch Assy, Squib	175 (Nut.)	15559-1	
Elevation Frame	41509	18266-1	
Elevation Bearing	117, 120	20325-1	TS18788, TP18789, TN67-174
Sail Frame Assy.	37329, 37327	19303-1, -502	
Panel Cntr. Front	37324	19307-1	
Panel Cntr. Back	3732	19309-1	
Az Frame Assy.	45016	19534-1	
Nutation Damper	2	19558-501	TS15610, AP15646, QP15614, TN68-130
Pneumatic System	12	19535-501	TS15904, TS19582, TP16704, AP17018, TN68-157
Tubing Assy.		19535-501	TS19582, AP16920, QP15647, TN68-157
Cap Quick Disconn.	28509, 28298	21992-1	TS21999, TP22200, TN67-57
Spin Gas Install.	1920A	19538-501	
Spin Gas Bottle	22,27,29	17518-1	TS15606, TS15904, TN67-171

Table 3-1 (continuation)

Assembly/Subassembly	Serial No.	Drawing No.	Documentation/Notes
Press. Transducer	19339,19340	8380-1	TS15604, TS15904, TP16712, TN67-140, TN68-101
Press. Regulator	108, 109	19549-1	TS15603, TS15904, AP16729, TN67-139, TN68-160
Solenoid Valve	114,115, 116,117	19548-1	TS15602, TS15904, AP16728, TN67-141, TN68-102, TN69-109
Check Valve	105, 106	21884-1	TS15601, QP15642, AP16727, TN67-120, TN68-90
Pitch Gas Install.	1959A	19539-501	
Pitch Gas Bottle	13	17517-1	TS15605, TS15904, TP20354, TN65-268, TN67-170
Electrical Distr.			
Wheel		21964-1	
Temperature Probe	22982, 4507, 105, 22972, 22981, 7661, 22973	7737-503	MS18804, TS18805, TP22888, TN69-105
Slip Ring Install	11	15491-1 16093-1	TS16248, TP16249, TN67-144, TN65-250
Terminal Boards	47271, 47272	20835-1	
	1	21993-1	
	1	21965-1	TS21987, TN68-139
	1	21966-1	TS21988, TN68-138
	1	21967-1	TS21989, TN68-137
	1	21968-1	TS21990, TN68-136
	2	21969-1	TS21991, TN68-135
	1	21970-1	
	1	21068-1	
	1	21068-501	
	1	21068-503	
Sail	12	21963-1	TS24242
Flex Cable Assy.	200	30128-1, -501,-503, -505,-507	TS24241, TN69-93, TN69-114, TR69-51
Sail Junction Box	3	21790-1	TS24243, TR68-69, TN68-163, TN69-113
Squib Term. Bd.	2G	20933-1	
Temperature Probe	22977,4708, 4709,4710, 7670,129	7737-503	MS18804, TS18805, TP22888, TN69-105

Table 3-1 (continuation)

Assembly/Subassembly	Serial No.	Drawing No.	Documentation/Notes
RF Choke Module	NRL-2	22684-1	TS24245, TN68-131, TN69-104, TR68-73
Filter Module-HCO	HCO-2	30461-1	TS31225, TN69-69, TR68-68
Overload Protection	2	30750-1	TS31351, TN69-83, TR69-56
Electronic Cont. - Wheel	12	19578-505	
Spin Control Assy.	1	22002-1	MS22194, TS22195, TP22196, TN68-156
Temperature Probe	102,117, 126	7685-503	MS18804, TS18805, TP22888, TN69-106, TN69-105
Spin Sensor Assy.	15	16082-501	MS18818, TS18819, TP18820, TN67-132
Pitch Coil Assy.	41695	15784-1	TS15697, QP15698, AP15699, TN67-62
Electronic Cont. - Sail	6357A	23645-1	
Pitch Control Sensor	15	7865-501	MS18766, TS18821, TP18822, TN68-89
Power Amp Assy.	3	21554-1	MS21556, TS21557, TP21558, TN68-140, TR68-34
15V Regulator		19632-1,-501, -503,21600-1	MS19469, TS19470, TP19471, TP15476
Temperature Probe	29274, 29277, 8	7685-503, -507	MS18804, TS18805, TP22888, TN69-106
Servo Amp Assy.	3	21581-1	MS21583, TS21584, TP21585, TR68-35, TN68-133
Point Readout Sensor	12	15535-1	MS16689, TS18810, TP18811, TN69-39
Fine Control Sensor	14	15536-503	MS16686, TS18810, TP18811, TN69-41, TN68-155
Coarse Sensor	15	16215-1,-2, -501,-502	MS18815, TS18816, TP18817, TN68-116
Az Torque Motor	64-572-22	19172-1	TS18791, TP18792, TN68-48
El Torque Motor	64-573-9	19173-1	TS18791, TP18792, TN68-70
Pitch Readout Sensor	14	8232-501	MS18589, TS18807, TP18808, TN67-138, TN68-93
Scan Readout Sensor	16	19053-505	MS18812, TS18810, TP18811, TN68-150, TN69-38

Table 3-1 (continuation)

Assembly/Subassembly	Serial No.	Drawing No.	Documentation/Notes
Roll Coil Assy.	3A, 2B	30823-1,-2	TS31379, TP31561, TN69-94
Launch Sequence System	1885	19563-503	
Launch Seq. Timer	1190,1191	18851-5	TS16387, TP16388, TN67-152, TN67-153
Press. Cartridge	168, 175 131, 133 164,165, 166,167, 176,178	21974-501 21974-1 21112-1	TS21154, TP22403, TN67-176
Sail Squib Batt.	13, 14	19592-1	DS15864, TS15862, TP15863, TS15860, TP15861, TN69-86
Wheel Squib Batt.	7, 8	19590-501	DS15864, TS15862, TP15863, TS15860, TP15861, TN69-88
Nut. Damper Mon. Sw.	2	28334-1	TS28542, TN69-110
Separation Sw.	2	30884-1	TS30883, TN69-95, TR69-45
Data Handling System	1933A	17416-501	
PCM J-Box	1	22019-1	MS22027, TS22026, TP22028, TN68-147
Frame Counter	1	21942-1	DS17317, TS21956, TP21957, TN68-143
FC Pwr. Supply	601003	16123-3	DS20227, TS20228, TP20290, TN67-130
Tape Recorder	805,808	30418-1	TS30415, TS30416, TP30417, TN68-104
DME-PCM	8; 10	19571-1	DS15342, TS15343, TP17290, TN68-123, TN68-124
ASC-PCM	8, 13	19572-1	DS15342, TS15343, TP17290, TN67-110, TN68-125
ASSC	3	17208-501	DS17191, TS17192, TP21103, TN67-189, TN68-118
Word Gate Gen.	2	28000-501	DS22743, TS22744, TP30776, TN69-76, TR69-22

Table 3-1 (conclusion)

Assembly/Subassembly	Serial No.	Drawing No.	Documentation/Notes
Command System	1932A	17417-501	
Command Receiver	003, 004	20819-1	DS20818, TS20820, TP20821, TN69-78
Command Decoder	G1A, G2A, G3A, G4A	16578-3	DS16846, TS21933, TP22285, TN68-148, TN69-43
RF System	1934A	17415-501	MS21042, TS21043, TP21044, TN69-92
VHF Antenna (Match)	41829, 27286	17350-1	
VHF Transmitter	15, 17	31697-501, 20171-501	DS20329, TS20330, TP20331, TS31698, TP31699, TN68-74, TN68-76
Diplexer Assy.	17	20096-501	MS16116, TS16117, TP16437, TN68-88
Hybrid Circulator	8	17361-1	MS17456, TS17457, TP17458, TN68-87
RF Cable Assy.			MS21042, TS21043, TS21044
Coaxial Relay	2A	21113-1	DS21151, TS21152, TP21153, TN67-194, TR67-83
Aspect Meas. System	12	19568-501	
SORE Package	2	15874-501	MS18784, TS18785, TP16397, TN67-156
Mag. Sensor	1606	16995-1	DS18823, TS18824
Mag. Electronics	1606	19343-1	TP16785, TN68-71
Power System	1900A	19559-503	
Current Monitor Assy.	6, 4F	25106-1	MS27379, TS27380, TP27381, TN67-101, TN68-142
Solar Cell Array	004	17036-1	DS17042, TS20586, TP20587, TN69-107
Power Cell Assy.	14,15,16, 17,18,19	17292-1, 17292-503	TS18872, TP18873, DS17149, TS15856, TP15857, TN69-98

Program personnel visited vendor facilities as necessary to
conduct quality surveys and vendor evaluation. These efforts
provided criteria for placement of orders and subcontracts.
Further visits expedited delivery of items.

Quality assurance test and inspection records, failure reports,
and other documentation related to items served to document the
history of those items. Records of operating and storage time
were kept and used to evaluate items subject to wearout and
fatigue.

Major OSO assemblies which were qualified on previous programs
and substantially changed for OSO-6 were requalified when
necessary. System reliability, failure-mode, and criticality
analyses were updated at predetermined milestones during the
design, fabrication, and test programs.

BBRC personnel were trained to comply with MSFC-PROC-158B
soldering specification. All flight-model printed-circuit boards
were epoxy conformal-coated after tests showed that such coating
allows two to three times as much power dissipation in certain
components. A continuing program of laboratory analyses and
tests by the materials and processes section assisted in
the resolution of design and production problems.

Acceptance testing of finished subassemblies was accomplished
using production-test stations, designed and fabricated for
that purpose on this and earlier OSO programs. Use of these
stations provided increased efficiency and component safety in
testing.

3.6 SPACECRAFT INTEGRATION AND TEST

The spacecraft integration and test phase covered the period from the delivery of the basic sail and wheel structures to the integration and test (I&T) organization to the partial (90 percent) spacecraft buyoff on 29 November 1968. Before such delivery, integration and test personnel reviewed functional drawings and wiring diagrams and prepared and adapted test equipment and procedures for use on OSO-6.

In simplified terms, the activities performed in the spacecraft integration and test phase were:

(1) Checkout wiring of assemblies before interfacing

(2) Alignment of sensors and equipment

(3) Installation of equipment in the sail and wheel structures

(4) Data collection for dynamic and long-term performance analysis

(5) Mating of sail, wheel, and azimuth shaft assemblies

(6) Preparation and installation of dummy pointed instruments to support sail components until installation of flight instruments

(7) Preparation and execution of tests to cover the applicable spacecraft test specifications

(8) Identification and resolution of potential and actual trouble areas

(9)   Preparation and maintenance of test equipment
      supporting the above activities, and

(10) Spacecraft maintenance.

The OSO-6 spacecraft was tested in accordance with a formal test specification (Dwg. 26494, Test Specification for the OSO-G Spacecraft). All measurable parameters were recorded and analyzed, including the spacecraft functions at the experiment-interface connectors. A summary of spacecraft I&T activity is presented in Figures 3-1 through 3-3.

The spacecraft I&T phase ended with the 90-percent spacecraft acceptance on 29 November 1968. Certain items of spacecraft integration were carried forward for completion during the observatory I&T phase. These were necessitated by late changes to the contract scope-of-work and late delivery of command receivers. The late hardware changes related to offset point mode circuits, word-gate generator development, and adaptation of the launch-sequence system for operation with a two-stage launch vehicle.

3.7   EXPERIMENT ACCOMMODATION

Experiment accommodations·personnel worked with spacecraft and experiment designers to accomplish:

(1)   Conciliation of experiment operational require-
      ments and spacecraft dynamic, dimensional, power,
      command, and telemetry constraints

(2)   Definition of experiment/spacecraft electrical
      and mechanical interfaces

WHEEL DELIVERY
6/26/68

WHEEL RINGOUT
6/27-73/68

CMD ROUTING VERIFICATION
7/2-7/3/68

BUS Z & VOLTAGE MEAS.

VOLTAGE & RES. VERIFICATION AT INTERFACE

SUBASSEMBLY INSTALLATION
7/3

LOAD I MEAS. & FUNCTIONAL TESTS

EXP. INTERFACE VERIFICATION
7/11/68

SUPPL. TESTS (PRELIM, BUYOFF)
7/15-10/4/68

BATTERY CONDITIONING
7/15-7/19/68

SUB COMM CALIBRATION

ANTENNA MATCHING
7/22-9/18/68

CONT.

REWIRED O4TBI TO CONFORM W/CKT LIST
REWIRED CONSOLE CONNECTORS
REWIRED DECODER HARNESS
CLEANED ALL CANNON CONNECTORS W/FREON
REPLACED O4RT4 TEMP. PROBE
REMOVED RF SUBSYSTEM FOR PROD. TEST
INSTALLED ARM DAMPER SYSTEM
REWIRED RCVRS & DECODER CABLES FOR PROPER GROUNDS

SPACECRAFT PROBLEMS
1. BATTERY & CONT. BUS TIED TOGETHER ON O4TBI
2. CONSOLE CONNECTORS WOULD NOT INTERFACE WITH SPACECRAFT
3. CONNECTOR SEX REVERSED ON DECODER HARNESS
4. BAD SOLDER CONNECTIONS ON CURRENT MONITORS
5. CONTAMINATION ON SOLDER TYPE CANNON CONNECTORS
6. NO OUTPUT FROM TEMP. PROBE O4RT4
7. COAX SWITCH WIRED BACKWARDS

1. GALLING ON RF CONNECTORS
2. ARM LOCKED MONITOR READS IMPROPERLY
3. SPIN DOWN JET CONNECTED TO SPIN UP CKT
4. SPIN UP BURST OCCURS WHEN SWITCHING TO NITE
5. MAG PULSE OCCURS AT WRONG INTERVAL
6. POOR INTERFACE ON ARM NO.I RF CABLE
7. BROKEN TERMINAL ON CHARGE I MONITOR
8. BROKEN TERMINAL ON PITCH COIL
9. XMTR I RF CABLE CONNECTOR BROKEN
IQ. PCM J-BOX PIN BROKEN

Fig. 3-1 OSO-G Wheel Pre-Mate I and T Activity

- REPLACED S/N 4711 TEMP PROBE
- ISOLATED GROUNDS (7/29 - 8/5)
- REDUCED LINE C FOR SASC TRIGGER (BYPASSED J-BOX)

INSTALLED:
- DUMMY POINTEDS
- ENGR. J-BOX
- ENGR. SERVO BOX
- ENGR. PWR BOX
- DUMMY ARRAY
- CONTROL EYES (TEST SN 11)
- SCAN EYES (F SPARE SN 13)
- R.O. EYES (TEST SN 7303)

- C/F RESISTOR CHANGED TO 5.1K
- CHANGED CHG. I MON RES TO 6.2K
- REWIRED TERM BDS TO COMPLY WITH CRTS LIST
- REMOVE ALL SAIL SUBASSY FOR OVERVOLTAGE ANALY
- REWIRE SOLAR ARRAY TO C/L
- REPLACED ALL CABLE CLAMPS

SAIL DELIVERY — 7/26/68

RINGOUT & FUNCTIONAL TESTS — 7/26 - 8/8
- CMD VERIFY W/ DUMMY DEC. — 8/8/68
- BUS VOLTAGE & Z MEAS.
- V & R AT ALL CONNECTORS — 8/8 - 8/10

SUBASSY INSTALLATION & LOAD I MEAS. — 8/7 - 8/21
- CMD VERIFY W/ FLIGHT DEC — 8/12
- SUB COMM CALIBRATION — 8/13 - 8/28
- WGG FUNCTIONAL TESTS — 8/14 - 8/16

INITIAL SUN POINTING — 8/21 - 9/1
- OFFSET MODE PROGRAM (PMS) — 8/26
- SCALER OUTPUT EVALUATION — 8/26
- DIAG. LINE EVAL AT OFFSET SCAN GEN. (538 & 539) — 8/27

PRELIM. SUPPL. TESTS — 9/10 - 10/2
- INSTAL & TEST SOLAR ARRAY — 9/16 - 9/24
- INSTAL FLIGHT PWR & SERVO — 9/21/68
- INSTALL & WIRE TEST SQUIB BAT. — 9/16

CONT.

SPACECRAFT PROBLEMS
1. GROUNDS NOT ISOLATED
2. TEMP PROBE (IORT3) HAD NO OUTPUT
3. SASC TRIGGER INP LOADED BY EXCESS LINE C
4. S/F RATE PULSE TO WGG NO.2 NOT PARALLEL AT J-BOX

1. NO RASTER (M/F 32 INTEGRATOR TC TOO LONG)
2. ALL CABLE CLAMPS NON FLIGHT
3. ASSC NOT PAINTED PROPERLY
4. AZ COARSE AND FINE POLARITY REVERSAL
5. SCAN GEN POLARITY REVERSAL
6. SOLAR INTENSITY MONITOR RANGE HIGH
7. PITCH SOLENOID CONNECTORS REVERSED
8. C/F RESISTOR VALUE HIGH
9. SOLAR ARRAY CENTER CKT WIRED TO S.G. INSTEAD OF P.G.
10. WRONG NO. OF WIRES ON TERM BOARD TERMINALS

Fig. 3-2 OSO-G Sail Pre-Mate I and T Activity

Ball

REWIRED CONNECTOR TO WGG
INSTALLED FLIGHT PWR & SERVO (11/6)
INSTALL RF SUBSYSTEM (10/7/69)
INSTALL SPIN BOX (10/8/69)
INSTALL SAIL J-BOX (10/14/69)
INSULATED PWR & SERVO TORROID MOUNTING BOLT
REVISE MOTOR I MEAS. METHOD (USE DC SUN)
REVISE TRIGGER ANGLE TEST TO 50% SOLAR INTENSITY

MATE SAIL TO WHEEL 10/4-10/5	BUYOFF TO TN68-05 10/10-11/29	T.V. LEAK RATE TEST 11/14-11/18/68	OFFSET MODE EVALUATION 11/19-11/28	S/C ACCEPT. 11/29/68 CONT.
	EXP INTERFACE VERIF. 10/17-10/24/68	SUN POINTING 11/6-11/29/68	REPEATABILITY DMS (3 RUNS) 11/19	
	ARMS LOCKED TEST 10/8/69	SENSOR ALIGN & CALIBRATION 11/2-11/6	SCALER OUTPUT PROGRAM 11/21	
	BATTERY CONDITIONING 10/11-11/1/68	PITCH CONTROL TESTS 11/6-11/19	XYY PLOTTER RASTER EVAL. 11/22-11/24	
	DATA HANDLING TESTS 10/10-10/15	SUB COMM. CALIB. 11/20-11/28	SACRED PT & TURN AROUND SUBROUTINES 11/25	

SPACECRAFT PROBLEMS

1. BROKEN WIRE IN WGG CONNECTOR

1. EL MOTOR CURRENT SURGE AND POWER SWITCH OSCILLATION

Fig. 3-3 OSO-G Spacecraft Integration and Test

(3) Preparation and execution of experiment interface compliance test procedures

(4) Development of policies and procedures for handling experiment hardware (including environmental controls)

(5) Development of experiment test hardware as needed

(6) Installation of experiment hardware in the OSO-6 spacecraft (and removal when necessary)

(7) Preparation of experiment test procedures and their execution as a part of observatory acceptance testing

(8) Maintenance, repair, and modification of instruments, and

(9) Development of documentation to support and report experiment activities.

Highlights of the experiment-accommodation effort are presented in this subsection.

3.7.1   Experimenter Coordination

Purpose. The program experiment accommodation personnel maintained constant liaison with the experimenters' organizations throughout the OSO-6 effort. The purpose of this liaison was to ensure continued compatibility of the experiments with the OSO spacecraft and with each other.

Areas of consideration involving the conciliation of experiment operational requirements and spacecraft constraints were:

(1) Power budgeting, load characteristics, and surges

(2) Mechanical interfaces, weight, balance, and center-of-gravity locations

(3) Data word and command allocations

(4) Electrical interface pin lists

(5) Non-standard signal allocations

(6) Radioactive-source requirements

(7) Specification waivers and changes

(8) Special-handling and environmental requirements

(9) Test procedures and equipment

(10) Spacecraft components mounted to, and signals passing through, the pointed instruments, and

(11) Magnetic interfaces between spacecraft and experiments.

The initial OSO-6 experimenters meeting was held on 7 October 1965, with GSFC, BBRC, and experimenter representatives attending. Initial allocations and interface definitions were established at this meeting and published in TN65-307, OSO-G Experimenters' Meeting Summary.

Detailed development of the allocations and definitions was thereafter promulgated in TN66-40, the OSO-G Interface Definition Summary (IDS). Evolution of interface specification was coordinated by BBRC with approval by GSFC, and the IDS was updated as necessary.

Handling and test procedures were prepared for each instrument. Table 3-2 shows the evolution of power, command, and telemetry requirements and weight for each instrument. Specific command and telemetry channel assignments are presented in Sections 9 and 10 of this document. To simplify the interpretation of experiment test data, special computerized decommutation and print routines were developed for each experiment (see Section 4).

3.7.2   Experiment Preintegration Testing

During the spacecraft acceptance-test phase, all spacecraft interfaces which would mate with experiment hardware were verified to meet the applicable specifications. Upon receipt of the scientific instruments, each instrument was tested in a spacecraft simulator to verify compatibility with OSO-type electrical signals. Mechanical interface checks were made by trial installation in the spacecraft.

The spacecraft simulator consisted basically of a PCM simulator, power supplies for both the experiments and the simulator, a binary-to-decimal converter, a Franklin printer, and an oscilloscope. Figure 3-4 shows a photograph of the simulator. Details of operation of the simulator are provided in TM70-05, OSO Spacecraft Simulator Model II Operator's Manual.

Table 3-2
OSO-6 EXPERIMENT POWER, COMMAND, TELEMETRY, WEIGHT

Experiment		Day Pwr Circuit ma-hrs/orbit	Orbit Pwr Circuit ma-hrs/orbit Day	Orbit Pwr Circuit ma-hrs/orbit Night	MF Words	DSM Words	ASC-1 Words	ASC-2 Words	ASSC Words	Sail Cmmd	Wheel Cmmd	Weight Pounds
HCO (Ptd)	Requested	131.5	10.5	7.9	4	0	--	6	--	20	--	35
	Orig. Alloc.	131.5	10.5	7.9	8	4	--	4	6	28	--	40
	Final Alloc.	131.5	10.5	7.9	8	0	--	4	7	41	--	41.6
	Actual Meas.	113.84	9.6	7.3	--	--	--	--	--	--	--	36.16
NRL (Ptd)	Requested	191.1	--	--	14	4	--	2	--	7	--	36
	Orig. Alloc.	191.1	--	--	6	8	--	10	4	24	--	44
	Final Alloc.	191.1	--	--	6	8	--	10	0	24	--	44.0
	Actual Meas.	182.1	--	--	--	--	--	--	--	--	--	49.92
Rutgers (Wheel)	Requested	--	52.6	39.5	1	0	2	--	--	--	7	28
	Orig. Alloc.	--	52.6	39.5	2	0	2	--	--	--	7	28
	Final Alloc.	34.6	57.5	43.1	2	1	2	--	--	--	7	33.0
	Actual Meas.	41.0	78.0	58.5	--	--	--	--	--	--	--	33.68
Los Alamos (Wheel)	Requested	52.6	--	--	1	0	2	--	--	--	4	10
	Orig. Alloc.	52.6	--	--	1	0	3	--	--	--	6	10
	Final Alloc.	70.3	--	--	1	0	3	--	--	--	8	19.0
	Actual Meas.	59.4	--	--	--	--	--	--	--	--	--	20.45
U. of Bologna (Wheel)	Requested	--	57.9	43.4	1	0	2	--	--	--	3	10
	Orig. Alloc.	--	57.9	43.4	5	2	5	--	--	--	8	25
	Final Alloc.	--	57.9	43.4	5	2	5	--	--	--	7	25.0
	Actual Meas.	--	41.8	31.3	--	--	--	--	--	--	--	23.60
UCL (Wheel)	Requested	31.6	--	--	1	0(S/B)	4	--	--	--	3	25
	Orig. Alloc.	31.6	--	--	1	3(S/B)	3	--	--	--	3	30
	Final Alloc.	31.6	--	--	1	10	3	--	--	--	7	36.0
	Actual Meas.	19.3	--	--	--	--	--	--	--	--	--	22.70
U. of N. Mex. (Wheel)	Requested	--	73.0	54.7	1	0	4	--	--	--	2	25
	Orig. Alloc.	--	73.0	54.7	4	0	6	--	--	--	2	25
	Final Alloc.	--	73.0	54.7	4	12	7	--	--	--	2	32.5
	Actual Meas.	--	83.6	62.7	--	--	--	--	--	--	--	32.25
Experiment Totals	Requested	406.8	194.0	145.5	23	4	14	8	--	27	19	169
	Orig. Alloc.	406.8	194.0	145.5	27	17	19	14	10	52	26	202
	Final Alloc.	459.1	198.9	149.1	27	33	20	14	7	65	31	231.1
	Actual Meas.	415.64	213.0	159.8	--	--	--	--	--	--	--	218.76
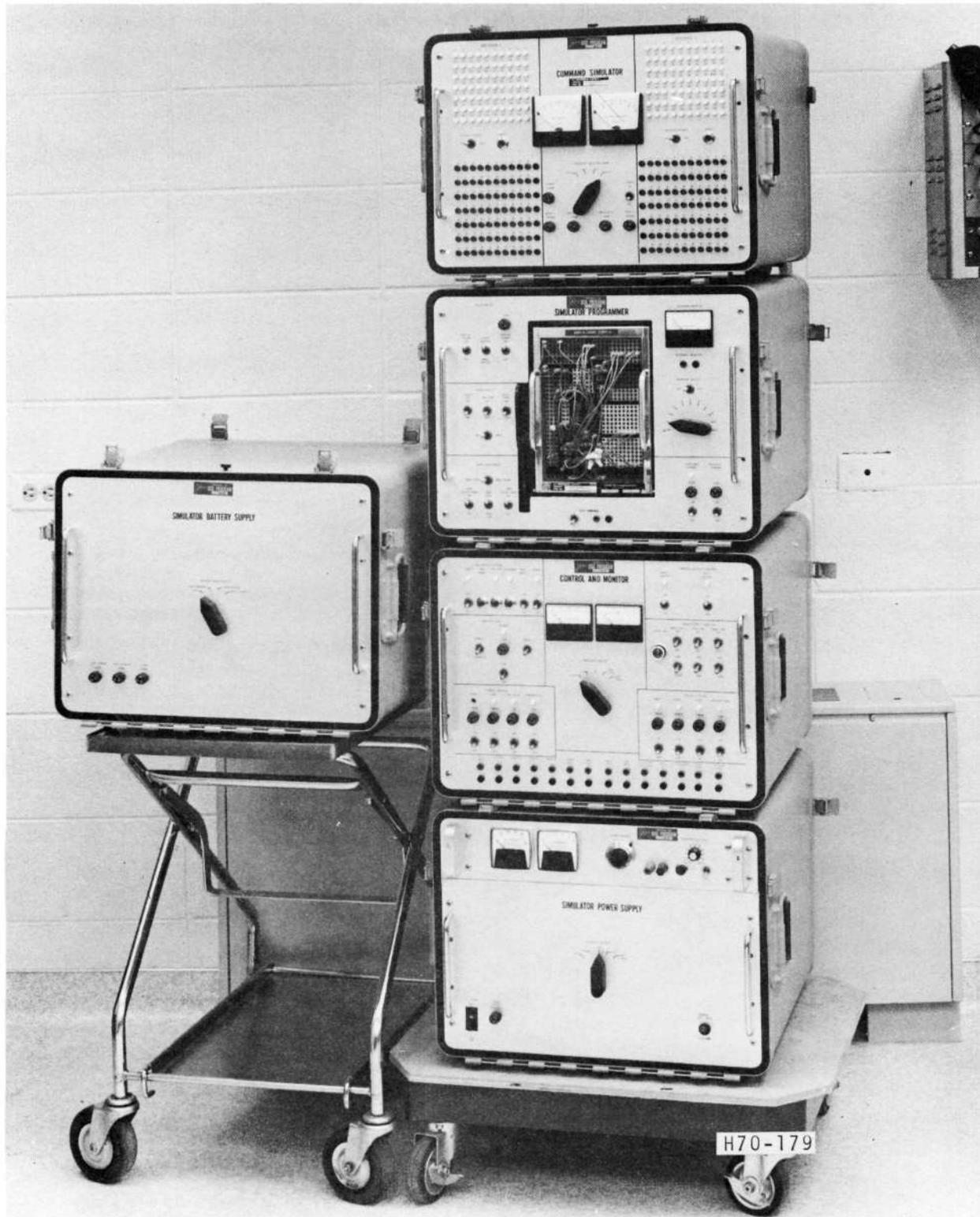
Fig. 3-4   OSO Spacecraft Simulator (Model II)

The purpose of the PCM simulator was to duplicate the OSO telemetry, command, sun pulse, magnetometer pulse, and power distribution signals. The purpose of the binary-to-decimal converter was to convert the experiment serial data into parallel data format which would be compatible with the Franklin printer.

All instrument handling was accomplished in a controlled-access clean room except when otherwise noted in the applicable handling procedure. Each major phase of instrument testing was certified by the cognizant experimenter to have been performed in accordance with the approved procedures.

3.7.3   Experiment Integration and Test

Experiment integration and test activities were conducted in conjunction with the observatory acceptance program. After the interface-compliance tests mentioned in paragraph 3.7.2, experiments were installed in the spacecraft and further tested as a part of the observatory. Normal experiment accommodation activities continued throughout the observatory acceptance and environmental-test program.

Test procedures, handling procedures, telemetry print routines, and special test equipment continued to be refined throughout the observatory test program. Close liaison between GSFC, BBRC, and experimenters, coordinated by the BBRC and GSFC experiment-accommodation organization, expedited the refinement process. Experiment, spacecraft, and test hardware and documentation was updated as necessary.

3.7.4   Experiment Support Documentation

The program experiment-accommodations personnel were responsible for preparing documentation supporting experiment development, integration and acceptance. The documentation was used to

communicate between the experimenters, GSFC, and BBRC, and aided in identification of areas requiring continuing coordination.

The major support documents for OSO-6 experiments are identified in Table 3-3. Additional documentation was provided through daily, weekly, and monthly reports, and observatory logs and records.

All support documentation required experimenter, GSFC, and BBRC approval.

3.8   OBSERVATORY INTEGRATION AND TEST

This phase of the OSO-6 project covered the period between spacecraft buyoff and observatory buyoff. It included the experiment integration and test activities described in the previous paragraphs and spacecraft catch-up items as required. A graphic history of observatory activities during this phase is presented in Fig. 3-5.

The basic purpose of the observatory integration and test phase was to combine the scientific payload with the OSO spacecraft, iron out incompatibilities, and conduct the integrated acceptance-test program. The acceptance tests included testing under the anticipated environmental conditions of launch and in-orbit operation.

Parameters tested under the required conditions were specified in 26494, Test Specification for the OSO-G Spacecraft, and the experiment operation specifications and procedures shown in Table 3-3.

In the paragraphs which follow, major observatory integration and test activities are discussed.

Table 3-3

OSO-6 EXPERIMENT SUPPORT DOCUMENTATION

Instrument	Experiment Test Report Prototype	Flt I	Flt II	Experiment Test Procedure	Elect. Interface Control Drawing	Mech Interface Control Drawing
Wheel						
UNM	30 Aug 68 TR68-52	7 Jul 69 TR69-32	---	9 Jul 69 TN68-12-A	2 May 69 21832-C	26 May 69 24785-F
UCL	20 Sep 68 TR68-51	7 Jul 69 TR69-31	25 Aug 69 TR69-41	9 Jul 69 TN68-09-B	11 Aug 69 21836-E	26 May 69 24783-E
Bologna	30 Jul 68 TR68-44	7 Jul 69 TR69-30	7 Jul 69 TR69-40	9 Jul 69 TN68-11-A1	10 Jan 69 21838-D	4 Feb 69 24784-C
LASL	16 Aug 68 TR68-46	7 Jul 69 TR69-29	---	9 Jul 69 TN68-10-A1	11 Aug 69 21834-D	26 May 69 24786-D
Rutgers	24 Oct 68 TR68-53	25 Aug 69 TR69-28A	---	9 Jul 69 TN68-13-A	3 Apr 69 21840-D	26 May 69 24787-E
Sail						
NRL	29 Oct 68 TR68-50	7 Jul 69 TR69-27	---	9 Jul 69 TN68-07-B	1 Apr 69 21842-D	26 May 69 24782-F
HCO	14 Jan 69 TR68-49	7 Jul 69 TR69-26	---	9 Jul 69 TN68-08-B	11 Aug 69 30914-B	26 May 69 24782-F

OTHER DOCUMENTS:

Experimenters' Meeting (TN65-307-A:5 Jan 66)
Spacecraft Simulator (TM70-05:3 Jun 70)
Interface Definition Summary (TN66-40-F: 5 May 69)
Experiment Computer Programs (TN68-19:18 Apr 69)
Test Parameters List (TN65-74-A:23 Apr 68)

Duty Cycle Plan (TN68-92:16 Aug 68)
Contamination Tolerance (TN69-07: 11 Feb 69)
Magnetics (TN67-10:15 Mar 67)
Payload Description (TN69-04:16 Jun 69)
Interface Specification (298000:1 Apr 69)

FOLDOUT FRAME 1

FOLDOUT FRAME 2

(Ball logo)

Top branch (left to right):

RETROFITS START — 12/6/68

S/C PAINTING — 1/4-1/7/69

SUBSYSTEM FUNCTIONALS — 1/16/69

PRELIM. RUN S/C COMP. — 2/21/69

SYSTEM INTEG. COMP. — 2/28/69

RF CORONA & PROOF TEST — 3/7/69

NMR 13910 REWORK

REMATED SAIL TO WHEEL — 3/27/69

LEAK RATE AND PROOF TEST — 4/7/69

SYS. INTEGRATED COMP. — 4/8/69

EXPERIMENT INTEGRATION — 4/9-5/12/69

SUN POINTING & J-BOX RECAL — 4/18-5/1/69

OBS SYSTEMS INTEG TEST — 5/14/69

OBS. ACCEPT. TESTS — 5/16-5/22/69

CONT.

Sub-boxes:

ROLL COIL MOD — 12/7-3/6/68
EMISSIVITY MEAS. — 1/8/69
ENGR NOISE TESTS — 12/2-1/20/69
BATTERY CONDITIONING — 2/20-2/24/69
PREPS FOR TV — 3/3-3/6/69
RING OUT ALL SAIL WIRING — 3/11-3/19/69
INSTALL ALL SAIL SUBASSY — 4/1-4/3/69
VERIFIED NUT. DAMPER RELEASE — 4/9/69
PRE MOD FILTER TESTS — 4/25-4/26/69
EXP XTALK TESTS — 5/15/69

IN SEQ MOD — 12/6-12/22/68
TV PAINT BAKE OUT — 1/17/69
OFFSET SYST EVALUATION — 1/27-1/3/69
FLIGHT TEST ALL SQUIBS — 2/25-2/26/69
RF SUBSYSTEM TEST — 3/3/69
STRUCTURAL & MECH INSPECT — 3/11-3/18/69
SAIL SUBSYS FUNCTIONALS
SAIL NOISE MEAS — 4/14/69
VSWR MEAS — 4/28/69
EXP COMPS — 5/16-5/22

TAPE RECORD TEMP PROBE — 12/6/68
SUN POINTING (NEW GAINS) — 2/1-2/7/69
RF SUBSYSTEM TEST (S/C) — 3/3/69
WHEEL MECH. BALANCE — 3/27/69
SUN POINTING — 4/5/69
FINE CONTROL ALIGN — 5/13/69
PNEUMATIC FLOW TEST — 5/21/69

Notes (left column):

REMOVED 09TBI FOR TAPE RECORDER MOD (OUTPUT NETWORK)
WHEEL TB CONFORMANCE COATING
REMOVED 04TBI FOR ADDITION OF FUSES
REMOVED SERVO BOX FOR GAIN CHANGE
ADDED SAIL STRUCTURE TEMP. PROBE
INSTALLED LONG SLOPE EYES
INSTALLED BOLOGNA FLIGHT 2 EXP (2/10/69)
INSTALLED FLIGHT SQUIB BATTERIES
INSTALLED FLIGHT LASL EXP (2/18/69)
REMOVED SAIL J-BOX TERM BDS FOR MOD TO MF32
INSTALL FLIGHT RCVS & DECODERS (2/22/69)
INSTALLED ROLL COILS (3/6/69)

REMOVED ALL SAIL SUBASSEMBLIES FOR EVALUATION TESTS PER NMR
REMOVED SOLAR ARRAY AND REPLACED IN S/N 4
REMOVED SAIL TO REPLACE AZIMUTH SHAFT
REPLACED AZIMUTH MOTOR AND PITCH BOTTLE
REMOVED BOLOGNA AND LASL EXPERIMENTS FOR BENCH TESTS
RETERMINATED DWM SHIELD TO PWR GND
RETERMINATED AZ AND EL MOTOR TO PWR GND
REMOVED 03+ BI AND 06TBI FOR RUTGERS SQUIB MOD
INSTALLED FLIGHT TAPE RECORDERS (4/3/69)
REMOVED XMTR I FOR PRE MOD FILTER

INSTALLED LASL EXP (4/9/69)
INSTALLED BOLOGNA EXP (4/9/69)
INSTALLED NRL EXP (4/10/69)
INSTALLED FLIGHT SPARE J-BOX S/N 3 (4/17/69)
INSTALLED PRE MOD FILTER & XMTR NO.1 (S/N 15)
INSTALLED UNM EXP (4/24/69)
REMOVED J-BOX TERM BDS 1 & 3 FOR HCO MOD
INSTALLED UCL EXP FOR FLIGHT (5/8/69)
INSTALLED RUTGERS EXP FOR FLIGHT (5/8/69)
INSTALLED HCO EXP FOR FLIGHT (5/9/69)

Bottom (problems):

SPACECRAFT PROBLEMS
1. BARE WIRE ON 08TBI
2. BARE WIRE ON 04TBI
3. 19V CONNECTED TO REG BUS

1. PWM NOISE ON OFFSET GROUND BUS

1. CHAMBER BACKFILL CAUSED EXCESSIVE SPIN RATES

1. TIMER RUN RELAY FAILURE ON 06TBI
2. DAMAGED CABLING IN COMPT 8

1. EXP MOUNTING HARDWARE INTERFERES W/TERM BD RELAYS
2. UCL SENSITIVE TO NOISE ON SIG GND (LEA 8281)
3. UNM MTG HOLES DO NOT INFACE W/ WHEEL (NMR 13423)

1. EL CONTROL SENSOR REVERSAL THRU HCO (5/13/69)
2. HI SENSITIVITY ON UNM EXPERIMENT (5/16/69)

Fig. 3-5   OSO-6 (OSO-G) Integration and Test History (Sheet 1 of 2)

3-24

3.24.1

FOLDOUT FRAME

FOLDOUT FRAME

REVERSED ELEVATION CORNER SENSOR POLARITY IN SAIL J-BOX
REMOVED UNM EXPERIMENT TO CHANGE POLARITY
FILLED ARM DAMPER SYST. W/ALCOHOL

SECURED ALL COMPARTMENTS FOR VIBRATION
INSTALLED FLIGHT NOZZLES
PRESSURIZED SPIN & PITCH PNEUMATICS TO 1500 PSI
INSTALLED NRL SPECTROMETER

PRESSURIZED PNEUMATICS TO 1500 PSI FOR T.V.
REPLACE ARM RELEASE SQUIBS
REMOVE NUT DAMPER TO CLEAN SQUIB BLOCK, RECAGED AND INSTALLED NEW PRESSURE CARTRIDGES

BREAK VACUUM TO:
(1) CHANGE FLEX CABLES
(2) REPLACE L.A.S.L. PREAMPS
(3) REPAIR RUTGERS FILTER WHEEL

REMOVE 06TBI TO REPLACE LAUNCH SEQ. RESISTOR
REMOVE 09TBI TO ADD SUPPRESSION DIODES ON RUTGERS POWER LINE

REMOVED NUT DAMPER CHANGE ELEVATION LOCK

ROUGH MASS BALANCE — 5/23/69
- STATIC & DYN BALANCE OF POINTEDS — 5/21/69
- PREPS FOR VIBRATION

VIBRATION — 5/24-5/25/69
- IN & LIFTOFF READY OVERLAYS
- Z AXIS SINE & RANDOM VIBRATION
  - OVERLAY
- Y AXIS RESONANT SEARCH & SINE VIBRATION
  - MECH. INSPECTION OVERLAYS T.R. NO.2 P.B.
- Y AXIS HALF PWR & FULL PWR RANDOM VIBRATION

VIBRATION — 5/25/69
- X AXIS RESONANT SEARCH & SINE VIBRATION
  - OVERLAYS T.R. 2 P.B.
- X AXIS HALF POWER RANDOM VIBRATION
  - OVERLAYS
- X AXIS FULL PWR

SHOCK TEST — 5/25/69
- LAUNCH SEQ. TIMER RUN
- ARMS UP/ARMS SLACK TEST

POST VIB OBS TESTS — 5/25-5/27/69
- SUN POINTING — 5/26/69
- EXPERIMENT COMDS — 5/27/69
- BATTERY CONDITIONING — 5/29-6/4

SYSTEMS INTEGRATED — 5/28/69

CG, MOI & WT MEAS. — 6/2-6/4/69
- SAIL STATIC BALANCE

OBS THERMAL VACUUM — 6/4/69
- LAUNCH SEQ. & CORONA TEST

RESTART T.V. (CORONA & LAUNCH SEQ.) — 6/8-6/19/69
- ORBIT 201-253 (HOT) SPECIAL HCO TEST
- ORBIT 254-258 TRANSITION / 259-266 15°C / 267-275 TRANSITION
- ORBIT 276-298 COLD / 299-307 TRANSITION / 308-345 HOT
- ORBIT 346-354 TRANSITION / 355-385 COLD

CONTINUE T.V. NEW FLEX CABLE — 6/19-6/24/69
- ORBIT 386-388 HOT / 389-398 TRANSITION
- ORBIT 399-433 COLD / 434-440 TRANSITION
- COMPLETE VACUUM MECH. INSPECTION — 6/24/69

PROBLEM INVESTIGATION — 6/25-6/27
- EXP. CHANGE-OUT & COMPS — 6/26
- SAIL ASC & DSM NOISE TESTS — 6/25-6/29/69

FINAL PREPARATIONS AND LAUNCH

SPACECRAFT PROBLEMS

1. LAUNCH SEQUENCE TIMER STARTED DURING T.V. CONSOLE CHECKOUT (NMR 13918)
2. NUT DAMPER LATCH DID NOT FULLY UNCAGE (NMR 15374)

1. NO OUTPUT ON CHANNELS 4 & 5 OF L.A.S.L. EXP.
2. SUN/SHADE LOGIC FAILURE ON L.A.S.L.
3. FLEX CABLES NON FLIGHT
4. RUTGERS FILTER WHEEL NO.2 JAMMED
5. SAIL ASC SYNC LOSS W/SAIL DAY ENABLE CMD.
6. DSM F/C SLIP W/UNM & RU

1. RUTGERS FILTER WHEEL NO.2 JAMMED
2. BOLOGNA H.V. FAILURE
3. L.A.S.L. CHANNEL 4 FAILURE

1. LAUNCH SEQUENCE RESISTOR MAY NOT BE RELIABLE

Fig. 3-5 OSO-6 (OSO-G) Integration and Test History (Sheet 2 of 2)

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3.8.1 General Procedures

The procedures described in this paragraph are those used repeatedly throughout OSO-6 integration and test activity. They were applied as shown in Fig. 3-5 for acceptance and survival testing, and for maintenance and troubleshooting.

Spacecraft Comprehensive Test (TN68-14). The spacecraft comprehensive test was a combination of test sequences, largely drawn from the supplementary tests, which collected a large amount of test information in a small time. All systems were tested to some extent by the comprehensive. The test was performed in the clean room under controlled atmospheric conditions.

The spacecraft comprehensive test was executed with the help of the CDC-160A computer system, using the special BBRC-developed Observatory Automatic Test System (OATS) program. OATS provided the following advantages in testing:

(1) More efficient use of test time

(2) Magnetic tape records of test data were available for subsequent data processing and history

(3) Reproduction of review and/or data copies of procedures was greatly simplified

(4) Test accuracy was increased, adding to the safety of the items tested

(5) Levels of testing which had previously been extremely difficult or impossible were made simple

(6) An editing routine was used for efficient
     updating of procedures

(7) Greater test repeatability was achieved, and

(8) Records of mistakes and unscheduled operations
     were automatically inserted with the test data.

Spacecraft Supplementary Tests (TN68-05). The tests in this
category were each formed to provide a thorough testing of related
spacecraft parameters. In the aggregate, the supplementary
tests cover all spacecraft test specifications (except environ-
mental). Some engineering and performance tests not specified
were also included.

Supplementary tests were adapted for the OATS technique whenever
such adaptation offered a significant net advantage.

A detailed mechanical-inspection procedure was included in the
supplementary procedures. It was used to verify mechanical
integrity of the observatory after each environmental-test
sequence.

A supplementary procedure for conditioning the power and squib
batteries was also prepared and used throughout the integration
and test phase. The conditioning procedure consisted of fully
discharging and recharging the batteries.

Experiment Tests. Experiment tests were published in the operation,
specification, and procedures documents shown on Table 3-3. The
tests were executed using special computer print routines which
arranged and labelled the telemetered experiment data for easy
interpretation.

3.8.2   Thermal-Vacuum (T-V) Test (TN68-21)

The purpose of the T-V test was to provide for evaluation of OSO's performance under conditions of temperature and pressure similar to those anticipated for flight. The test was conducted with OSO installed in the large, 10-foot thermal-vacuum chamber shown in Fig. 3-6.

Control of the observatory spin rate during T-V test was exercised through the T-V test console. This console also controlled battery charging, in-chamber stimulus lamps, and observatory switching functions, and provided monitoring of selected status points. Control of the observatory was also maintained through OSO's command system.
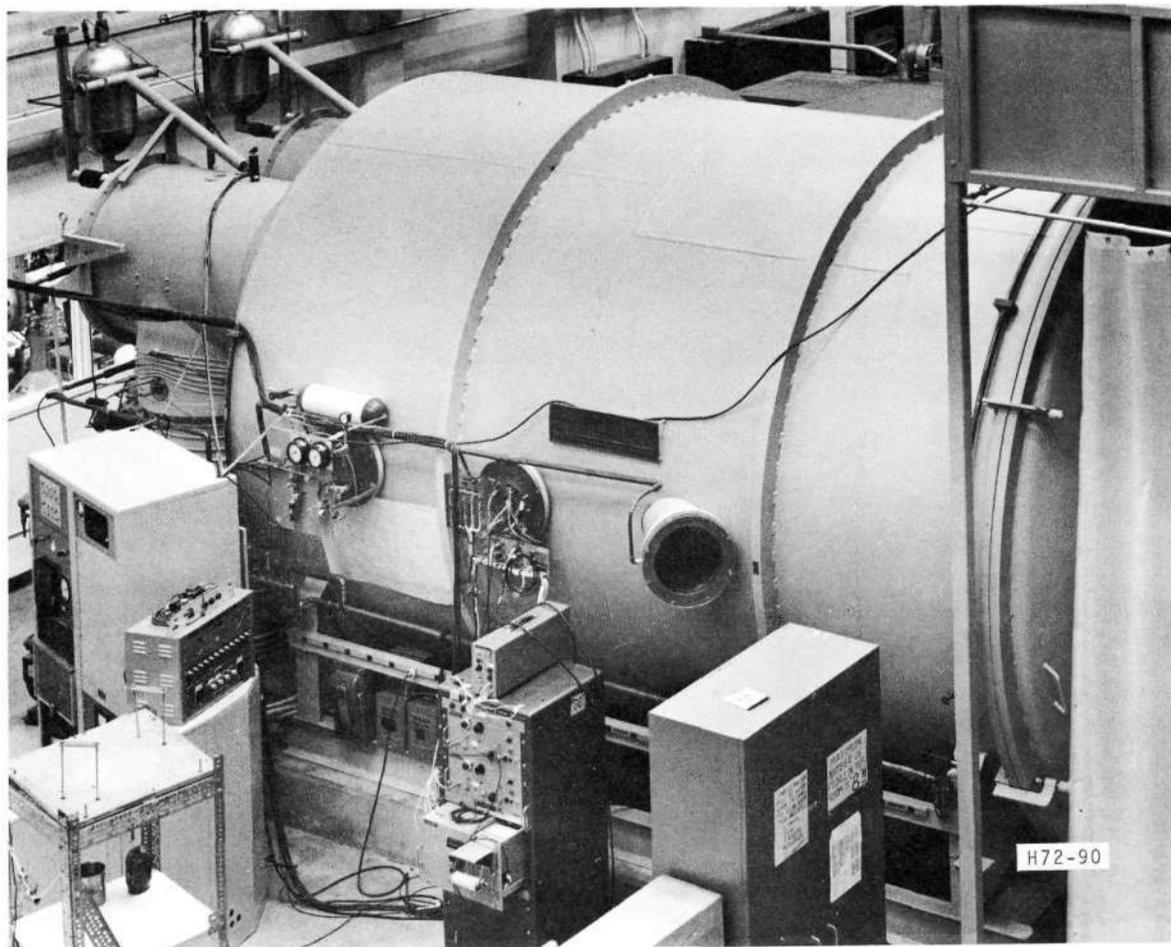
Fig. 3-6   Thermal-Vacuum Chamber

The OSO-6 thermal-vacuum test consisted of 15 days of observatory operation in vacuum ($<10^{-5}$ torr) as follows:

(1) A full exercise of the launch-sequence control system under simulated flight conditions

(2) Operation for four days at +25°C (hot exposure, corona, outgas)

(3) Transition to 0°C and operation at that temperature for one day (cold exposure)

(4) Transition to +25°C and operation at that temperature for three days (hot soak)

(5) Transition to 0°C and operation at that temperature for three days (cold soak)

(6) Transition to +25°C for test-chamber bleedup.

The launch sequence was started by simulating the actuation of the separation switches. This started the launch-sequence timers.

Experiments were not turned on during the first three days of testing, to allow time for them to outgas thoroughly. This provided protection against corona-discharging from their high-voltage circuits.

Spacecraft tests comparable to the comprehensive test (Para. 3.8.1) were conducted periodically throughout the test at selected conditions of temperature and voltage. Experiment tests were conducted as described in the experiment operation, specifications, and procedures manuals (Table 3-3). Tape recorders were played back regularly throughout the test and thoroughly evaluated several times during each test period. A leak check of pneumatic systems

was conducted at the beginning and end of each temperature phase of testing. Voltage-extreme and special tests were included as required by specifications or by the test engineer.

During T-V testing the observatory was operated in a 90-minute day/night duty cycle. This was accomplished by turning spin-eye stimulus lamps on and off by timer control. Telemetry data points were sampled and verified at regular intervals. The flight output of the solar-cell array was simulated by applying battery charge power through a timer and an impedance network.

Spacecraft solar sensors and experiment solar and cosmic ray detectors were stimulated by sources placed in and around the test chamber. The spacecraft communications systems tests and the experiments tests were executed using the 160A computer system.

Contamination Monitoring in T-V. During the observatory thermal-vacuum tests, a series of measurements were made to determine the outgassing and deposition levels of potential contaminants evolving from the observatory.

A mass spectrometer/residual-gas analyzer was used to monitor gases evolved during the tests. Materials that condensed on optical-quality mirrors and refrigerated collectors in the test chamber were analyzed after the test. Analysis of residual gas analyzer data showed that the replacement of outgassing materials discovered in tests of earlier OSO's reduced outgassing to insignificance on OSO-6.

3.8.3   Vibration Test (TN68-20)

The purpose of the vibration test was to verify that the structural members of OSO-6 and its payload were strong enough to survive the

vibration anticipated during launch. Electrical circuits on board OSO-6 were energized in the launch mode to demonstrate that these circuits were not vulnerable to vibration. Comprehensive and supplementary tests (Para. 3.8.1) before and after vibration demonstrated that all electrical circuits and components could survive the launch.

The levels to which OSO-6 was vibrated, predetermined by the characteristic vibration of the launch vehicle, are shown in Table 3-4. The axes of vibration defined for vibration testing are shown in Fig. 3-7. Figures 3-8 through 3-10 show OSO-6 mounted on the driver for each of the three axes of vibration.

Vibration Accelerometers. Before each vibration sequence, accelerometers were mounted to the observatory. The outputs of each of these accelerometers was recorded during the vibration. The accelerometers were classified as control, monitor, crosstalk, or response, commensurate with their use.

The control accelerometers were located immediately below the separation plane on the vibration-fixture adapter. Their sensitive axes were aligned parallel to the axis of vibration. Their outputs were fed to the vibration equipment for amplitude control and to recorders for documentation of control-axis vibration.

The monitor accelerometers (placed adjacent to the control accelerometers) allow real-time visual evaluation of the controlled input. The crosstalk accelerometers were aligned perpendicular to the axis of vibration to record any off-axis motion at the control plane. Response accelerometers were those used as required to study the observatory in vibration.

Ball

Table 3-4

OSO-6 VIBRATION ACCEPTANCE TEST LEVELS

Axis	Frequency (Hz)	Acceleration (g, 0 to peak)	Sweep Rate (octaves per min)
Thrust Z (Sinusoidal)	5-11 11-17 17-23 23-200	0.33 in. D.A.* 2.0 4.0 1.5	4 4 3 4
Transverse X (Sinusoidal)	5-9.5 9.5-11 11-20.5 (1) 20.5-31 31-37 37-200 (1)	0.33 in. D.A.* 1.5 0.15 1.0 0.37 1.0	4 ↓
Transverse Y (Sinusoidal)	5-9.0 9.0-17.5 (2) 17.5-27.0 27.0-32.5 (2) 32.5-200	0.33 in. D.A.* 0.20 1.0 0.48 1.0	↓

NOTES:

  (1) Resonance occurs within this range in the X-axis.
  (2) Resonance occurs within this range in the Y-axis.
  *   Double Amplitude (peak-to-peak)

Axes	Frequency (Hz)	PSD Level (G^2/Hz)	Acceleration (G-rms)	Duration (min)
Z, X, and Y (Random)	20-300 300-2000	0.0017 to 0.020 increas- ing 3 db/ octave 0.020	6.1 ↓	2 each axis
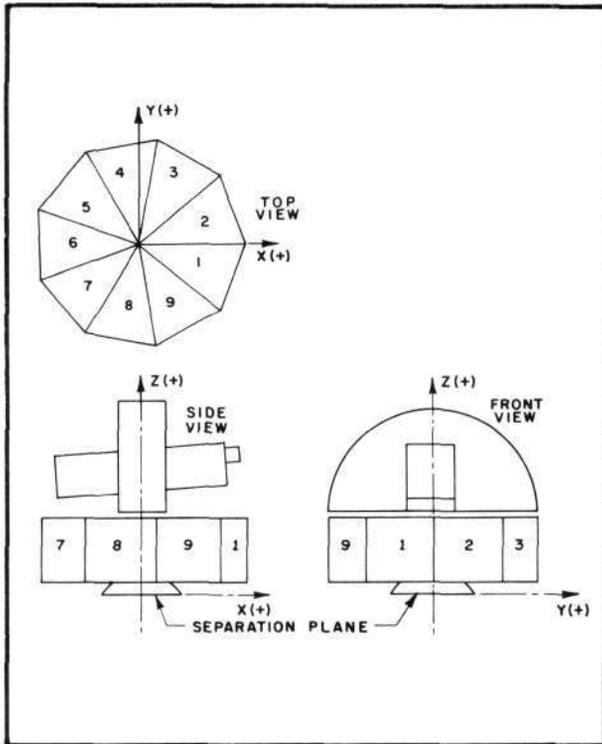
Fig. 3-7 OSO Axes of Vibration
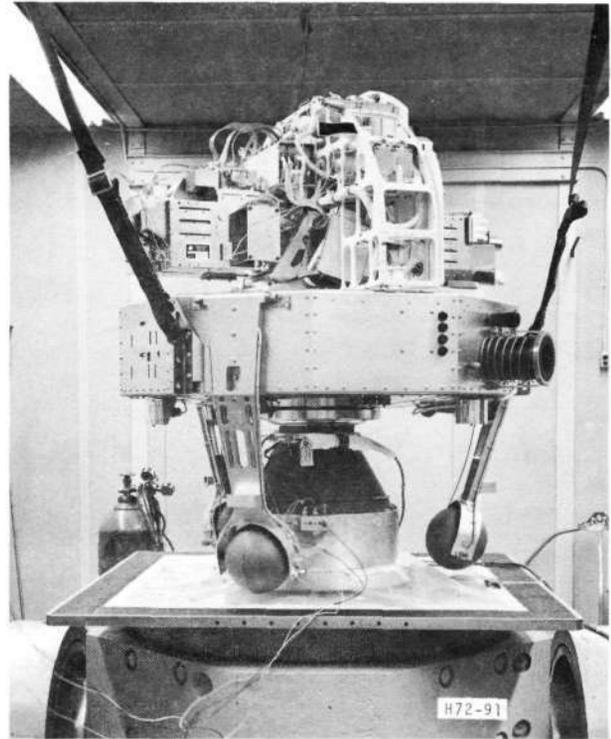
Fig. 3-8 OSO-6 Z-Axis
          Vibration
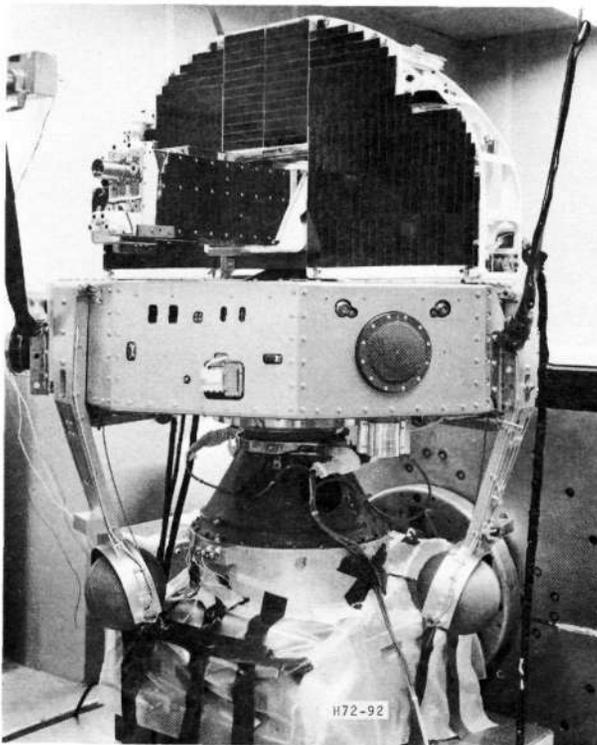
Fig. 3-9 OSO-6 X-Axis
          Vibration
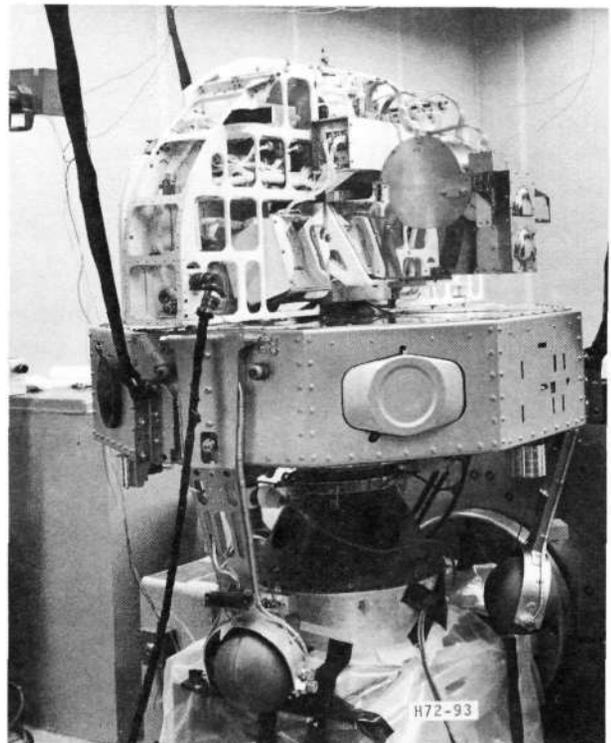
Fig. 3-10 OSO-6 Y-Axis
           Vibration

Resonance Search. To avoid exceeding the Delta-N vehicle launch loads on the OSO structure, the driver signal was attenuated, or notched, at the frequencies of OSO's mechanical resonances. These frequencies were identified before the transverse-axis vibration runs by conducting resonance searches at about one-half the calculated notch levels.

Monitor during Vibration. During bursts of vibration, the following records were kept:

(1) Magnetic-tape records of accelerometer outputs (from which both transmissibility plots (db plots) and filtered g-level plots were subsequently made)

(2) Telemetry-printout of spacecraft housekeeping data, and

(3) Magnetic-tape record of observatory DME output.

In addition to those records listed above, charts and photographs were prepared to document accelerometer locations.

3.8.4   Dynamic Balance

For proper performance during launch and in orbit, the OSO-6, like other OSO's, had to be balanced statically and dynamically. The launch-condition balance was specified by MDAC to be compatible with the Delta vehicle dynamic characteristics. The in-orbit balance was specified to limit the effects of wobble on spacecraft systems and experiments performance.

Approximate balance is required to prevent excessive wobble during thermal-vacuum and sun-pointing tests. Since removal of equipment from the observatory and its subsequent reinstallation upsets the balance, OSO-6 was balanced several times as indicated in Fig. 3-5.

The balancing machine used on OSO-6 is shown in Fig. 3-11. Each balancing operation required several repetitions of the basic balance procedure, until balance was within the required limits. After the balance machine was erected and aligned, the observatory was mounted to the spin table. The spin axis of the observatory was aligned with the bearing axis of the spin table.

The wheel, with arms extended, was balanced first with the sail tied to the machine frame to prevent rotation. The sail was then tied to the wheel and the balancing process was repeated. The PIA center of gravity was made coincident with the elevation axis by weight additions. The pointed instruments were then allowed to settle at their natural spinning angle during sail balance, and the elevation lock was subsequently adjusted to hold them at that position. The observatory arms were then lowered and the balance was checked in the launch configuration. Balancing was accomplished by adding weights to the spacecraft as needed.

3.8.5   Mass-Properties Measurements

The OSO moment-of-inertia, center-of-gravity, and weight measurements were needed to determine booster performance requirements and the number of spin rockets needed on the second-stage of the launch vehicle. These parameters were also needed for in-orbit performance analysis, the results of which are used to improve future OSO designs.

Moment-of-Inertia (MOI). Two types of MOI measurements were made on OSO-6: spin-axis MOI and transverse-axis MOI.

The spin-axis MOI was measured suspending OSO as shown in Fig. 3-12 and causing it to oscillate as a torsional pendulum. The

oscillation frequency was then measured with the arms both extended and stowed, and with the sail locked and unlocked. MOI for the different configurations was calculated using the formula:

$$I_{spin} = \frac{Wr^2P^2}{K4\pi^2L} \quad (slug\text{-}feet^2)$$

where:

> K = 0.995 (amplitude correction factor)
> W = weight of OSO (1b)
> r = distance from spin axis to cable attach points (ft)
> P = period of oscillation (sec)
> L = length of cables (ft)

Corrections were made for the lift lug and for the spin gas added for launch.

The transverse-axis MOI was calculated using data obtained with OSO mounted on the transverse MOI fixture (Fig. 3-13). OSO and the pivoted yoke formed a compound pendulum of which the period, mass, and length were measured. The following equation was developed to remove the effects of the pivoted yoke and includes several measured constants necessary to accurately determine OSO transverse MOI:

$$\begin{aligned} I_{transverse} = {} & 6.5 \times 10^{-4}T^2 (L + 10) - 6.46 \times 10^{-12}T^2W^2 \\ & (L^2 + 11L + 30.5) - 2.22 \times 10^{-2}L^2 - 0.206L \\ & -1.87 \end{aligned}$$

where:

> T = time for 10 oscillations (sec)
> L = distance from yoke axis to system CG (in.)
> W = weight of OSO (1b)
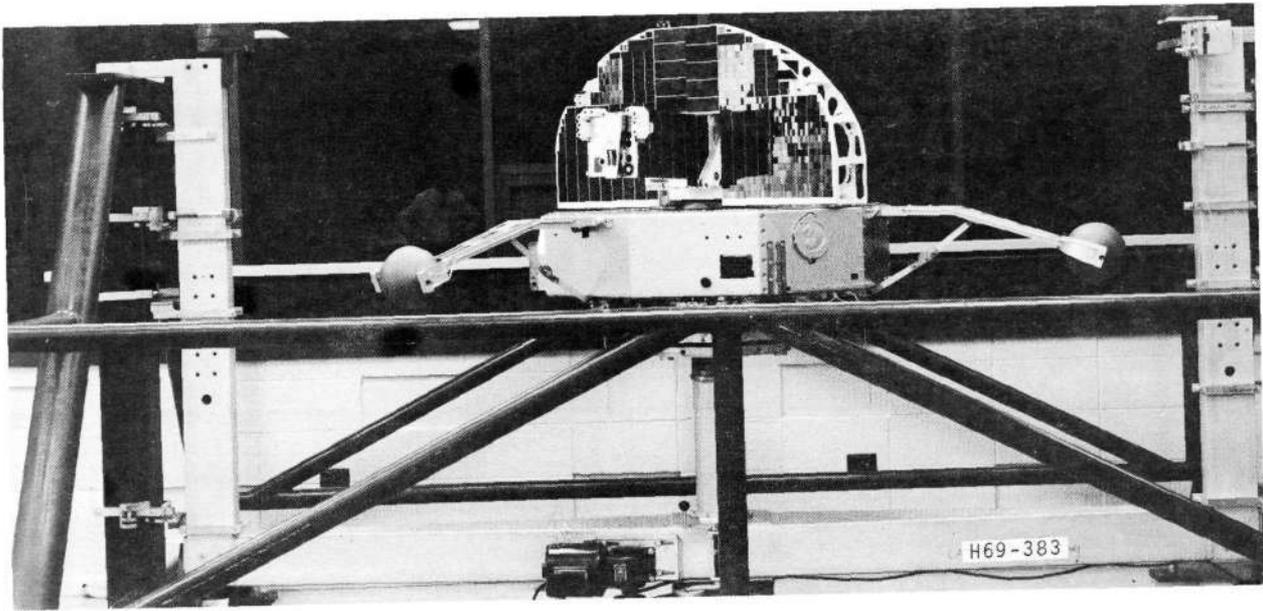
Fig. 3-11   OSO-2 on Balance Machine
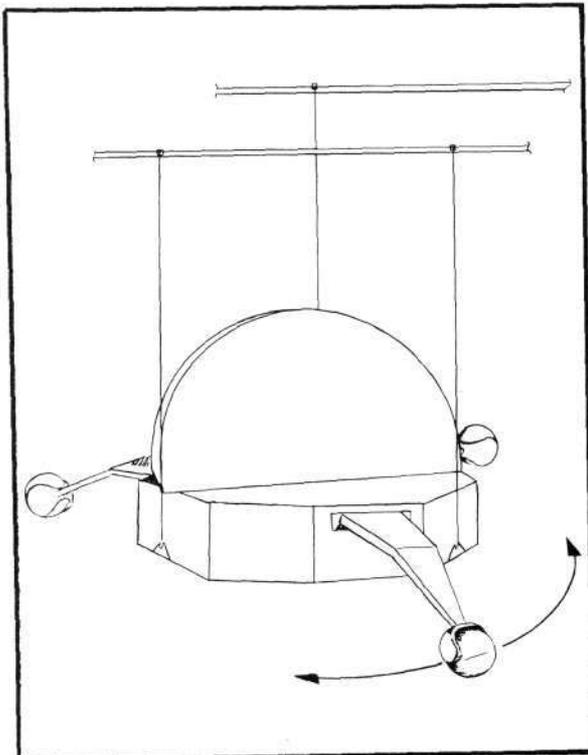
Fig. 3-12   Spin MOI Test Setup
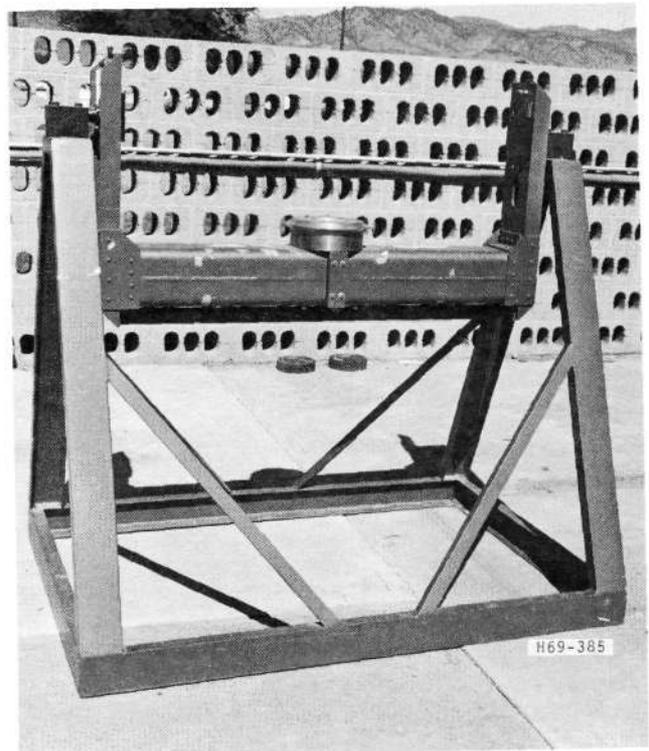
Fig. 3-13   CG/MOI Test Fixture

The maximum transverse MOI about the pitch and roll axes was then calculated using these data, and is shown in Table 3-5. Corrections were made for the addition of spin gas before launch.

Table 3-5

OSO-6 WEIGHT, CG, AND MOI

Item	Measured or Calculated Values
Weight	639.91 lb.
CG (distance from separation plane)	12.42 inches (arms down) 12.97 inches (arms up)
Spin-Axis MOI in Launch Condition	22.1 slug-ft^2 (wheel only) 27.5 slug-ft^2 (wheel & sail)
Spin-Axis MOI in Orbit Condition	29.9 slug-ft^2 (wheel only) 35.3 slug-ft^2 (wheel & sail)
Maximum Transverse MOI	24.8 slug-ft^2 (Launch condition) 25.0 slug-ft^2 (Orbit condition)
Ratio of Spin MOI to Max Transverse MOI in Orbit Condition	1.20 (daytime) 1.41 (nighttime)

Center-of-Gravity (CG). The transverse MOI test fixture (Fig. 3-13) was also used to locate the OSO-6 CG. The yoke was adjusted up and down to the position where OSO would remain at any angular orientation without being reoriented by the pull of gravity. That position defined the CG of the OSO/yoke combination.

This procedure was performed with OSO in both the launch and orbit configurations. The CG location was then calculated for each configuration using the formula:

$$CG = dp + \frac{My}{Ms} (dy + dp)$$

where:

        CG = OSO CG location in inches above the separation plane

        dp = OSO/yoke CG location in inches above the separation plane

        My = mass of yoke = 2.23 slug

        Ms = mass of OSO in slugs

        dy = yoke CG location in inches below the separation plane = 0.6227 in.

The center-of-gravity locations for OSO-6 are listed in Table 3-5.

Weight Measurement. The OSO weight measurements were made on Hydraset scales (Fig. 3-14). The weight of handling equipment was subtracted from the indicated weight and the weight of nitrogen gas to be added for launch was added to arrive at OSO-6's flight weight. This weight is indicated in Table 3-5.

3.8.6   Magnetic Balance

When OSO is in orbit, its dynamic behavior (including pitch and roll attitude drift) is governed in part by the interaction of observatory magnetic dipoles with the geomagnetic field. Proper adjustment of the OSO-6 magnetic dipoles was used to increase its in-orbit life by reducing attitude gas consumption. From a magnetic control study, it was concluded that the OSO-6 dipole moments should be adjusted as shown in the first column of Table 3-6.
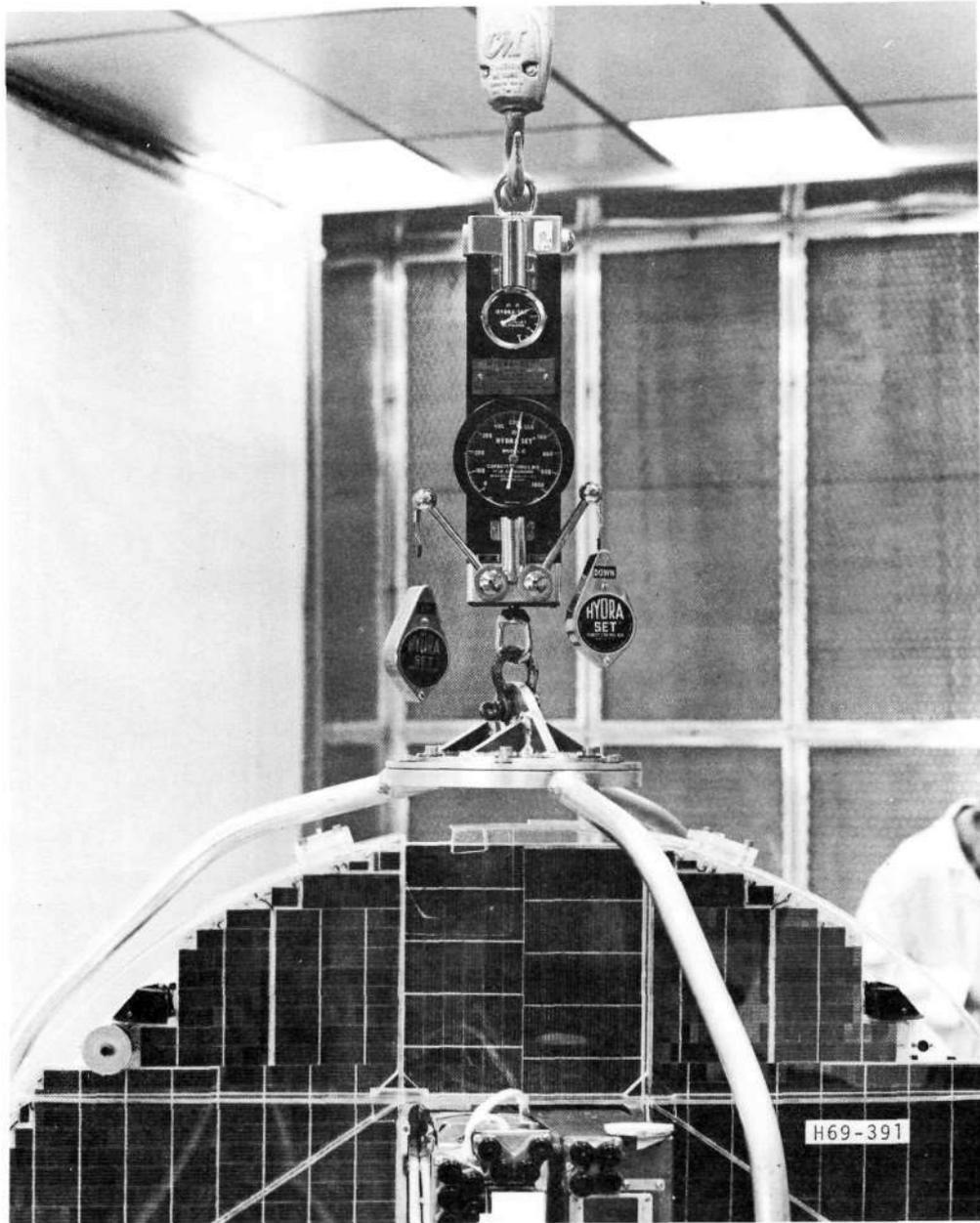
Fig. 3-14   OSO on Weighing Scale

Table 3-6

OSO-6 MAGNETIC DIPOLE MOMENTS *

OSO Axis	Specification	After Demag./ Before Balance	Final Value
X	-200 + 200	-84	-84
Y	0 + 200	+936	+26
Z	-2200 + 200	+792	-2133

* All moments shown are in dyne-cm/gauss

The magnetic balancing operation uses two major pieces of equipment; the observatory demagnetizer and the observatory magnetotropometer. Both these, together with their auxiliary apparatus, are used in the BBRC magnetotropometer building. This building is constructed of extremely-low-magnetic materials in a location far from things which could disturb the local geomagnetic field.

Magnetic balance was performed in four steps:

    (1)   OSO was demagnetized to eliminate easily-variable accidental magnets.

    (2)   Temporary test magnets were attached, which had dipole moment components that were the negative of the desired values.

    (3)   The net magnetic dipole was measured and compensating magnets were permanently installed, making the net moment zero. The dipole moment was then remeasured, to verify that the net moment was near zero.

    (4)   The temporary test magnets were removed, thereby bringing the net moment close to the target value.

Condition of OSO for Magnetic Balance. Before magnetic balance was started, OSO was prepared as follows:

- Observatory batteries were fully charged.

- The "launch-turnon plug" was installed so that the pitch coil could be operated by RF commands.

- As much OSO electrical equipment as possible was turned off to reduce the drain on the batteries.

- The pointed instruments were locked with their optical axes perpendicular to the spin axis.

- The OSO arms were in the launch position.

- The sail was wedged to the wheel with the pointed instruments aimed over Compartment 1.

- The solar-array cover was removed.

- All ferromagnetic protective covers were removed.

- Magnetic-equivalent dummy plugs and caps were installed.

- OSO was covered with a polyethylene bag and purged with dry nitrogen.

Demagnetization. Before meaningful dipole moment measurements could be made, the easily-changed parts of the moment had to be removed. This was done using a slowly-decaying alternating field applied by the observatory demagnetizer (Fig. 3-15). The field was not strong enough to affect permanent magnets that were a necessary part of OSO. The field was strong enough, however, to partly demagnetize the accidental magnets so that their value was stabilized.

After the demagnetization was complete, the temporary test magnets were attached, as described above. The test magnets were used so that the final dipole moments measured would be near zero. The magnetotropometer readings are more accurate near to zero than for larger moments.
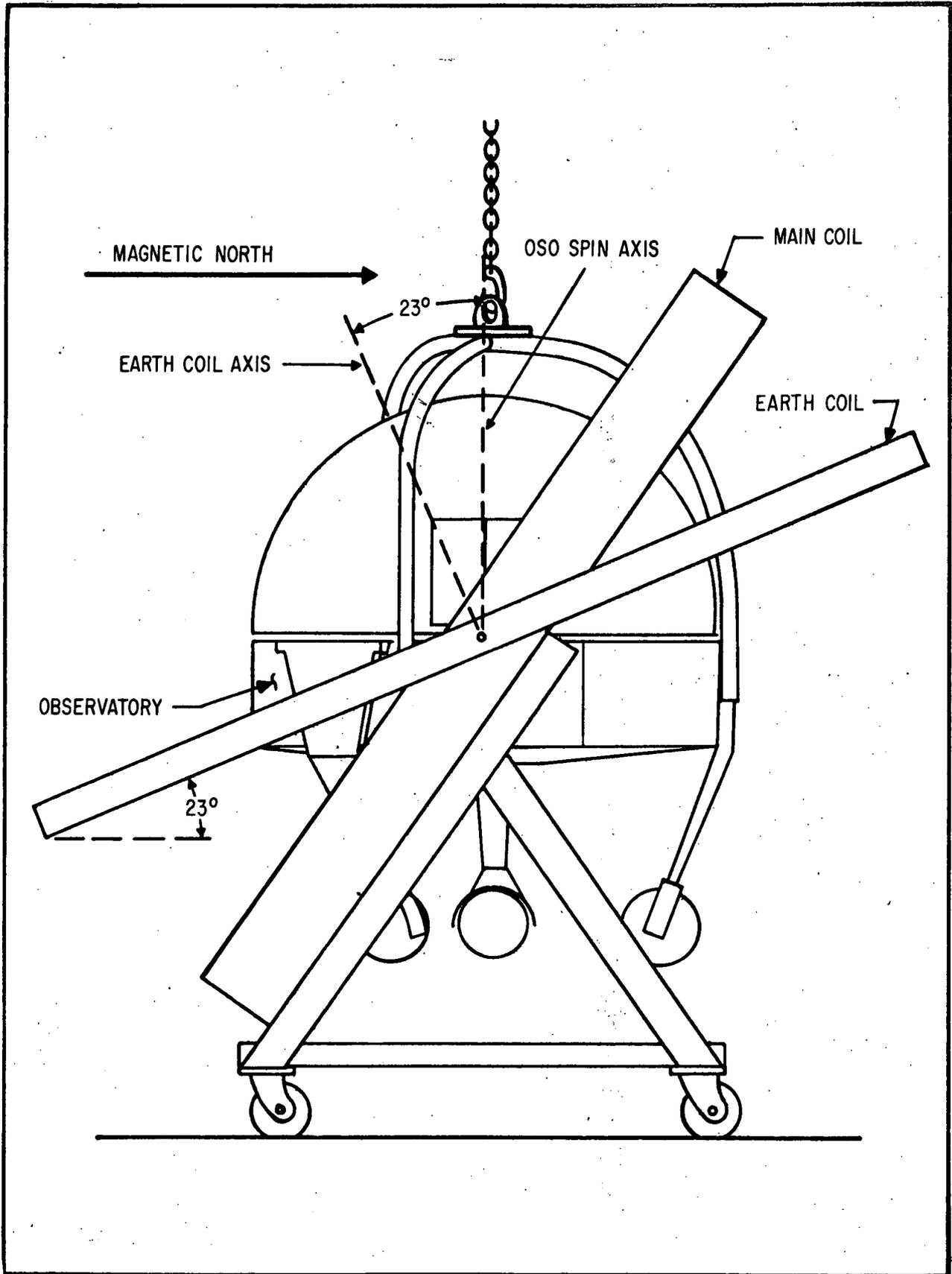
Fig. 3-15 Observatory Demagnetizer

Magnetic Dipole Compensation. The dipole moment was measured by measuring OSO's tendency to act like a compass in the geomagnetic field. The measurement was accomplished on the observatory magnetotropometer (Fig. 3-16).

OSO was suspended, as shown, from a frictionless support, in a region of uniform geomagnetic field. The torque produced by OSO in attempting to align its dipole with the field was then measured. The three components of the dipole were measured by mounting OSO in three mutually-perpendicular orientations on the magnetotropometer.

Permanent compensating magnets were then prepared for each of the three components of the dipole. These were mounted permanently to the sail structure. The observatory dipole moment was then remeasured and verified to be near zero. Removal of the test magnets, then, brought the dipole moment to the desired value, as shown in Table 3-6.

3.9   LAUNCH ACTIVITIES

This phase of the OSO-6 project included shipment to the Eastern Test Range (ETR) at Kennedy Space Center (KSC), Florida, and subsequent activities leading to the launch on 9 August 1969.

3.9.1   Shipment to ETR

OSO-6 was shipped from BBRC, Boulder, to ETR in the OSO shipping container shown in Fig. 3-17 on 17 July 1969. Shipment was via North American Van Lines air-ride truck to Buckley ANG Base, Colorado. Ground-support equipment was packed in 8 by 6 by 6 shipping vaults and transported to Buckley on flat-bed trucks. At Buckley the shipment was loaded into a cargo plane for transfer to ETR.
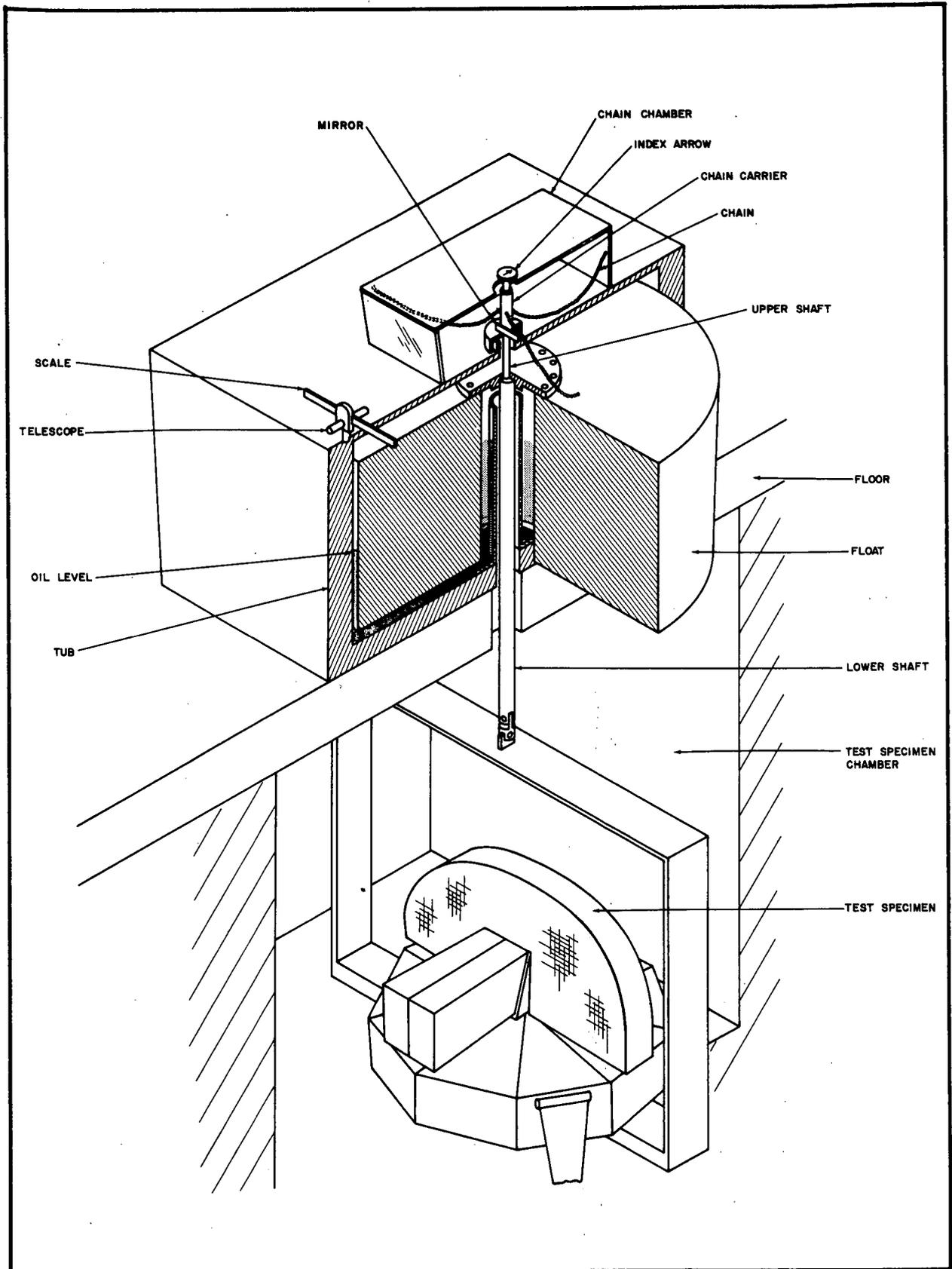
Fig. 3-16   OSO on Magnetotropometer
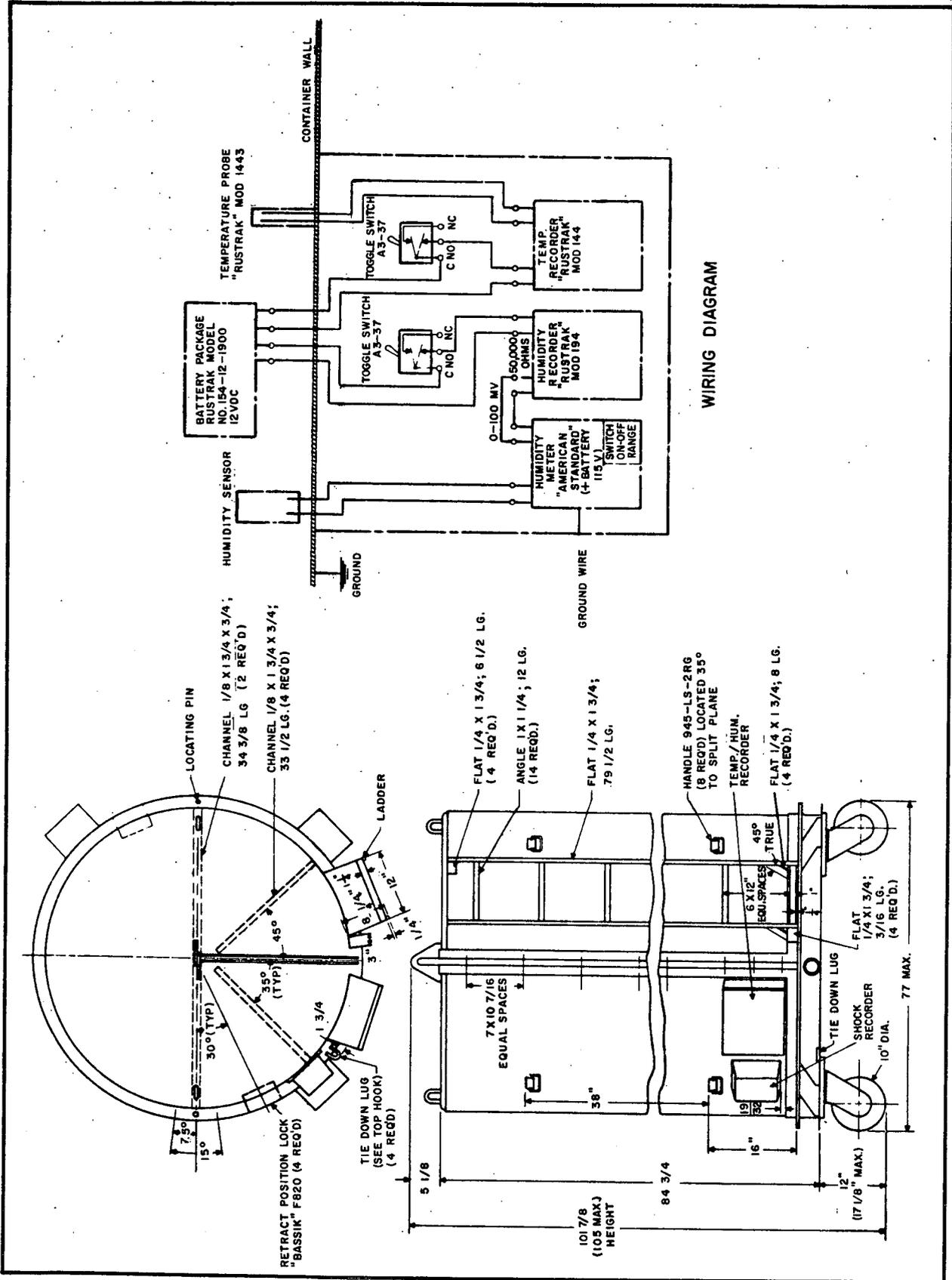
Fig. 3-17 OSO Shipping Container

Before shipment OSO-6 was given a thorough mechanical inspection and appropriate protective covers were installed. The observatory protective capsule, shown in Fig. 3-18, was placed around the observatory before it was installed in the shipping container.

Environmental Controls and Monitors. The OSO shipping container was equipped with eight maximum-reading accelerometers. Four were placed with their sensitive axes vertical, and four horizontal. Each set of four consisted of accelerometers set to trip at 3, 5, 10, and 15g.

The observatory was mounted on three rubber-cushioned shock mounts to isolate it from the canister. A continuously recording accelerometer with a 0-6g range was attached directly to the observatory.

A positive pressure of about 2 psi was maintained inside the canister during shipment. The can was thoroughly purged with dry nitrogen before being sealed and pressurized. A portable nitrogen system was used to maintain the pressure in the can. Seven bags of desiccant (MIL-D03664B) were included as an additional safeguard against moisture.

3.9.2   Prelaunch Activities at ETR

The major items of observatory prelaunch activity at ETR are shown in Table 3-7. Appropriate environmental controls were maintained throughout all these activities. Observatory tests were performed in the assigned hangar clean room whenever possible.

PURGE PORT

LIFTING EYE (REMOVABLE FOR ACCESS TO ALIGNMENT NUBBIN)

ALUMINUM TRIPOD FRAME

VELOSTAT SHEETING

LIFT LUGS MOUNTED TO HARD POINTS ON SPACECRAFT FORMING A COVER TO SPACECRAFT GROUND PATH

GROUND LUG

NOTE:

    (1) MAXIMUM RADIUS = 35.06 INCHES

    (2) COVER WEIGHT 25 POUNDS
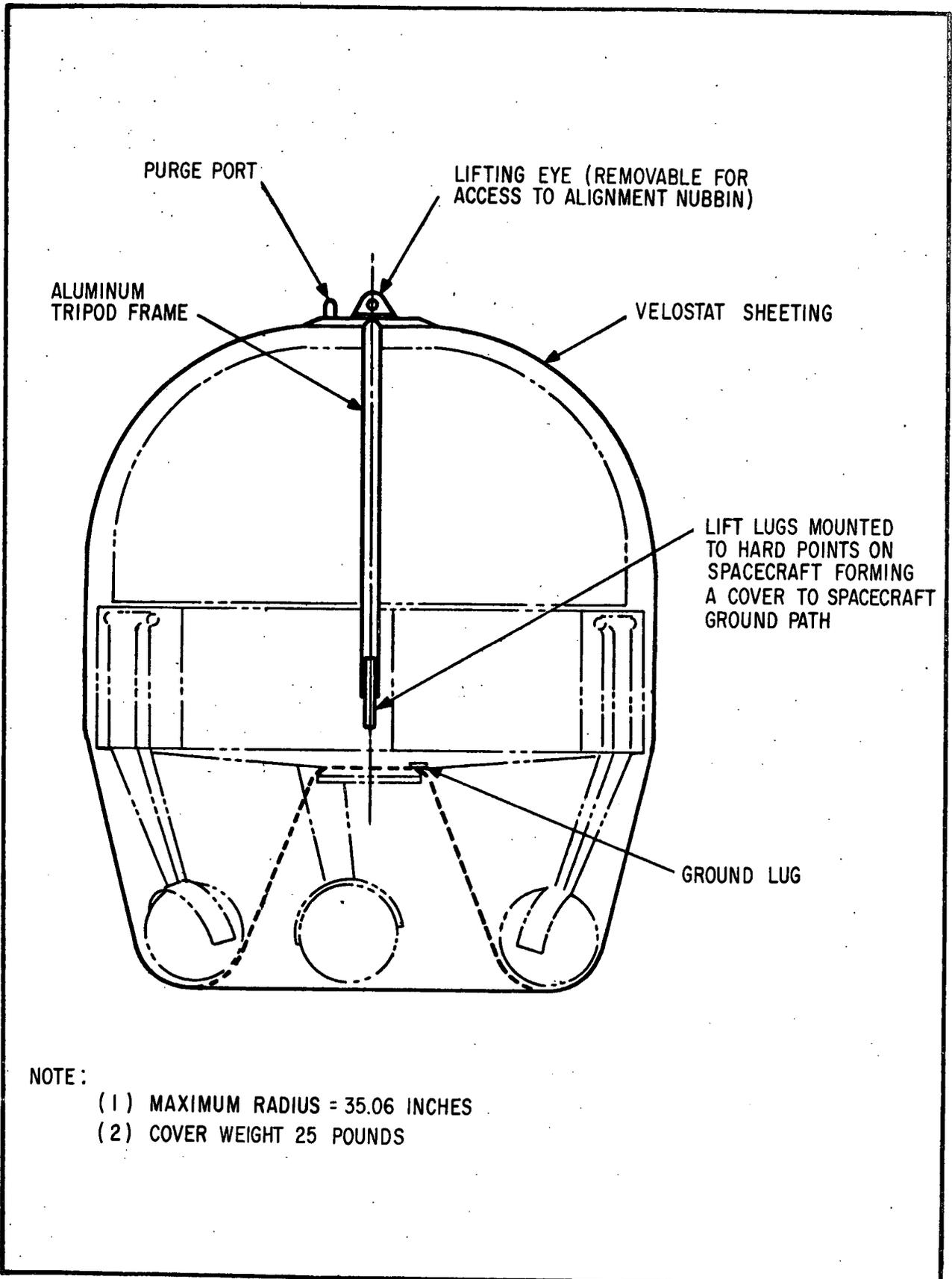
Fig. 3-18 OSO Protective Cover

Table 3-7
OSO-6 ETR PRELAUNCH MILESTONES

Date	Event
7-18-69	Spacecraft and GSE arrived at Cape Kennedy
7-19-69	OSO Ground Station set up and checked out. The spacecraft was removed from the shipping container and placed in the OSO clean room in Hangar AM where a visual inspection was performed.
7-21-69	Spacecraft overlays and experiment comprehensive test were performed.
7-22-69	Spacecraft comprehensive test conducted
7-22 to 7-23-69	Spacecraft supplemental test conducted
7-24-69	Sun pointing, the Rutgers experiment special test, and initial pressurization were performed
7-25-69	F-3 Day procedures were rehearsed
7-26-69	Spacecraft batteries were charged
7-28-69	F-1 Day launch stand rehearsal was conducted
7-29-69	Spacecraft moved to clean room in Building AE
7-29 to 8-1-69	Mechanical balance
8-1-69	Weighed spacecraft
8-5-69	Spacecraft erected on Pad 17A and F-3 Day test successfully conducted

The primary purpose of the OSO tests in the hangar was to verify that the observatory had survived the shipping activities without degradation. The observatory procedures used at Pad 17A are contained in BBRC document TN68-27, OSO-G Launch Tower/Payload

Countdown Procedures. These procedures were executed using the Observatory Automatic Test System computerized technique (Para. 3.8.1) for increased test speed and accuracy.

3.10 POST-LAUNCH ACTIVITIES

Liftoff of the Delta 72 launch vehicle, carrying OSO-6 into orbit about Earth, occurred at 0352 hours (EST) on 9 August 1969. During the following two years, BBRC provided operational surveillance, consulting services, data reduction, and performance analysis as required by the NASA/GSFC OSO Project Office. These post-launch activities are described briefly in this subsection.

Short-Term Activities. During the OSO-6 liftoff and early-orbit phases, selected personnel of the BBRC OSO project team were stationed at Hangars AE and AM at Cape Kennedy, and at GSFC Mission Control Center in Greenbelt, Maryland. Other BBRC personnel were assigned at the BBRC ground station (Fig. 3-19) in Boulder, Colorado. These people were on station during the liftoff, early orbit, and experiment turnon phases to acquire, reduce, and evaluate data from OSO and serve as consultants if needed. No significant problems arose during this part of the mission.

Data acquired during this short-term monitoring activity was used by BBRC in analyzing the initial temperature stabilization and battery charge rates, and in verifying spacecraft injection attitude. Spacecraft-systems performance during flight is presented in Section 2 of this report.

As directed by the GSFC project office, BBRC's post-launch support efforts at Kennedy Space Center concluded after the experiment instruments were energized. All ground-support equipment, including computer facilities and flight spare hardware were shipped back to BBRC, Boulder, Colorado.
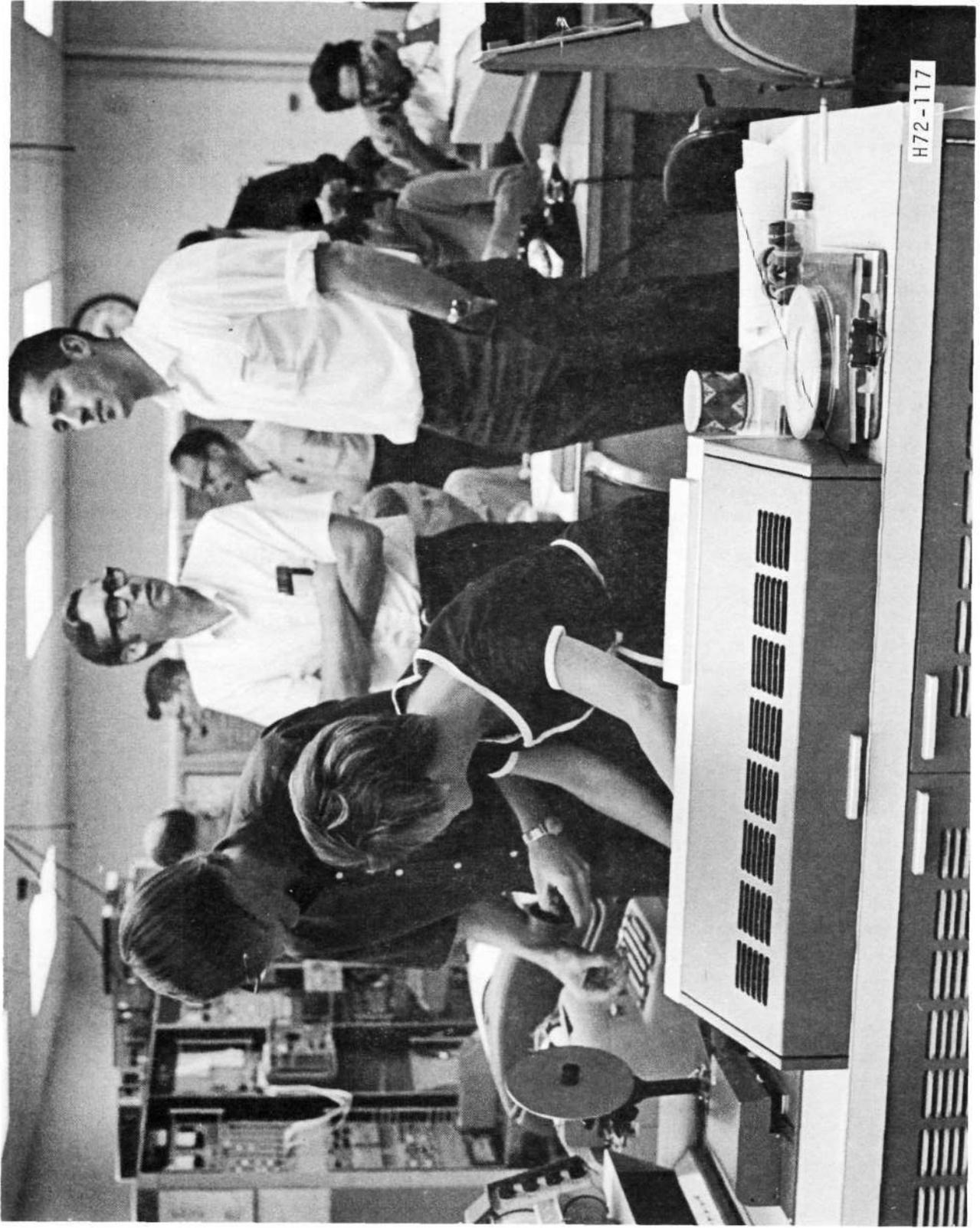
Fig. 3-19 OSO Ground Station During Launch

H72-117

Long-Term Activities. Post launch support and data reduction
continued at the Boulder facility for two years of orbital
operation. Daily TWX information, forwarded by GSFC to BBRC,
provided readouts of spacecraft performance parameters for
timely evaluation by BBRC project engineers. Performance factors
were then reported back to the GSFC-OSO project office as part
of the monthly progress reports.

Orbit-profile and noontime plots were prepared by the BBRC-OSO
data-analysis group as described in Para. 4.3. These plots
were used by BBRC engineers to analyze systems performance. BBRC
was thus equipped to identify problems as they developed, and to
recommend remedial action when necessary.

Ultimately, analysis and interpretation of the reduced and plotted
OSO-6 data were used by BBRC engineering personnel to:

- Summarize OSO-6 performance for inclusion in
  Section 2 of this report

- Predict performance of future observatories, and

- Recommend future spacecraft design improvement.

Subsystem performance analysis was accomplished by the engineering
teams responsible for the design of the major subsystems.

Section 4
DATA REDUCTION COMPUTER PROGRAMS

Various computer software systems used to support the OSO-6
program are described in brief in this section.

4.1   EXPERIMENT PRINT ROUTINES

The OSO-G experiment data processing system provided the following
basic functions:

- It interfaced with the ground station facility
  to enable the realtime acquisition of telemetry
  data and the sending of commands.

- It enabled the operator to establish the operating
  modes of the system, send commands, or select
  processing routines to decode and display the
  data from chosen experiments. (The operation
  of the system is controlled by operator instructions
  through the typewriter.)

- It was able to write and process magnetic tape
  records of the test data.

The OSO-G system employed an integrated array of hardware. The
interfacing of the hardware components (observatory systems,
ground station facility, and computer facility) is shown in
Fig. 4-1.

The observatory data and command systems were an integral part
of the system hardware. Measurement and status data from each
instrument appeared in the telemetry format as presented in
Section 9 (Fig. 9-2) of this report. Commands were allocated
to each experiment instrument as shown in Section 10 (Table 10-1).

OBSERVATORY
SYSTEMS

DATA

COMMANDS

GROUND STATION
FACILITY

TELEMETRY
RECEIVERS

BACKUP
RECORD

COMMAND
CONSOLE

AMPEX
RECORDER

DATA
DECOMMUTATOR

REALTIME   OFF-LINE

COMPUTER
FACILITY

DATA
INPUT

DIGITAL
DATA
TAPE

MASTER
PROG
TAPE

CDC
160A/
8090

COMMANDS

TYPED
INSTRUCTIONS

EXPERIMENT
DATA
PRINT OUT
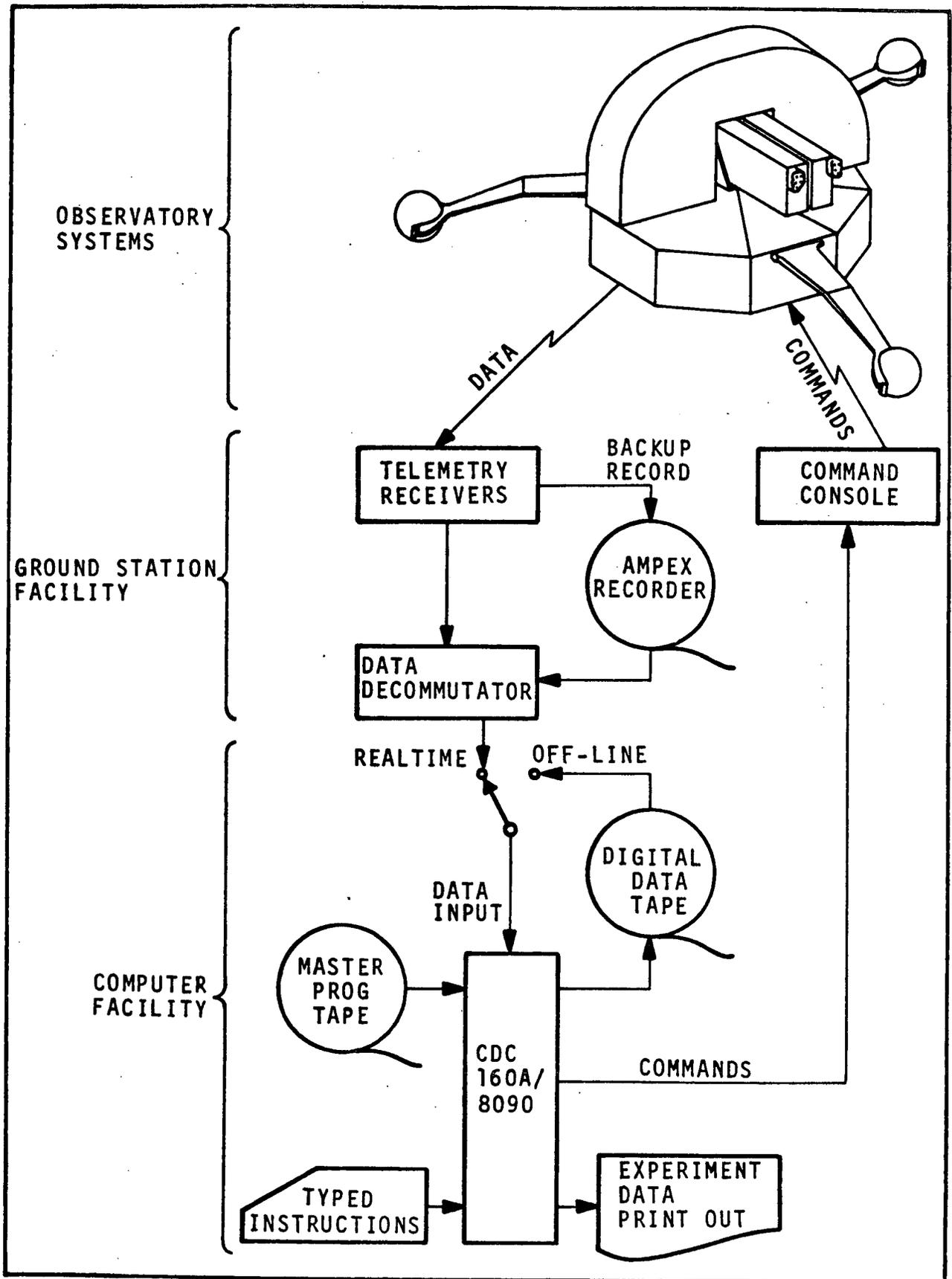
Fig. 4-1 Data Processing System Interfaces

The OSO-6 telemetry data was routed to the computer through the ground station. Radio signals from the OSO's transmitter were sent to the ground station receivers which converted the signals to a bi-phase, 800-cycle, pulse train. This endless data pulse train was grouped into 8-bit words by a telemetry decommutator. The decommutator is able to do this by observing the unique two-word main frame sync pattern. The decommutated data was then sent to the computer.

Commands from the computer were routed to the OSO through the command console which was equipped with a transmitter. This console could be monitored by a command verifier which printed a record of the commands sent.

The computer used for the testing activity was a Control Data Corporation system, consisting of:

    (1)   CDC 160-A, main frame core and processing unit (8K)
    (2)   CDC 161, input/output on-line typewriter
    (3)   CDC 166-2, 300 line/minute printer
    (4)   CDC 604, magnetic tape units (2)
    (5)   CDC 165-2, plotter
    (6)   Paper tape punch and reader
    (7)   Friden flexowriter

Items 1, 3, 4, 6, and 7 were used in the preparation and maintenance of the experiment data processing program. Items 1, 2, 3, 4, and 6 were used in the operation of the program during testing.

A second computer system was used to provide faster turn-around times for printing of test data and permitted simultaneous test operations of the spacecraft, and an experiment, or two experiments. This second system was identical to the system specified above except that the central processor was a CDC-8090 unit. There was 100% program compatibility between the two systems.

A large portion of the experiment data processing program remained in core during testing. This portion, called the "Main Program", provided the basic capabilities of the system, including:

- Initialization of the system's operating modes, realtime operator control, and command capability.

- Input and decommutation of realtime telemetry data from the ground station.

- Realtime input/output processing for the control typewriter, printer, and paper tape reader.

- Realtime output of magnetic tape digital history tapes, and the off-line processing of these tapes.

The interpretation and display of experiment telemetry data was performed by special print or evaluation routines. Only one print routine could be selected at a time to print the status and scientific data from a given experiment. One or more special evaluation routines could be selected to run concurrently with a print routine or by themselves. Critical messages from the evaluation routines were sent to the operator via the typewriter.

The interfacing of the computer program routines and experiment print routines is illustrated in Fig. 4-2.

FOLDOUT FRAME #1

FOLDOUT FRAME #2
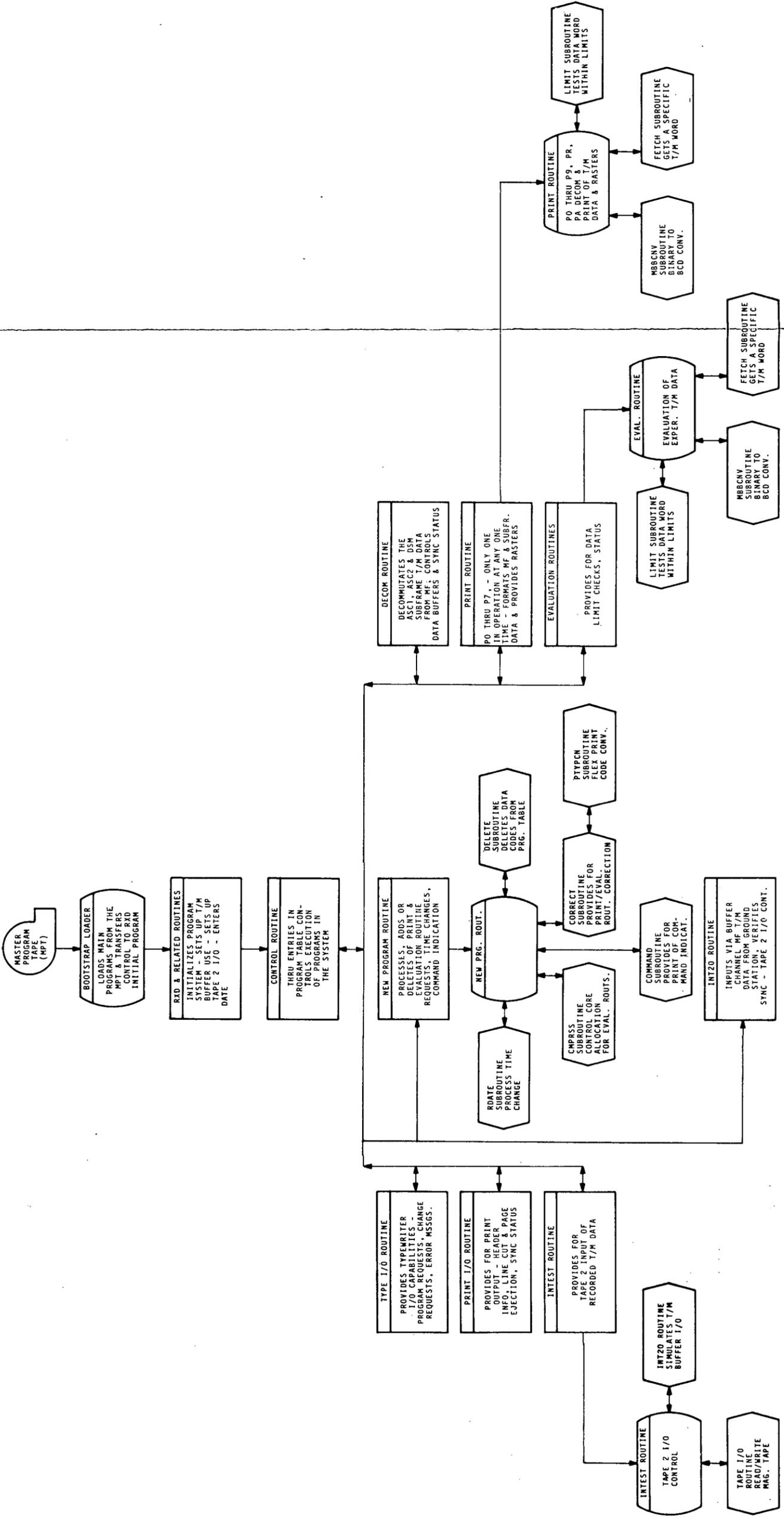
Fig. 4-2 Data Processing System

4-5

The experiment print routines decommutated individual experiment and subframe data and placed this data in the desired format for hard copy output. Each routine indicated the current sync status of the telemetry system on every detail line of print and indicated each OSO command at its occurrence. Only one routine could be in operation at a time. The routines developed were:

- P0   Main Frame Telemetry Data

- P1   Los Alamos Scientific Laboratory (Wheel)

- P2   University of Bologna (Wheel)

- P3   Rutgers State University (Wheel)

- P4   University of New Mexico (Wheel)

- P5   University College, London (Wheel)

- P6   Naval Research Laboratory (Sail)

- P7   Harvard College Observatory (Sail)

The main frame telemetry data routine (P0) extracted the 30 main frame data words from the main-frame buffer area, converted them to BCD, and moved them to a print output area. A sync check was made of the main frame and the three subframes. When any of the frames were out of sync, an asterisk was placed in the first print position; otherwise, an "S" was printed indicating full sync.

Any print routine requested following the operation of P0 required that the complete system be reloaded. This was because the loading of the P0 routine destroyed the sync ID and command indication

4-6

heading words in core. See Fig. 4-3 for the main frame print format.

To place the system in operation, a special "bootstrap loader" routine was used to load the main program from the master program tape. The loader then started the program at the initialization routine. After the operator completed his initialization instructions, the control routine started the realtime functions.

Once the system was started, the operator could select print routines, one at a time, which were loaded from the master program tape and executed in realtime. The operator could also select one or more realtime evaluation routines, which were also loaded from the master program tape. Evaluation routines could either run concurrently with a print routine or by themselves. During the realtime test activities, the operator could send commands via the computer to establish observatory or instrument operating modes. In addition, the operator had the option to write digital telemetry data output tapes for historical purposes.

Digital telemetry data tapes which were made during testing could be reprocessed by the system later. All the same data processing options were available for later processing as for realtime processing of data. Commands sent during the acquisition of the data appeared in the printouts of reprocessed data.

A typical experiment print format is shown in Fig. 4-4. The print formats for each experiment included the experiment data and any spacecraft parameters required for proper interpretation of experiment performance. Available information included the following items as appropriate:

- Telemetry sync condition
- Wheel spin rate data
- Telemetry frame count

Fig. 4-3 Main Frame Print Format

NAVAL RESEARCH LABORATORY SOLAR X-RAY   P. STRINGHAM   R. KREPLIN   E. TINDALL   E. WORNER   MM/DD/YY PAGE XXXX

M12D	PROP-PHA	SCINT-PHA	LIF	EDDT	KAP	SCAN-FLARE	SPOT POSITION	BURST BKGD	SYSTEM STATUS	CMD
****	AXXXXX X	AXXXXX X	952	952	952	896 3968	2040 AAAA XXX	4080 255		
SSSS	XXXXX 1	HI				S-O-L XX	EL PT XXX		BATT V XXX	
	3	65535 1				S-O-R	EL XXX			
	5	* 8192 3					AZ PT XXX		BRAGG: HV1 IN ON AT XXX	
	7	5							*STEP- XXXX	
	2	7					AZ XXX		SLOW MODE	
	4	2							LAMDA S-TO-L	
	6	4					EL XXX		MOTOR 1 SEL	
		6					AZ XXX		+4V-HI	
							MODE XXX		CHANLS NORMAL	
									XTAL T XXX	
							AZ XXX			
	LO								PROP: HV1 IN ON AT XXX	
							EL XXX		SCINT: HV1 IN ON AT XXX	
				XXX * XXX * XXX			AZ XXX			
									PHA: +4V-HI	
							AZ XXX		RASTR: HV1 IN ON AT XXX +4V-HI	
							EL XXX		IN H-I MODE	
							AZ XXX			
							EL GN XXX		BURST: HV1 IN ON AT XXX +4V-HI	
							AZ GN XXX		CHANLS NORMAL	
									LV -1 IN +4V XXX	
							AZ XXX			
							AZ XXX		+9V T XXX +12V XXX	
									FWD T XXX CAST T XXX	

Fig. 4-4 Typical Experiment Print Format

- Day or Night condition flag
- Cumulative total of commands during print
- Observatory temperatures as required
- Main battery voltage
- Experiment operating mode
- Instrument aspect
- Az and El Raster and Point Position
- Raster signal generator monitor
- Servo coarse/fine (on target) signal
- Offset mode status
- Raster SOL or SOR signals

Use of these experiment print routines contributed greatly to accuracy and efficiency in testing instruments during the I and T phase of OSO-6 development.

4.2   OBSERVATORY AUTOMATIC TESTING SYSTEM (OATS)

The computerized system used to test OSO-6 was called OATS (Observatory Automatic Testing System). It was used with the CDC-160A computer to help execute selected OSO tests, where automation offered significant advantages over manual testing.

The OATS system had two parts - program and procedure. The program prepared the computer to print, read, and interpret steps in the procedure, and to execute automatic-type functions and verifications through ground-station equipment. The procedure was a serial description of the test. It was formatted and coded on magnetic tape for computer interpretation. The format allowed the use of language that is easily read (in printed form), even by a reader not familiar with computer language. Fig. 4-5 presents a sample page of OSO test procedure in OATS format.

The editing system used to prepare or revise OATS procedures was called "Super-Edit". Super-Edit was itself a computerized

```
7         ACT.  DIR.
          BY    OF          OPERATION                                      ITEM

2         160A         READ AND VERIFY DSM CHANNELS
   013                      SORE SPIN RATE              D-12(000-255)
   014                      SORE SPIN RATE              D-13(000-192)        (---)

2         NOTE--THE ABOVE TWO SETS OF DSM-12 AND DSM-13 READINGS MUST BE
          CONSECUTIVE. USE THE TABLE BELOW ONLY IN THE EVENT OF A BREAK-
          DOWN IN THE AUTOMATIC-TEST SYSTEM.

2         160A         COMPUTE SPIN PERIOD                    02

2                           TABLE 3-2
              -- ASPECT SPIN PERIOD CALCULATION IN DAY CONDITION --
          GSE-CT    OBTAIN TWO CONSECUTIVE SETS OF READINGS OF DSM
                    CHANNELS 12 AND 13 AND CALCULATE SPIN PERIOD IN MSEC.  (---)

2                                                    SET 1
                                     DSM-12          ---                  (---)
                                     DSM-13          ---                  (---)
                                  4(DSM-13)          ---                  (---)
               TOTAL COUNT = DSM-12 + 4(DSM-13)      --- (S1)             (---)
                             2.5 MSEC X S            --- (T1)             (---)

2                                                    SET 2
                                     DSM-12          ---                  (---)
                                     DSM-13          ---                  (---)
                                  4(DSM-13)          ---                  (---)
               TOTAL COUNT = DSM-12 + 4(DSM-13)      --- (S2)             (---)
                             2.5 MSEC X S            --- (T2)             (---)

2                               DSM FRAME TIME      15360   MSEC
                                   MINUS T1       - ---     MSEC          (---)
                                   DIFFERENCE       ---     MSEC          (---)
                                   PLUS T2          ---     MSEC          (---)
                                   SUM              ---     MSEC          (---)
                           (SEE NOTE) DIVIDED BY     -                   (---)

2  011                 SPIN PERIOD (1983-2023 MSEC)   ----    MSEC        (---)

2         NOTE-- IF T2 IS GREATER THAN T1, DIVIDE BY 8,
                 IF T1 IS GREATER THAN T2, DIVIDE BY 7.
          (THE ANSWER CALLED FOR IN THE ABOVE ITEM IS THE TEST-EQUIPMENT
          CALIBRATION NUMBER PLUS OR MINUS 20 MSEC OR 1 PERCENT TEST-
          EQUIPMENT TOLERANCE.)

2    CT         VERIFY THAT CALCULATED SPIN PERIOD IS WITHIN THE
                INDICATED LIMITS                                          (---)

2    OT         VERIFY THAT OSO IS OPERATIN ON BATTERY POWER.             (---)

2    OT         CONNECT A VOLTMETER TO THE *MAIN-BATT VOLT* TERMINALS
                AT THE BACK OF THE CONSOLE.                               (---)
```

Fig. 4-5   Sample Page of OATS Procedure

system, consisting of edit instructions and a program which
enabled the computer to accept the instructions. The computer
accepted existing sources of OATS statements along with the editing
instructions to produce or revise an OATS procedure. Using
Super-Edit in this way saved a great deal of retyping of
procedures.

4.2.1   OATS Background

Early in 1964, Experiment Accommodation personnel started using
the 160A to assist in experiment testing. This was required
since most OSO experiments provide data so quickly that normal
ground-station printing equipment could not keep pace. Special
print programs greatly simplified analysis of experiment data.

It became obvious, after a brief experience with computerized
testing of experiments, that computerized testing could also be
used to advantage in OSO spacecraft-systems tests. OATS proved
its value very quickly, and was subsequently adapted for use on
all OSO programs.

4.2.2   Description of OATS Functions

In general, there were three types of statements in an OATS
procedure:

(1)   Automatic-Type Functions --
      These are functions which can be performed by the
      computer through control of ground-station equipment.

(2)   Manual-Type Functions --
      These are functions which must be performed
      manually by the test operator.

(3) Information Statements. --
These include explanatory notes, cautions,
warnings, and information for the test operators.
They generally provided all the instructions
necessary for manual execution of the test.

The test engineer decided whether to computerize on the basis
of his evaluation of the advantages offered in the particular
test being considered. This was ordinarily determined by
the number of steps in the procedure which could be adapted
for automatic execution.

The OATS-system automatic functions were generally limited to
ground station activities and to certain calculations. Some
of these functions are described as follows:

(1) The computer could verify main-frame and sub-
commutator sync and frame length. It sampled
incoming data until the sync appeared, then
indicated sync and proceeded to the next step
in the procedure.

(2) The computer could receive telemetry data from
any of the spacecraft subcommutators and compare
it with predetermined limits. If the reading was
out of limit, the computer would print an asterisk
(*) next to the reading and cause that group of
readings to indicate a "no go" condition. This
returned control to the computer operator. If
the reading was within limits, the computer
advanced to the next step without manual controls.

(3) The computer could address any OSO command decoder
and execute any command. If an attempt was made
to send an invalid command or address (one not
listed in storage), that command was blocked and
control was returned to the operator.

(4) Several "compute" functions were incorporated into the system. These enabled the computer to perform time-consuming calculations almost instantaneously and with great accuracy.

4.2.3   Improvements in Testing Provided by OATS

When using the automatic-function features of OATS for testing OSO, a great deal of test time was saved. The magnetic-tape records of tests greatly simplified data processing efforts, and reproduction of test records. Test accuracy was improved, providing added safety to the items tested, and variations in testing were substantially reduced.

It should be pointed out that certain types of testing of tape recorders, offset mode, and others, would have been economically impractical without the automatic functions.

During testing, certain types of unscheduled operations were automatically recorded with the test data, providing a more complete record of the test. The editing process used to revise procedures saved an enormous amount of publication effort.

4.3   FLIGHT DATA PLOTTING PROGRAMS

The data-plotting techniques used to support OSO-6 post-launch operational surveillance and performance analysis were essentially the same as those used on earlier OSO's. These techniques were thoroughly tested through use in the observatory integration and test phase.

Computer programs allowed the CDC-160A computer to read OSO data from the BBRC ground station or from the digital history tapes supplied by NASA/GSFC. Special programs were used to process the data and write the processed and edited data on magnetic tape.

Further programs were used to read the processed data tapes and operate the plotter to produce graphs. The programs were designed for maximum modularity. At times when real-time and history-tape data were not available, orbit-noon data for the long-term plots was obtained from available TWX and all-channel-printout communications from GSFC. The manual plots of this data were then available for timely surveillance and analysis.

The three types of plots for which computer programs were written are described below. Samples are included in Section 2 of this report.

Orbit Profile Plots (Type A). These plots each showed the behavior of a particular parameter throughout a period of about 1.2 orbits. The ordinates were in the physical units of the parameter, and the abscissas were graduated in percentage of orbit. These graphs documented performance throughout a full day/night cycle of observatory operation. They were prepared at frequent intervals during the days immediately following launch, and thereafter monthly throughout the first six months of operation.

Long-Term Orbit-Noon Plots (Type B). These were graphs of single-sample orbit-noon values of particular parameters. The ordinates were in physical units and the abscissas were graduated in whole-number orbits. A sample was plotted from each orbit noon. The Type B machine plots gave way to similar manually-prepared plots when difficulty was experienced in obtaining digital history tapes on a timely basis. The manual plots were then used for timely surveillance and analysis of OSO-6 performance.

<u>Refined Orbit-Noon Plots (Type C)</u>. These were similar to the
Type B plots except for provisions for deleting bad data points
and averaging from two to nine samples for each plot entry.
Telemetry-system anomalies and transient effects were thus
smoothed in the curves, making the Type C plot more useful for
identifying long-term trends.

Section 5

ASPECT MEASURING SYSTEM

5.1   GENERAL DESCRIPTION

Generally, data received from an OSO experiment will not be
useful unless it can be determined from what direction it came
from and at what time. The measuring of these directions and
recording the times at which the data was observed is called
"aspect determination". An on-board aspect system is used to
measure the various aspect parameters. This data is telemetered
to the ground where it is processed to determine the desired
aspect information such as roll attitude, pitch attitude, wheel
spin rate, and the instantaneous azimuth position of any point
on the wheel.

The aspect system consists of the Spin Orientation and Rate
Electronics (SORE) housed in compartment 8, a flux-gate mag-
netometer mounted near the end of arm 4, a spin sensor (aspect
eye), which is part of the spin sensor assembly mounted on the
rim panel of compartment 8, and a pitch readout eye mounted on
the front of the sail. This system is identical to that used
on OSO-5 and experience has shown that its roll aspect accuracy
is about one degree. ·

The heart of the coarse aspect system is the SORE assembly.
Its purpose is to accept inputs from the magnetometer and the
aspect eye and measure the time intervals between these signals
and DSM word gate 3. The intervals are encoded into eight-bit
binary words and telemetered to ground. During the day, the
interval between the magnetometer and aspect eye pulses provides
a means of measuring the wheel-plane angle between the geomag-
netic field vector (known at all points near earth) and the

solar vector. Comparison of this angle with the known angle between the field and solar vector is then used to calculate OSO's *roll angle*. Pitch readout data is used to correct for non-alignment of the wheel plane with the solar vector. This fixes the wheel plane in local coordinates. The roll angle is then fixed in the celestial sphere by simple coordinate-system transformations.

During day, *pitch orientation* is determined from the data telemetered from the pitch readout eye. The accuracy of this sensor is better than 0.2 degree. During night, the pitch angle is extrapolated by using the pitch drift rate measured during the day, and can easily be predicted to within 0.2 degree.

The *wheel spin rate* is found within ±1% by using the period of either of the two once-per-revolution signals. The SORE does this by measuring the intervals between consecutive magnetometer pulses or aspect eye pulses. As a backup, the spin rate can also be calculated using data from channel 12 of the wheel ASC. Of the two methods, the first is the most accurate and can also be used to compute the night spin rate while the latter cannot.

The same data used for spin rate computation is also used to calculate the instantaneous *azimuth position* of a point on the wheel. During orbit-day, the solar vector is used as the reference; during orbit-night, the geomagnetic field vector is used. Time correlation is referenced to DSM word gate 3.

5.2   ASPECT GEOMETRY

5.2.1   Coordinate Reference Systems

The five coordinate systems (experiment, spacecraft, ecliptic, celestial, and topocentric) are shown in Fig. 5-1. It is possible to go from any one system directly to any other; for practical reasons, however, a logical sequence of transformations is performed. For example, instead of direct transformation from the experiment system to the topocentric system, we proceed in four steps: Experiment → Spacecraft → Ecliptic → Celestial → Topocentric.

Coordinate system rotations are sufficient to arrive at the final aspect description required of OSO. Translations of systems are not necessary since the center of mass of the observatory can be considered the center of every coordinate system, due to the negligible parallax resulting from the separation of the spacecraft and earth centers.

The experiment coordinate systems and look axes, the magnetometer sensitive axis, and the aspect eye reference all rotate in the spacecraft coordinate system. The relationships between these elements are established by the wheel spin rate and a series of preflight calibration measurements.

Experiment Coordinate Systems. A separate system is defined for each experiment. The X-axis is always defined to be the experiment's line of sight. For convenience, the Y-axis is allowed to lie in the wheel plane so that only two rotations are needed for transformations between experiment and spacecraft systems. The X-axis need not lie in the wheel plane.
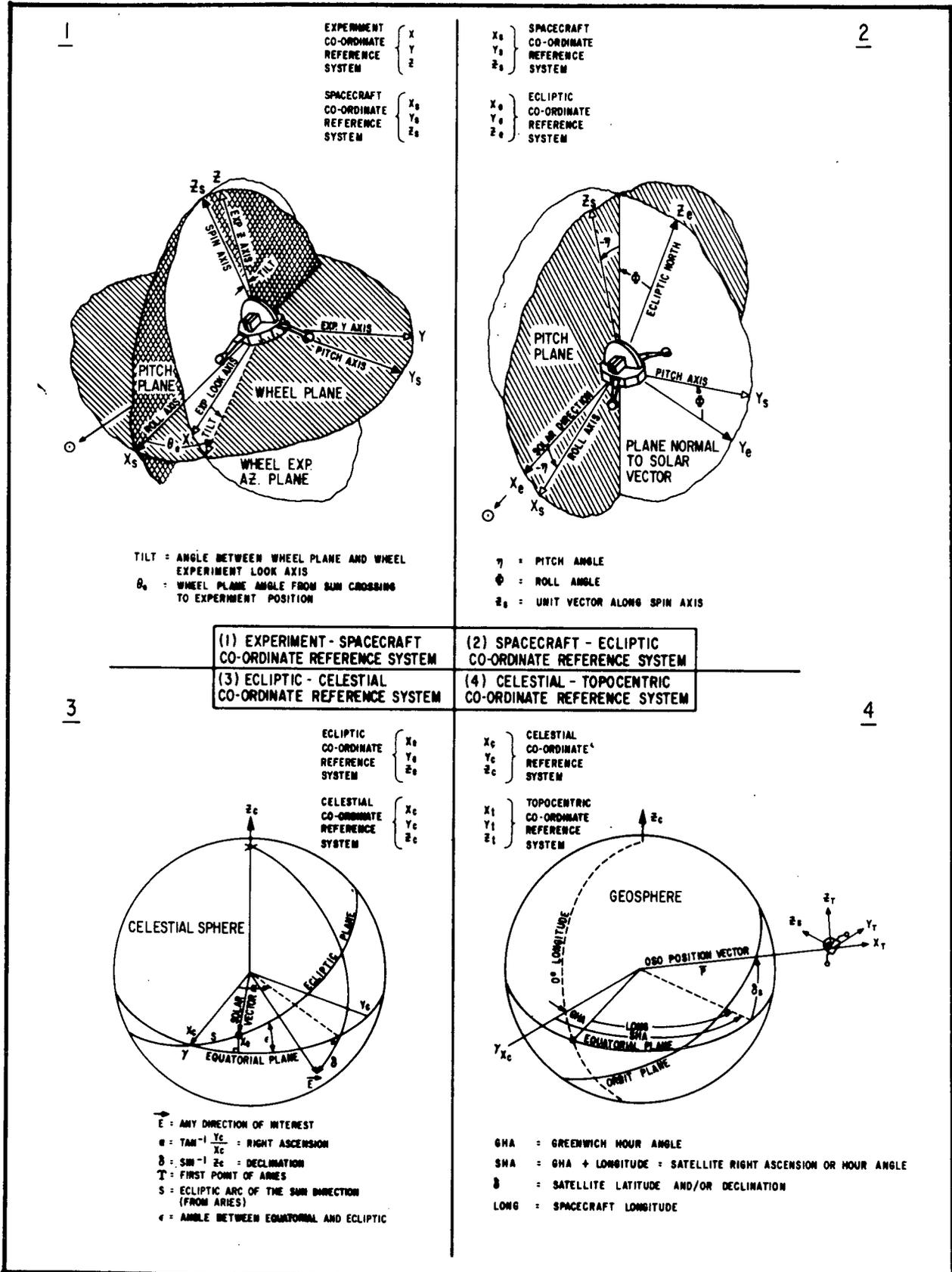
Fig. 5-1 OSO Aspect Coordinate Systems

Spacecraft Coordinate System. As seen in Fig. 5-1 (1), the spacecraft X, Y, and Z-axes are the roll, pitch, and spin axes, respectively. Also, note that the offset between the roll axis and the solar vector is simply the pitch angle (η). The wheel plane includes the roll axis and pitch axis and is perpendicular to the spin axis.

Ecliptic Coordinate System. This system is defined with the X-axis pointing from the center of the spacecraft to the sun's center. The Z-axis lies along the ecliptic pole and the Y-axis forms an orthogonal right-handed system, lying in the ecliptic plane. Note that this ecliptic system differs from the conventional ecliptic system which has the X-axis pointing to the vernal equinox.

Celestial Coordinate System. This is the conventional celestial coordinate system except that the origin may be considered either at the earth's center or at the spacecraft center of mass (interchanged at will without a significant loss of accuracy). Thus, the Z-axis is parallel to the earth's polar axis pointing north, the X-axis points to the vernal equinox and the Y-axis points 90° east of the vernal equinox in the equatorial plane forming an orthagonal system.

Topocentric Coordinate System. The topocentric system is defined with the X-axis directed upward along the local vertical (from spacecraft center of mass out from the earth's center). The Z-axis lies in the local horizontal plane (perpendicular to the X-axis) and points to the north. The Y-axis forms a right-hand orthagonal system pointing east in the horizontal plane.

5.2.2 Definitions of Angles Used to Determine Aspect

There are seven angles used in determining the various aspect parameters of OSO and its experiments. These angles are shown in Fig. 5-2 and are described below:

Pitch angle (η) — The angle between the wheel plane and the solar vector (See Fig. 5-2).

Sun-to-Mag (θ_m) — The angle (measured in a ccw direction) between the solar vector and the rising zero crossing of the geomagnetic field by the magnetometer.

Mag-to-Sun (β) — The angle (measured in a ccw direction) between the magnetometer zero crossing and the solar vector.

DSM3-to-Sun (β_s) — The angle (measured in ccw direction) between the occurrence of word gate DSM-3 and the solar vector.

DSM3-to-Mag (β_m) — The angle (measured in a ccw direction) between the occurrence of word gate DSM-3 and the magnetometer zero crossing.

Mag-to-Aspect Eye (β_a) — The acute angle between the magnetometer axis and the aspect eye axis.

Exp-to-Aspect Eye (β_e) — The angle (measured in a ccw direction) between the look axis of any wheel experiment and the aspect eye axis.

The purpose and the data source for the above angles are listed below. The aspect computations are described in subsection 5.4.

B = SORE COUNT ANGLE

B_a = CALIBRATION ANGLE

θ_m = $90° - (B + B_a)$
(VECTOR ANGLE
USED IN ROLL ANGLE
CALCULATION)

SOLAR
DIRECTION

Hmax (PROJECTION OF MAGNETIC
FIELD VECTOR IN WHEEL
PLANE)

B

ASPECT EYE
REFERENCE AXIS

θ_m

WHEEL
ROTATION

90° B_a

MAGNETOMETER PROBE SENSITIVE AXIS
(SHOWN IN POSITION OF THE POSITIVE-
GOING ZERO CROSSING, WHERE THE
MAGNETOMETER PULSE OCCURS)

(1) ASPECT GEOMETRY RELATED TO ROLL-ANGLE CALCULATION (DSM-32/33--DAY ONLY)

SOLAR
DIRECTION

WHEEL
ROTATION

B_s

ASPECT EYE
REFERENCE
AXIS (AT TIME
OF RISE OF
DSM WORD
GATE 3)

B_e

EXPERIMENT LOOK
DIRECTION IN THE
WHEEL PLANE

(2) TIME/POSITION REFERENCE (DSM)-12/13 IN DAY)

MAGNETIC NULL
(POSITIVE-GOING)

WHEEL
ROTATION

B_m

90°

ASPECT EYE
REFERENCE
AXIS (AT TIME
OF RISE OF
DSM WORD
GATE 3)

Hmax

B_a

B_e

MAGNETOMETER
SENSITIVE AXIS
(AT TIME OF RISE
OF DSM WORD
GATE 3)

EXPERIMENT LOOK
DIRECTION IN THE
WHEEL PLANE

(3) TIME/POSITION REFERENCE (DSM-12/13 IN NIGHT)

NOTE: ALL ANGLES SHOWN ARE MEASURED IN THE WHEEL PLANE AS SEEN LOOKING FROM THE SAIL
DIRECTION. THE ASPECT EYE REFERENCE AXIS, EXPERIMENT LOOK DIRECTION, AND MAGNE-
TOMETER SENSITIVE AXIS, ARE ALL FIXED IN THE WHEEL AND ROTATE WITH IT. THE SOLAR
DIRECTION AND Hmax LINES ARE A FUNCTION OF GEOGRAPHIC LOCATION IN ORBIT.

Fig. 5-2 Geometry of SORE Count Angles

η — obtained directly from telemetry on sail ASC channel 4 and converted to engineering units by use of OSO-6 Data Conversion Tables (TN68-16);used in the computation of roll attitude.

θ_m — Computed from relationship of $\theta_m = 90° - (\beta + \beta_a)$; used in the computation of roll attitude and wheel azimuth position at night.

β — Obtained from SORE telemetry data on DSM channels 32 and 33; converted to an angle by the relationship:

$$\text{Angle(deg)} = \frac{\text{SORE count}}{400 \text{ bps}} \times \text{wheel}$$

spin rate (in rps). Used in the computation of θ_m.

β_s — Obtained (during day only) from SORE telemetry data on DSM channels 12 and 13; converted to angle same as β; used in the computation of wheel azimuth position during day and during night by a special computation method.

β_m — Obtained (during night only) from SORE telemetry data on DSM channels 12 and 13; converted to angle same as β; used in the computation of wheel azimuth position during night.

β_a — A physical property of OSO and is measured before flight; used in computation of θ_m (day or night) and wheel azimuth position during night only.

β_e — A physical property of OSO and is measured before flight; used in computation of wheel experiment azimuth position.

5.3   ASPECT SYSTEM ELECTRONICS

5.3.1   Magnetometer

The function of the magnetometer is to indicate reversals of the geomagnetic field by providing a signal when its sensitive axis is perpendicular to the field. OSO-6 used the same type magnetometer used on all earlier OSO's. It is a flux-gate magnetometer built by Schonstedt Instrument Company and produces a dc voltage proportional to the geomagnetic field. A zero-crossing detector in the SORE is used to produce a pulse when reversal of the geomagnetic field is sensed.

The magnetometer consists of an oscillator, sensor, signal amplifier, and demodulator (Fig. 5-3). The oscillator provides an alternating current large enough to cyclically drive the core into magnetic saturation. If there is no external magnetic field acting on the core, the core saturates equally both the positive and negative portions of the cycle. As a result, the flux generated in the core has a symmetrical pattern. If there is a magnetic field acting on the core, the core remains saturated longer on one-half of the cycle than on the other. The flux generated in the core has an assymmetrical waveform con-taining even order harmonics of the excitation frequency, not present in the symmetrical flux pattern. For weak magnetic fields such as that of the earth, the even harmonics are pro-portional in magnitude to the intensity of the applied magnetic field.
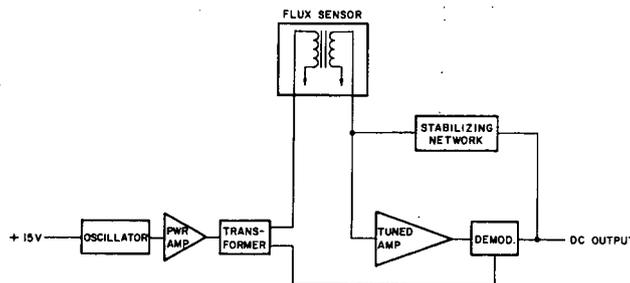
Fig. 5-3 Flux Gate Magnetometer

The amplifier signal from the oscillator excites the primary winding on the sensor. The signal from the sensor secondary passes through an amplifier tuned to the second harmonic of the oscillator frequency. The amplitude of this frequency is proportional to the strength of the magnetic field at the sensor. Using a second harmonic reference voltage (derived from the same transformer that drives the sensor) the signal is demodulated thus producing a dc output voltage proportional to the strength of the earth's magnetic field. A negative feedback network from the output to the sensor secondary is used to temperature-stabilize the circuit and provides automatic gain control.

5.3.2 Pitch Readout Sensor

The pitch readout sensor is identical to the one used on all earlier OSO's. The sensor consists of an analog sun-sensing unit and electronic signal-conditioning circuitry all contained in a common housing. It is mounted on the sail with its optical axis parallel to the spacecraft's X-axis.

The sensing unit consists of a filter/reticle and a pair of silicon photovoltaic cells arranged as shown in Fig. 5-4. The beam of sunlight that goes through the slit in the reticle falls equally on the two cells at null. As the angle between the sun-line and the sensor axis increases, the energy on one cell increases and decreases on the other.

The cells are hooked up in parallel opposition. The resulting output is linear to approximately ± 6.5 degrees (where the light beam no longer falls on one of the cells). Beyond 6.5 degrees, the output continues to increase slightly until the entire beam of light is on one cell. The output then stays constant until the light beam begins to move off the cell. The total field-of-view of the sensor is about ± 25 degrees. The electronics
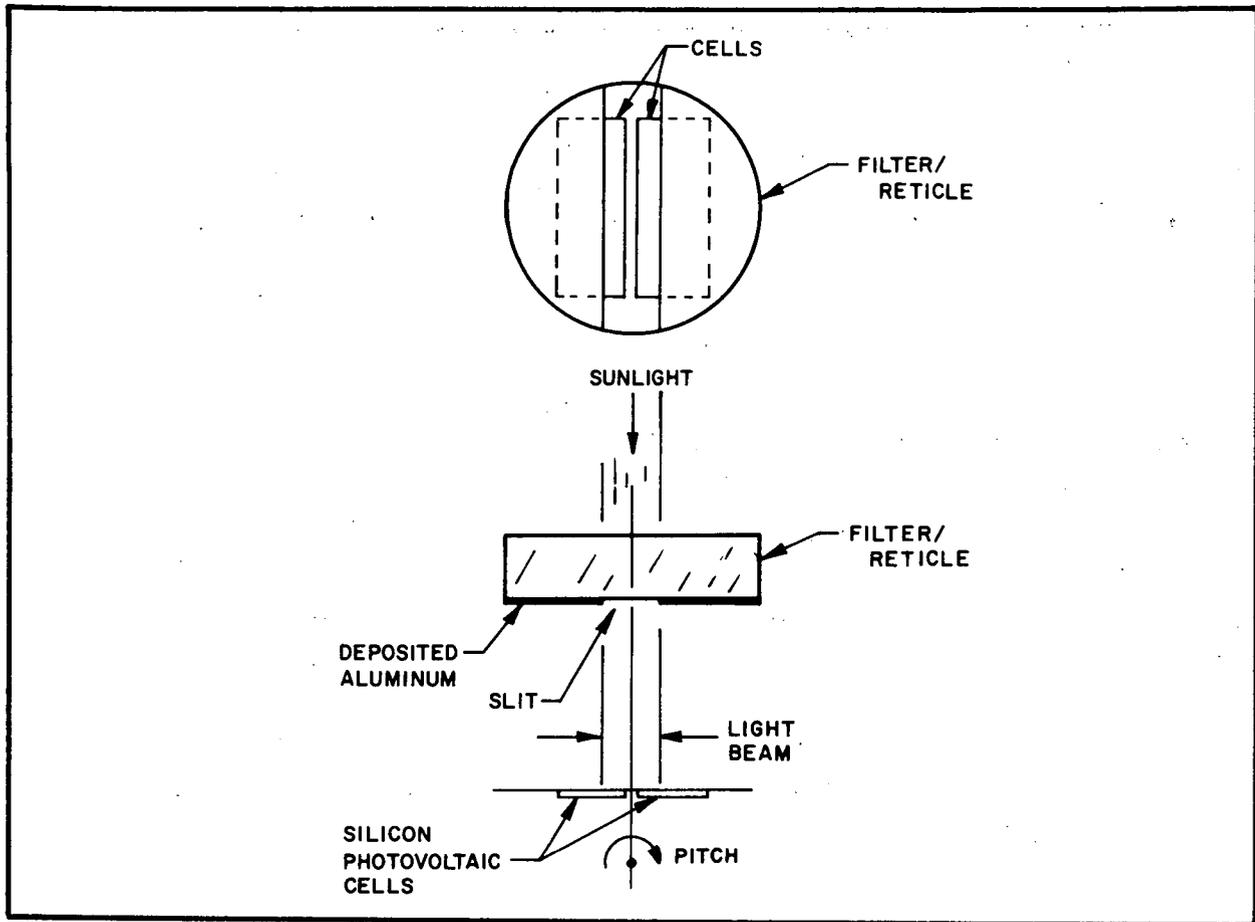
Fig. 5-4   Diagram of Pitch Readout Sensor

serve to limit the output of the sensor at an angle corresponding
to about ±5.5 degrees, resulting in the overall output-versus-
angle characteristics shown in Fig. 5-5.   This is typical
output-versus-angle curve for a unit with a 0 to 5 v output range.

5.3.3   Spin Aspect Sensor

The aspect sensor is part of the spin sensor assembly and is
identical to that used on earlier OSO's.   The spin sensor
assembly is an array of four spin target detectors on an aluminum
block, mounted on the rim of the wheel, with its sensing plane
parallel to the spin axis.   (See Section 7.)
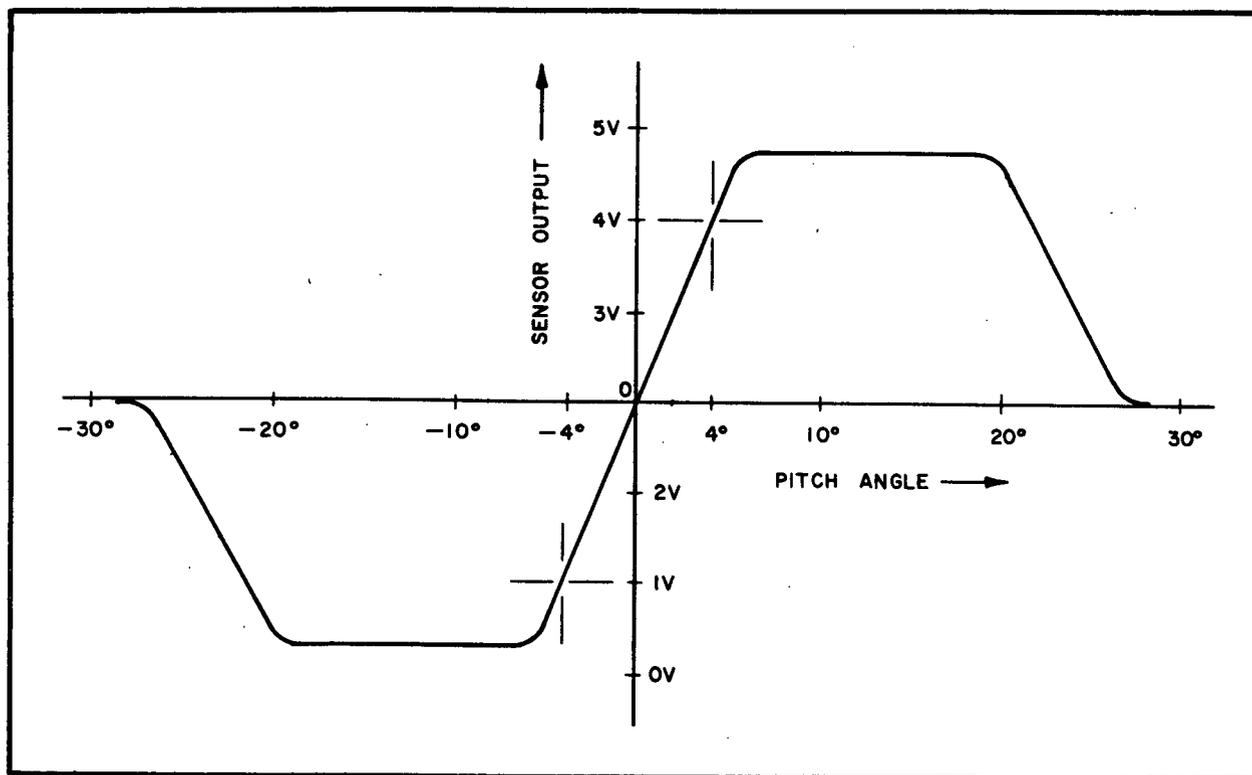
Fig. 5-5   Pitch Readout Sensor Output Curve

Each of the four spin target detectors consists of a housing, lens, aperture plate, filter, and a silicon photovoltaic cell (Fig. 5-6). The copper foil aperture plate is located in the focal plane of the lens and the slit in the aperture plate defines a long, narrow field-of-view, 26 degrees +2 degrees in elevation and 2.5 degrees +0.5 degree in azimuth. The target detectors are mounted so that the field of view of two of them are symmetrical about a line perpendicular to the spin axis while the other two are canted up 3 degrees and down 3 degrees in elevation, respectively. The outputs of the two canted detectors go to the automatic spin control system. The other two detectors provide outputs for the SORE package and the spin rate limit detector in the spin control assembly.
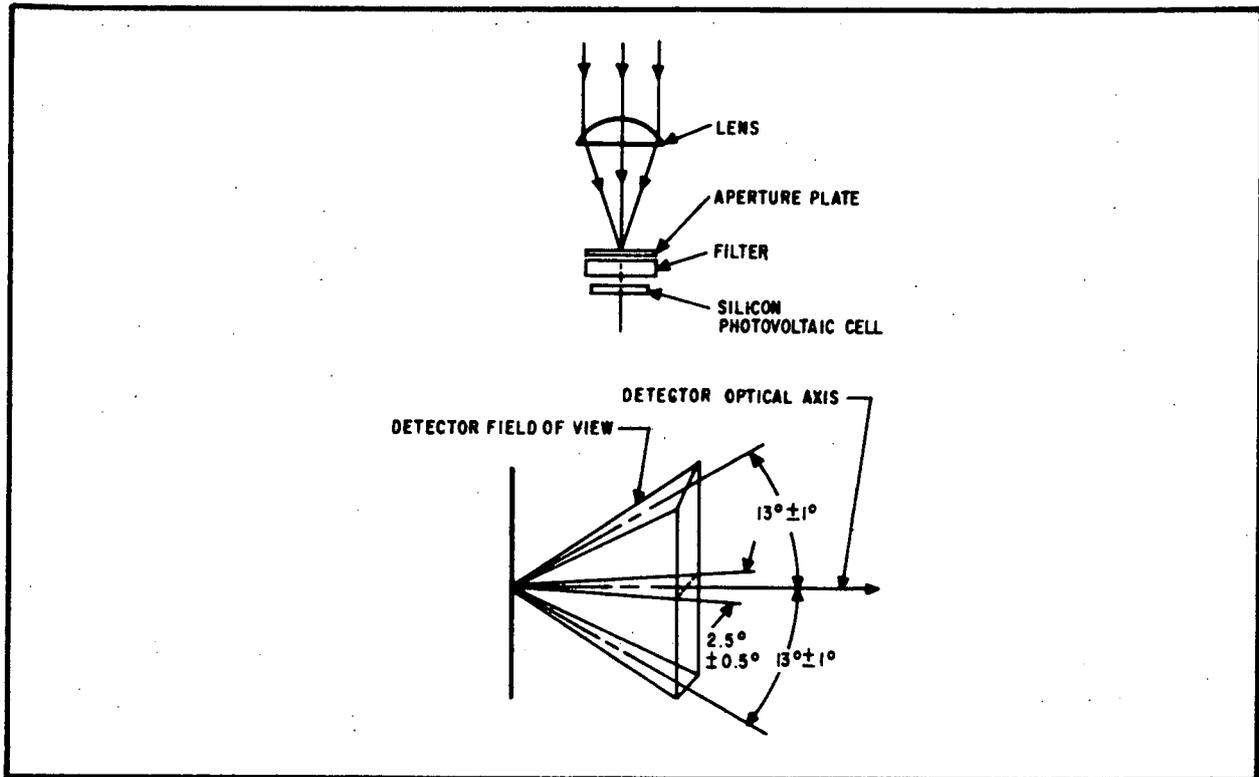
Fig. 5-6 Aspect Detector

5.3.4   Spin Orientation and Rate Electronics (SORE)

The purpose of the SORE is to measure (in telemetry counts) the period between: magnetometer pulse to sun pulse (β), DSM word gate 3 to sun pulse (β_s) and DSM word gate 3 to magnetometer pulse (β_m). These periods are telemetered to the ground where they are converted to angles and used for computing spin rate, roll angle, and wheel azimuth position (see paragraph 5.4). Figure 5-7 is a block diagram of the SORE and Fig. 5-8 is a timing diagram.

● *Measuring DSM-to-sun (β_s) and DSM-to-mag (β_m) Periods*

The sequence starts with the 400 bps clock (the result of dividing the 800 bps clock by 2 in the clock amplifier) being grounded through flipflops (FF) 1, 3, and 4. DSM word gate 3 then occurs and changes the state of FF 1, thus allowing the
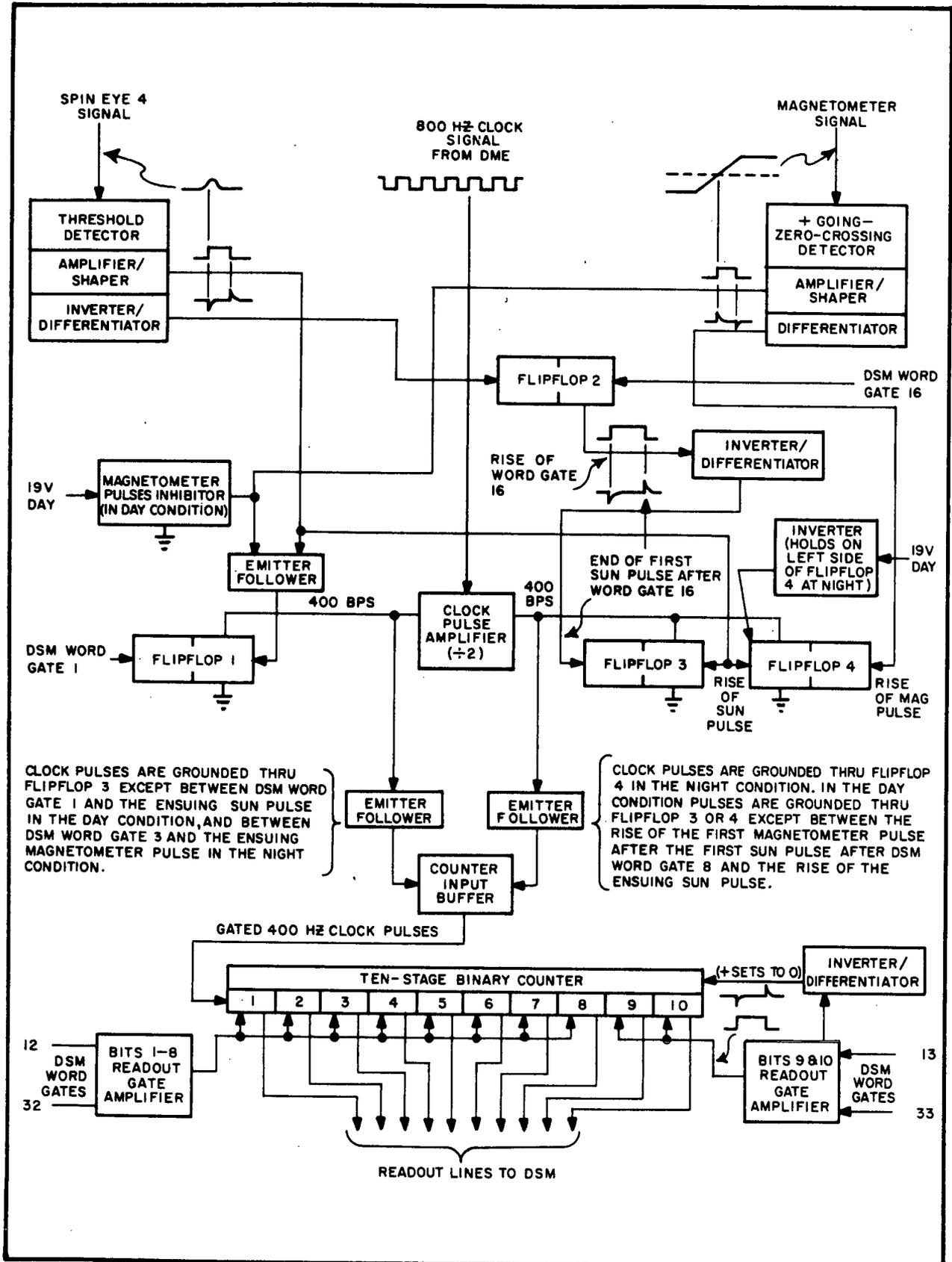
Fig. 5-7   SORE Block Diagram
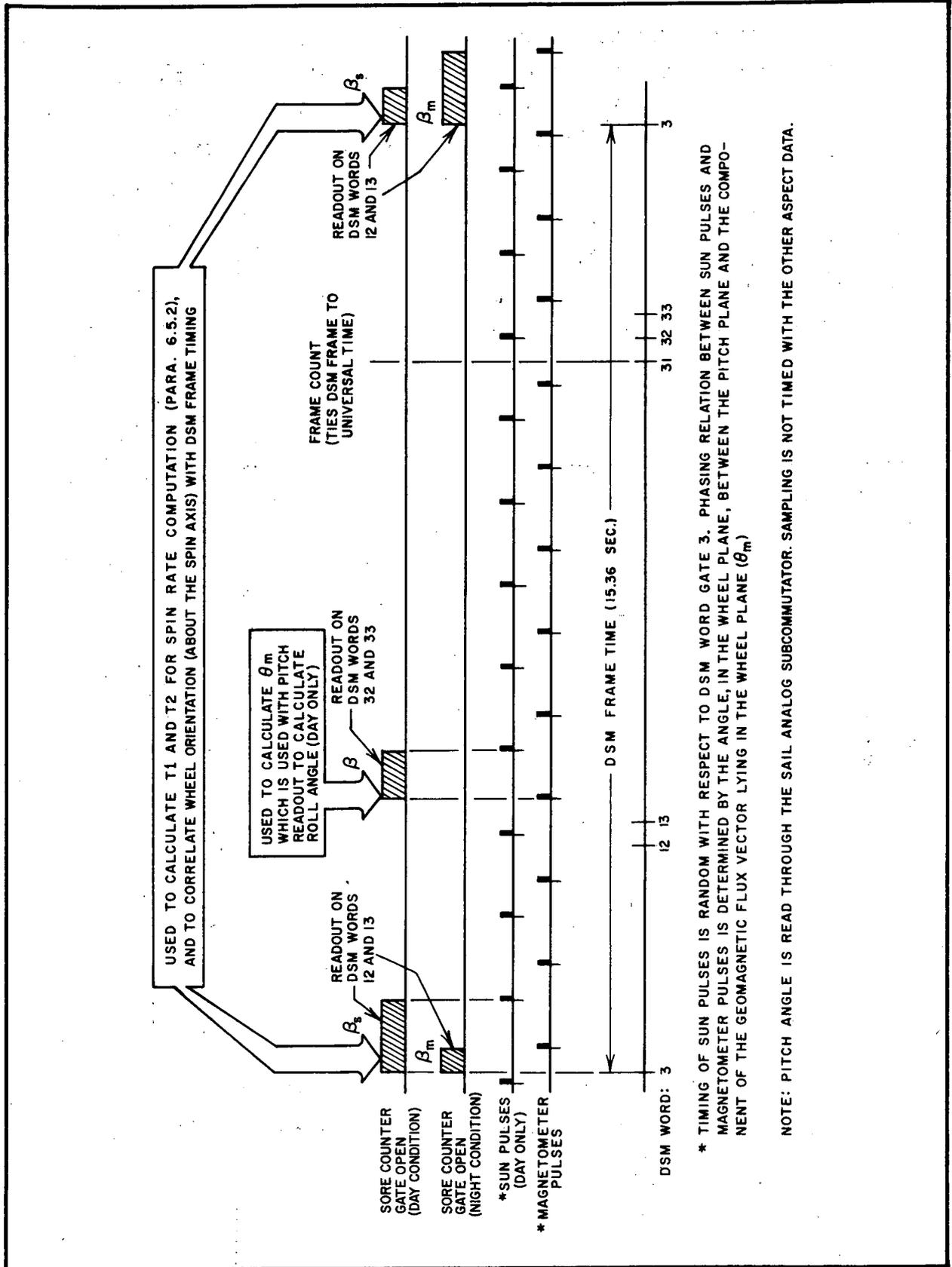
Fig. 5-8 SORE Timing Diagram

clock pulses to enter the 10-bit counter. The counter registers
these clock pulses until the occurrence of either the next sun
pulse (when in the day condition) or the next magnetometer pulse
(when in the night condition). Either of these two pulses stop
the count by setting FF-1 back to the state which grounds the
counter input. The magnetometer pulses are kept from reaching
FF-1 in the day condition by an inhibitor. The count so loaded
into the counter, (which represent β_s in day and β_m at night),
remains in the counter until DSM WG-12, at which time the first
8 bits are read out in the DSM subframe. The 2 most significant
bits are then read out by DSM WG-13, the trailing edge of which
clears the counter.

- *How the Mag-to-Sun (β) Period is Measured*

First, WG-16 sets FF-2 to its positive state. The following
sun pulse sets it back to its negative state, and an inverted
differentiated pulse sets FF-3 to the state in which it does
not ground the clock pulses. The same sun pulse, however, sets
FF-4 to the state which does ground the input to the counter.
The following magnetometer pulse, then, sets FF-4 to the state
which allows the clock pulses to enter the counter. The count
registers until the leading edge of the sun pulse, which then
grounds the clock pulses through FF-3 and FF-4. The count
representing β remains in the register until WG-32. Meanwhile,
FF-3 continues to block the input to the counter, while FF-4
cycles on and off with each magnetometer and sun pulse.

The first 8 bits of the count representing β are read out on
DSM word 32. The 2 most-significant-bits are then read out on
WG-33, the trailing edge of which clears the counter. DSM-3
starts the sequence over again 15.36 seconds later.

During orbit night, the absence of spacecraft day power is used to hold FF-4 in the condition which grounds the DSM 32/33 count, thus inhibiting this count in the night condition.

5.4   ASPECT COMPUTATION

This section describes the methods used in reducing PCM data from which the aspect or orientation of OSO-6 is determined.

Pitch angle reduction, spin rate calculations, frame counter operation, roll angle calculation, and spin axis and wheel azimuth position are covered. Of these, only the pitch angle and frame count are obtained directly from the OSO-G Data Conversion Tables (BBRC Doc TN68-16). The other parameters require calculations. The methods for these calculations follow.

5.4.1   How Wheel Spin Rate Is Calculated

The spin rate of OSO-6 is computed by two independent means. The first uses the data obtained from ASC 1, channel 12; the second uses DSM data. Of the two methods, the latter is the most accurate and can be used to compute the night spin rate while the first cannot.

● *Calculating Spin Rate From ASC Data*

When calculating wheel spin rate using ASC data, the telemetry count received on wheel ASC channel W-12 must be converted to integer steps from the data conversion tables (TN68-16). The 0 - 255 bit count corresponds to steps from 1 to 32. The step count advances one digit each wheel revolution and recycles to 1 after the count reaches 32.

In computing spin rate using this method the following procedure is used:

(1) Record "n" number of successive data readings from W-12.

NOTE:

The accuracy of spin rate calculation is directly proportional to the value of "n". The actual error term is:

$$\pm \frac{1}{15.36 \ (n - 1)}$$

(2) Convert the initial data reading and the final data reading to steps using nomograph W-12 in TN68-16.

(3) Examine the above data and determine the number of counter resets (a reset is indicated when a data reading is less than the data reading immediately prior to it).

(4) From the following formula calculate Δ steps (or revolutions):

$$\text{revolutions} = 32 \ (CR) + R_f - R_i$$

where

CR = number of counter resets

R_i = initial reading (step number)

R_f = final reading (step number)

(5) Calculate the spin rate (SR) by substituting the values for n and Δ steps into the following formula:

$$SR = \frac{\text{revolutions}}{15.36 \ (n - 1)}$$

where

n = number of data readings

C.4

5.4.2   Calculating Spin Rate from SORE Data

The following formula is used when computing the day wheel spin rate, or the approximate night spin rate*, using the data from DSM channels 12 and 13:

$$\text{Spin Rate (rps)} = \frac{N}{15.36 + 0.0025 \ [(12_2 + 4 \times 13_2) - (12_1 + 4 \times 13_1)]}$$

where

12_1, 13_1, and 12_2, 13_2 are consecutive readings of DSM channels 12 and 13.

N is an integer determined by dividing 15.36 by the approximate period of rotation (normally about 2 seconds). The larger whole integer quotient is used if the quantity $(12_1 + 4 \times 13_1)$ is smaller than $(12_2 + 4 \times 13_2)$. The smaller whole integer quotient is used if $(12_1 + 4 \times 13_1)$ is larger than $(12_2 + 4 \times 13_2)$.

NOTE:

The "15.36" in the equation is the recycle time of the DSM, and is the time between consecutive occurrences of words 12 and 13. The "0.0025" term is the period of the 400 bps clock obtained by dividing the 800 bps clock by 2.

---

*The computer software systems have provisions for accurately computing night-time spin rate. To do this, the drift in the geomagnetic field between DSM outputs is computed and accounted for in the spin rate computation. Because of the complexity of this operation, the method is not described here.

5.4.3   How Frame Counter is Used to Establish Time Reference

A reference frame count must be established and correlated with the corresponding universal time before roll aspect or wheel night azimuth position can be computed. Also, it must be known approximately when the frame counter will reset, because, with the 800 bits/sec. telemetry system, the frame counter resets twice in each orbit and, therefore, no one frame count number is unique for a particular orbit.

Channel 36 of the DSM contains the DSM frame count and steps once every 15.36 seconds which is the recycle time of the DSM. This channel indicates the frame count from 0 to 255 and resets every 65.536 minutes ($256 \times \frac{15.36}{60} = 65.536$).

The time (T_θ) at which the aspect data were taken must be known to determine the observatory's geographic position in the magnetic field. This time is calculated as follows:

$$T_\theta = U.T. \text{ (ref)} - [\#36 \text{ (ref)} - \#36_{\theta_m}] \ 15.36 \text{ sec.} - 12 \pm 1 \text{ sec.}$$

where

U.T. (ref) is the universal time of frame count reference word #36 (taken near the end of the given orbit by a STADAN station)

$\#36_{\theta_m}$ is the frame count word just preceding DSM words 32 and 33 that were used for the calculation of θ_m.

12 $\pm$1 sec. is the approximate time interval between the actual measurement of θ_m and the time of frame count word $\#36_\theta$.

Section 6

DYNAMICS

6.1 INTRODUCTION

After injection into orbit the OSO spacecraft will try to maintain, because of its gyroscopic properties, a constant orientation with respect to the celestial sphere. The spin rate will remain constant unless the spacecraft is acted upon by external torques. In the absence of external torques, the spacecraft spin axis needs only to be precessed enough to keep it normal to the earth-sun line. If the spin axis is in the ecliptic plane, this requires 360 degrees of correction a year; if the spin axis is normal to the ecliptic plane, no pitch correction is required.

However, both internal and external torques do act on OSO. They cause changes in both the position of the spin axis and in the spin rate of the wheel. Some of these torques also produce nutation which must be damped out. The following is a discussion of the sources of these internal/external torques, their influence on OSO, and how their effects are controlled.

6.2 ROTATIONAL DYNAMICS OF OSO (HOW OSO BEHAVES AS A GYRO)

The OSO behaves much like a simple, single-body gyroscope. Some of the similarities and differences between the OSO and a gyroscope are discussed in the paragraphs that follow.

6.2.1 The Basic Differences Between OSO and the Simple Gyroscope

The simple gyro and OSO are similar in that they both display "gyroscopic rigidity". This characteristic is due to the rotation of a body about an axis (called the spin axis) which

itself may or may not be rotating about some other axis. Because of this single mutual characteristic, the two bodies behave according to basic gyroscopic principles. There are, however, some significant differences that make the motion of OSO much more complicated than the simple gyro. The simplified analysis, then, does not accurately describe this complex motion. Some of the more important differences are:

(1) The simple gyro is usually considered to be a rigid body (i.e., not consisting of flexible parts); OSO is semi-rigid.

(2) The spin rate of the simple gyro is normally assumed to be thousands of times greater than any other angular rates (e.g., precession and nutation rates); with OSO these other angular rates are relatively large.

(3) The simple gyro consists of a single rotating body; OSO consists of two bodies one of which is constantly spinning (the wheel), while the other (the sail) is alternately being spun-up and de-spun as OSO makes transitions from orbit-day to orbit-night and vice versa. Also, the pointed instruments give the MOI of the sail a time-varying character.

(4) The simple gyro is acted upon only by external torques; OSO is acted upon by both external and internal torques. The internal torques come from the elevation and azimuth servos, the nutation damper, and other non-rigid components.

6.2.2   The Rotational Dynamics of OSO

The following is a discussion of dynamic properties of OSO:

● *Definition of Nutation, Precession and Wobble.*
  The OSO is a spin-stabilized satellite with a
  biaxial pointing control. The satellite
  "precesses" and "nutates" when external and
  interbody (internal) torques are applied and
  "wobbles" when the spinning part is not
  perfectly balanced. Both nutation and wobble
  disrupt spin axis inertial stability (with
  oscillatory periods of a few seconds), and
  contribute to short-term pointing instability
  (jitter) of the pointed instruments.

  The coning motion of a balanced, symmetrical
  satellite that is slightly disturbed from a
  condition of "pure spin" about its symmetry
  axis is referred to as "nutation". The term
  "precession" is used to describe the motion of
  the angular momentum vector during application
  of external torques. Both precession and
  nutation occur during application of external
  torques. Short-term interbody torques are
  produced only by the azimuth and elevation
  servos, which also effect the nutational motion
  of OSO.

  The term "wobble" is used to describe the "steady
  state" motion which results from imperfect
  balance of the wheel. Wobble and nutation
  may be present simultaneously.

● *Motion of OSO's Spin Axis During Pitch Precession.*
To adjust OSO pitch attitude, control torques
are applied about the X-axis (roll axis) by
means of gas jets. These torques cause precessional
motion about the Y-axis (pitch axis). The pitch
control circuits and gas system, described in
detail in Section 7, are designed to keep the
spin axis aligned to within 3 degrees of
perpendicular to the solar direction. The
spacecraft dynamics associated with this system
are discussed here. In this analysis of
pitch precession we will first present the
gyroscopic motion of a rigid, symmetrical OSO
being constantly torqued, the motions of the
same OSO after the external torquing has been
removed, and finally a non-symmetrical OSO being
torqued.

Because OSO's spin rate is much slower than that
assumed for the simple gyro, the basic equations
of motion during precession do not hold true.
To be specific, the spin rate of the simple gyro
is usually sufficient to essentially eliminate any
nutation while the spin axis is being precessed.
Thus, the spin vector $(\bar{S})$ is always coincident
with the angular momentum vector $(\bar{H})$ and,
therefore, the precessional motion of the spin
axis can be described by simple equations. With
OSO, however, the spin rate is low enough (about
0.5 rps) that $\bar{S}$ follows a cycloid path instead
of a straight line path as does $\bar{H}$. This cyclical
motion (called nutation) is depicted in Fig. 6-1*.

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* The spin axis in Fig. 6-1 is shown laying in the horizontal
plane so that an edge view could be obtained of the pitch
plane in order to illustrate the motion of $\bar{H}$ and $\bar{S}$ more clearly.
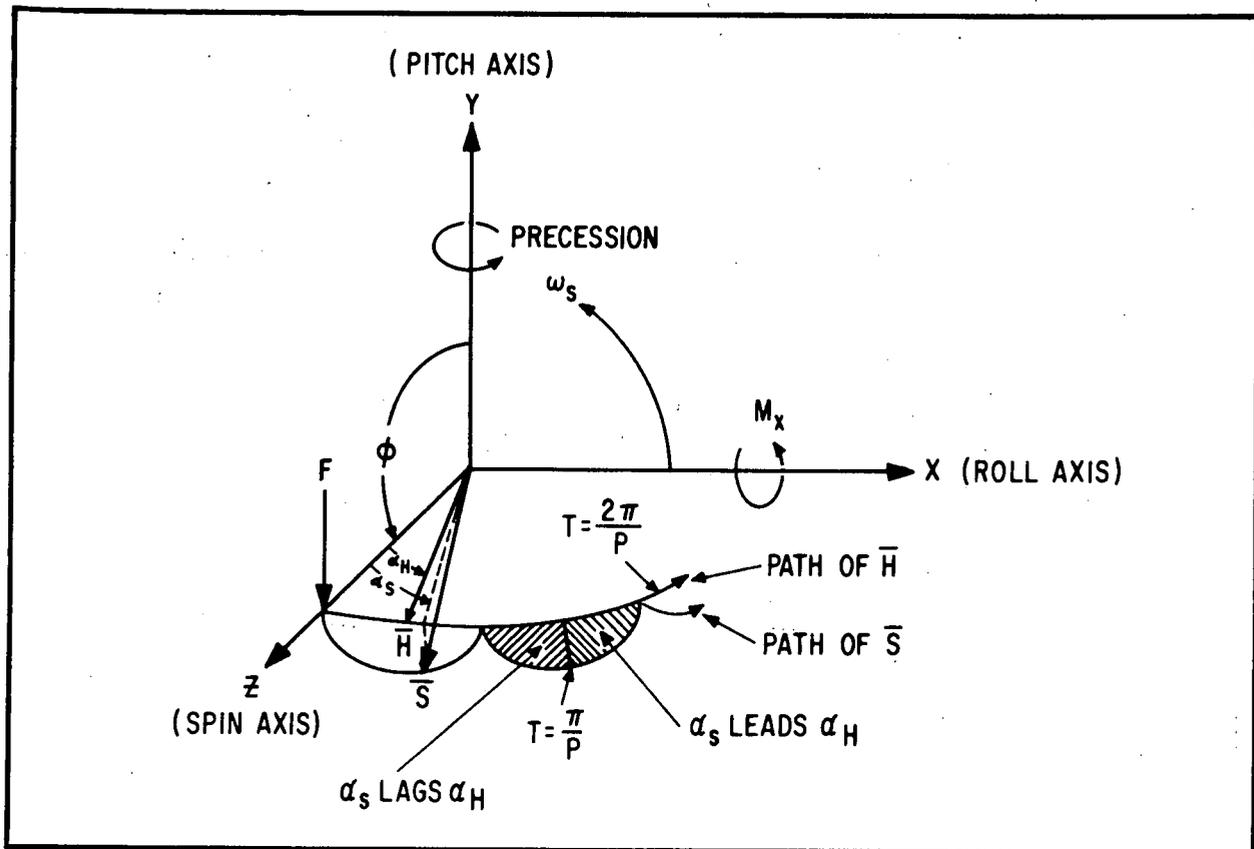
Fig. 6-1 Motion of $\overline{H}$ and $\overline{S}$ During Pitch Precession

In this figure, the x, y and z axes are shown as they would appear before the pitch thrust (F) is applied to the spin axis. After the pitch-down jet is activated, the product of F and its distance from the center of gravity of the spacecraft results in a torque (M_x) about the roll axis. Then by using the RH rule, we find that M_x actually causes the spin axis to rotate (precess) toward the roll axis. The paths of $\overline{H}$ and $\overline{S}$ during precession are shown as vectors with the precession angles in the XZ plane represented by α_H and α_S, respectively. Because the $\overline{H}$ vector moves only in the XZ plane, it always makes an angle of $\theta_H = \frac{\pi}{2}$ radians with the pitch axis. The $\overline{S}$ vector, however, moves

in both the XZ and YZ planes and, therefore, it makes an angle of $\frac{\pi}{2} + \theta_S$ with the pitch axis.

Figure 6-2 is a computer simulation of a 20-second pitch correction showing the nutation produced by such a maneuver. The decaying nature of θ_S is due to the action of the nutation damper.

- *Motion Equations of $\bar{H}$ and $\bar{S}$ During Precession (Symmetrical OSO).* The motion of $\bar{H}$ and $\bar{S}$ during precession can be described by a series of equations that are discussed below.

The position of $\bar{H}$ at any time during the precession is identical to that for the simple gyro, thus:

$$\alpha_H = \frac{M_x t}{I_s \omega_s} \quad \text{(Rad)} \qquad (6\text{-}1)$$

The rate at which $\bar{H}$ precesses in the XZ plane is:

$$\dot{\alpha}_H = \frac{M_x}{I_s \omega_s} \quad \text{(Rad/sec.)} \qquad (6\text{-}2)$$

The position of $\bar{H}$ in the YZ plane is always constant; thus θ_H is constant at $\frac{\pi}{2}$ radian = 90°

The position of $\bar{S}$ in the XZ plane at any time during precession is given by:

$$\alpha_S = \frac{M_x t}{I_s \omega_s} - \frac{M_x \, \text{Sin} \, Pt}{I_s \omega_s P} \quad \text{(Rad)} \qquad (6\text{-}3)$$

where "P" is the nutation frequency of the OSO and is given by:
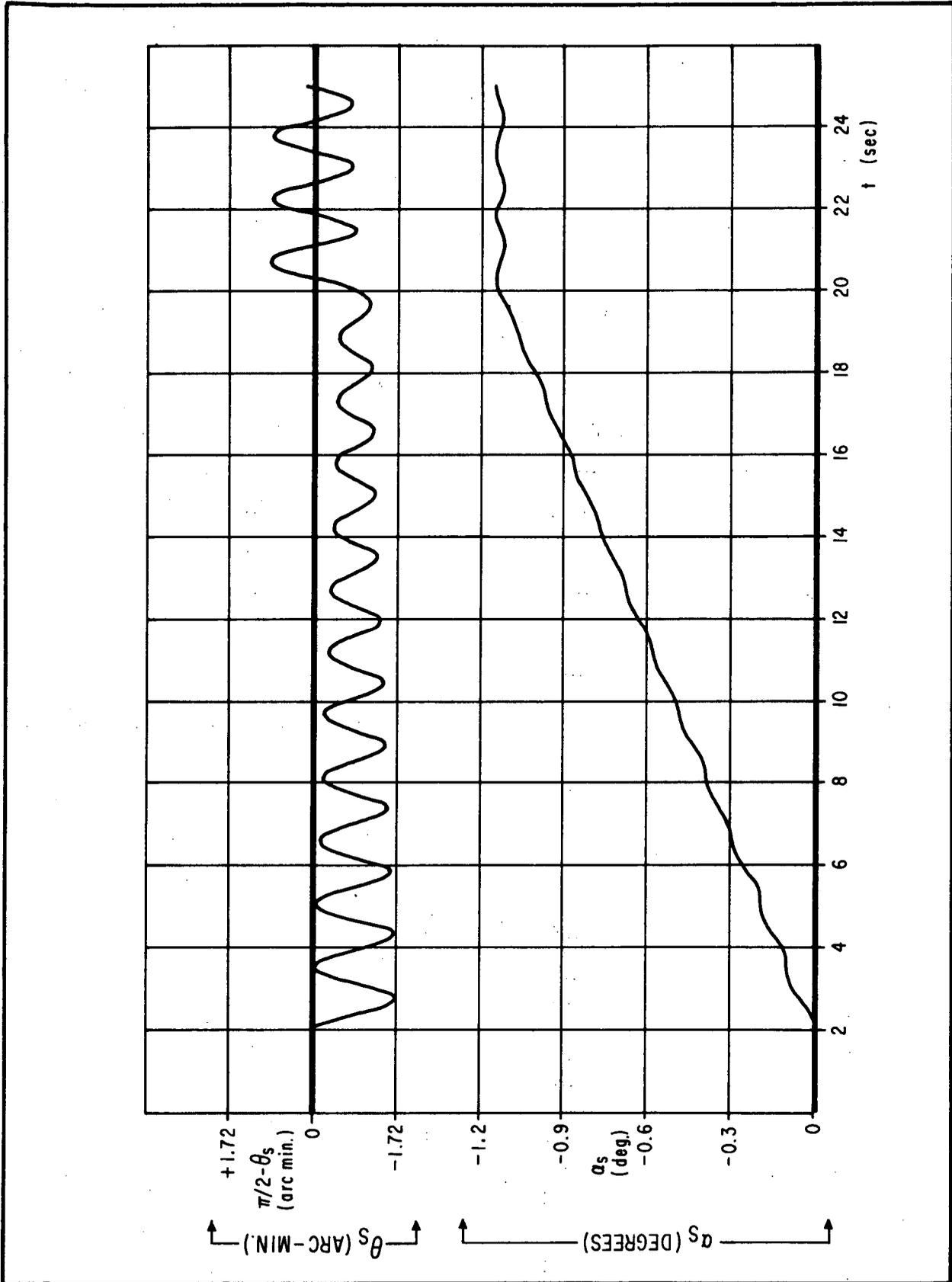
Fig. 6-2   Effect of Pitch Precession Torque (Oriented Sail -- With Nutation Damper)

$$P = \frac{I_s \omega_s}{I_1} \quad \text{(Rad/sec.)} \qquad (6-4)$$

where I_s is the spin MOI of the wheel and I_1 is the transverse MOI of the entire spacecraft.

Thus, it can be seen from Eq. 6-3 that $\bar{S}$ leads or lags $\bar{H}$ by the amount $\left[\dfrac{M_x \, \text{Sin} \, Pt}{I_s \omega_s P} \right]$ and in Fig. 6-1 we have shown that α_S always lags α_H during the first half-cycle of nutation, leads α_H during the last half cycle, and coincides with α_H at the end of each nutation cycle $(t = \frac{2\pi}{P})$.

The rate at which $\bar{S}$ moves in the XZ direction is given by:

$$\dot{\alpha}_S = \frac{M_x}{I_s \omega_s} - \frac{M_x \, \text{Cos} \, Pt}{I_s \omega_s} \quad \text{(Rad/sec.)} \quad (6-5)$$

Thus, $\dot{\alpha}_S = 0$ at the end of each nutation cycle $(t = \frac{2\pi}{P})$ and is equal to twice the α_H rate (i.e., $\dot{\alpha}_S = 2\dot{\alpha}_H$) at the half cycle point $(t = \frac{\pi}{P})$.

The position of $\bar{S}$ in the YZ direction is given by:

$$\theta_S = \frac{\pi}{2} + \frac{M_x(1 - \text{Cos} \, Pt)}{I_s \omega_s P} \quad \text{(Rad)} \quad (6-6)$$

From this it is seen that $\theta_S = \frac{\pi}{2}$ (the same as θ_H) at the end of each nutation period $(t = \frac{2\pi}{P})$ and equals $\frac{\pi}{2} + \frac{2M_x}{I_s \omega_s P}$ at the half cycle point $(t = \frac{\pi}{P})$. The term

$\dfrac{2M_x}{I_s \omega_s P}$ is, therefore, the maximum nutation amplitude of

$\overline{S}$ in the YZ plane and is denoted by:

$$A_\eta = \frac{2M_x}{I_s \omega_s P} \quad \text{(Rad)} \qquad (6\text{-}7)$$

The rate at which $\overline{S}$ moves in the YZ plane is given by:

$$\dot{\theta}_S = \frac{M_x \ \text{Sin Pt}}{I_s \omega_s} \quad \text{(Rad/sec)} \qquad (6\text{-}8)$$

Clearly at both $\dfrac{2\pi}{P}$ and $\dfrac{\pi}{P}$, $\dot{\theta}_S = 0$.

It can be shown that the amount of residual nutation at any time is given by the following equation:

$$r = \frac{1}{P} \sqrt{\dot{\alpha}_S^2 + \dot{\theta}_S^2} \quad \text{(Rad)} \qquad (6\text{-}9)$$

where "r" is the radius of the circular nutational motion (assuming that the transverse MOI's are equal) and $\dot{\alpha}_S$ and $\dot{\theta}_S$ are the velocity components of $\overline{S}$ at the time the torque is removed. Thus, the amount of residual nutation left after a pitch correction is determined by when the pitch torque is removed. This residual nutation is removed (dissipated) by the OSO nutation damper which will be discussed later in this section.

Motion Equations of $\overline{S}$ During Precession (Unsymmetrical Body). The equations of motion given in the previous sections apply to a single-body, symmetrical OSO under constant torque. However, the actual OSO normally consists of both a rotating "wheel" and a stationary "sail" and it is, therefore, necessary to determine

the effects of this stationary sail and the pointed
instrument assembly (PIA) on the motion of $\overline{S}$.

If the total* transverse MOI's of OSO are unequal (i.e.,
$I_x \neq I_y$) after the effects of the sail and PIA are
accounted for, then the OSO is said to be unsymmetrical
and both the frequency and amplitude of nutational
motion are different than for a symmetrical OSO. For
the unsymmetrical case, the nutational frequency
equation becomes:

$$P = \frac{I_s \omega_s}{\sqrt{I_x I_y}} \quad \text{(Rad/sec)} \tag{6-10}$$

However, the α_S equation remains the same as in the
symmetrical case but the θ_S equation changes to become:

$$\theta_S = \frac{\pi}{2} + \frac{M_x I_y (1 - \text{Cos } Pt)}{I_s^2 \omega_S^2} \tag{6-11}$$

Thus, for a composite OSO (wheel + sail + PIA) with
unequal total MOI's about the transverse axes, both
the frequency and amplitude of nutational motion are
different than for a symmetrical OSO.

---

* The "total" transverse MOI of OSO takes into account, in addition
  to the transverse MOI of the wheel, the MOI of the sail about the
  x and y axes and the MOI of the PIA about the x and y axes.
  Thus, I_x and I_y refer to the total (wheel + sail + PIA)
  transverse MOI of OSO about the x and y axes.

● *Residual Motion of $\bar{S}$ After Pitch Correction Has Been Completed.* It was shown in the preceding section that the angular momentum vector, $\bar{H}$, remains in the XZ plane ($\theta_H = \pi/2$) at all times during torquing. When torquing of the gyroscopic body is stopped, the residual motion is a circular motion of the spin vector about the angular momentum vector. From Eq. 6-6 it is seen that the amplitude of this motion is dependent upon the position of $\bar{S}$ on the cycloidal path at the time torquing is stopped. That is, a maximum residual amplitude of $2M_X/I_S\omega_S P$ will remain 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 Fig. 6-3.
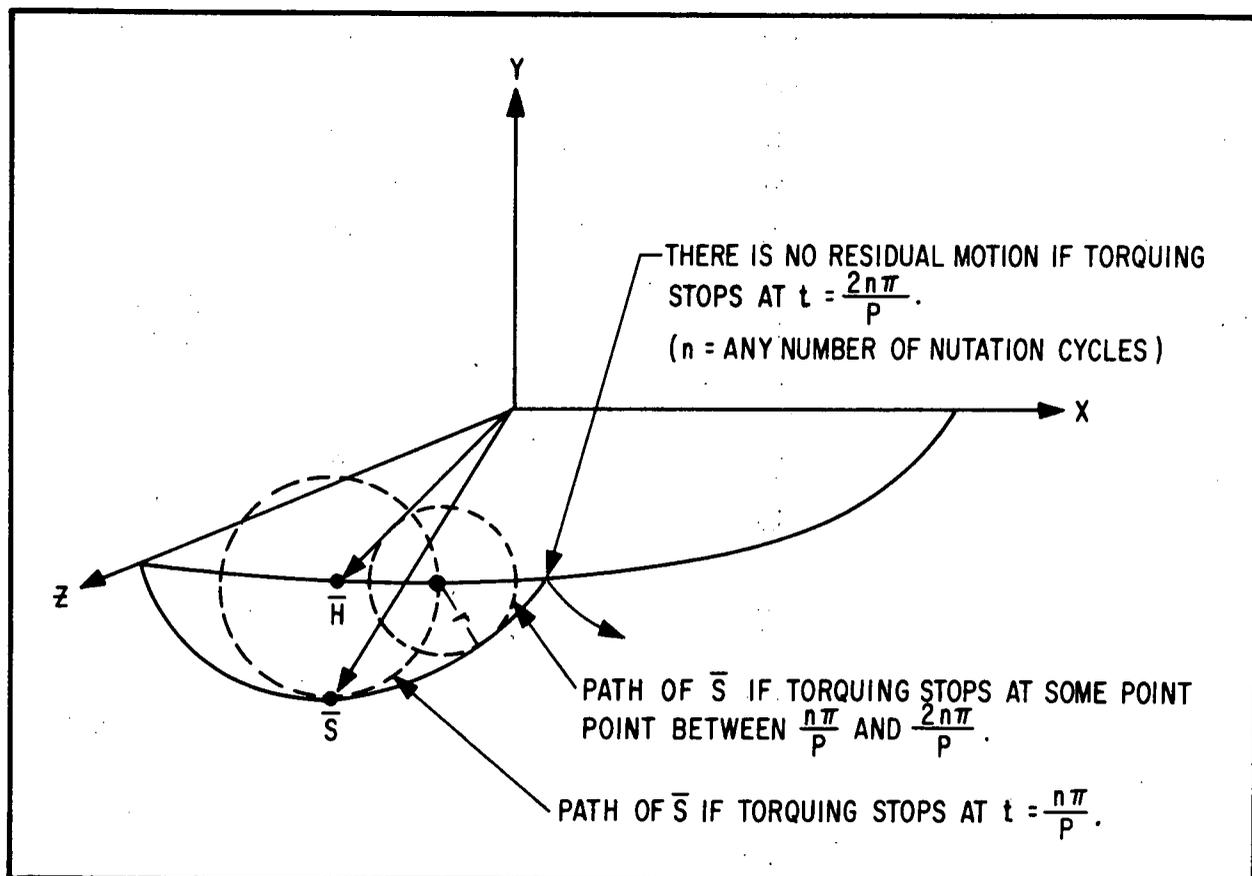
Fig. 6-3 Residual Nutation of $\bar{S}$ After Constant Torquing Stops

- *Values of Some OSO-6 Dynamic Parameters*

  Spin MOI of wheel only (orbit condition): I_s = 29.9 slug-ft^2

  Spin MOI of wheel and sail (orbit condition): I_s = 35.3 slug-ft^2

  Transverse MOI of wheel and sail (orbit condition): I_1 = 24.5 slug-ft^2

  MOI Ratio (orbit day condition): $\dfrac{I_s}{I_1}$ = 1.22

  Pitch Torque: M_x = 0.193 ft-lb.

  Nutation Frequency: $P = \dfrac{I_s \omega_s}{I_1}$ = 3.87 Rad/sec. (at ω_s = 30 rpm)

  Maximum Nutation Amplitude During Pitch Precession:

  $$A_\eta = \frac{2M_x}{I_s \omega_s P} = 1.05 \times 10^{-3} \text{ Rad}$$

  Angular Velocity of $\bar{H}$ During Pitch Precession:

  $$\dot{\alpha}_H = \frac{M_x}{I_s \omega_s} = 2.04 \times 10^{-3} \text{ Rad/sec.}$$

6.2.3 Interactions Between Pointing Control System and Nutation

The interactions between the OSO azimuth and elevation servos are much more complex, due to the short-duration impulses, than those for steady-state torquing (e.g., torquing by the pitch control system). Thus, the best way to illustrate these complex

interactions is to use the results of a computer analysis that simulated (after some simplification) these interactions. The details of this analysis are reported in Final Report on OSO Dynamic Analysis, (F66-06). In this computer analysis, a set of seven Lagrange equations of motion were used to simulate the effect of nutational motion on the azimuth and elevation servos, the effect of servo operation on nutation, and the effect of the nutation damper on nutation. Some conclusions obtained from this analysis are summarized in the following paragraphs:

- *Effects of the Azimuth and Elevation Servos on Nutation*
  The azimuth and elevation control servos form a biaxial pointing control system. When the sail is accelerated in azimuth by the azimuth servo, the reaction torque (acting about the spin axis) causes only a small change in the wheel spin rate (ω). This small change in ω indirectly influences nutation in a small way due to the cross-talk effects on the elevation pointing error when the PIA is moved in azimuth. On the other hand, when the PIA is accelerated in elevation by the elevation servo, the reaction torque is in the "cross-spin" direction (perpendicular to the spin axis) and thus severely affects the nutational motion of the spin axis.

  During the normal pointing mode, i.e., after coarse acquisition has occurred, any spin axis nutation causes a periodic disturbing torque to be applied to the PIA about the elevation bearing axis. This disturbance, caused by friction (sometimes called "stiction") in the elevation bearings, causes periodic changes in the elevation pointing error. The elevation servo generates corrective torques proportional to the magnitude and rate of change of the elevation pointing error.

A computer run (see Fig. 6-4) simulating the interactions during the normal pointing mode supports the important conclusion that the elevation servo response tends to increase nutation. In Fig. 6-4, the time histories of elevation pointing error (ERREL), nutation amplitude (α), and elevation torque (ESC) are used to show the inter- action between these parameters when an initial nutation amplitude of about 40 arc seconds is assumed. In this run the nutation damper was not simulated because its damping influence would have swamped out the effect of the elevation servo on nutation.

As can be seen in Fig. 6-4a, the elevation pointing error periodically moves between positive and negative values (given by the ratio between the servo gain and friction in the bearings). Also, the nutation amplitude (Fig. 6-4b) is seen to increase by about 2 arc seconds during each nutation cycle. This growth can be accounted for quantitatively by comparing the sense of the net elevation torque (ESC) with α. That is, positive torque causes positive acceleration in α.

● *Interaction Between the Elevation Servo and Nutation During Normal Acquisition.* When the OSO passes into the shadow of the earth, the solar reference is lost and azimuth bearing friction causes the upper structure to spin up until the wheel and upper structure are spinning at the same rate. The sun is reacquired every orbit as both the azimuth and elevation servos develop drive signals in order to point the sail and the PIA in the solar direction. Fig. 6-5 is a computer simulation of a normal acquisition (with the nutation damper effects omitted) in which the PIA is initially displaced about 52 arc minutes from the solar direction. This amount of displacement is normal when it is considered that the
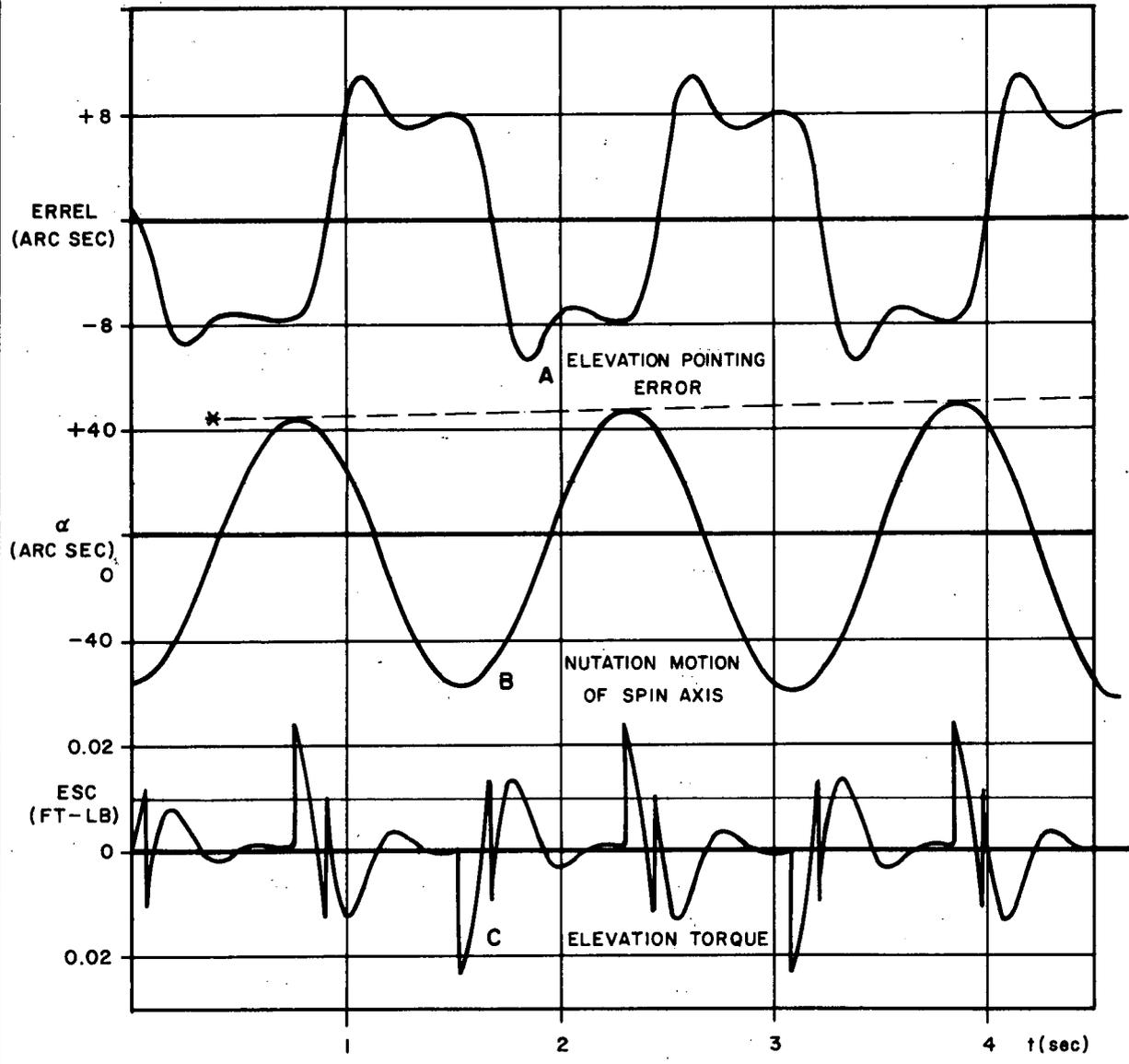
Fig. 6-4 Interaction of Elevation Servo with Nutation
(Normal Pointing Operation; No Nutation Damper)
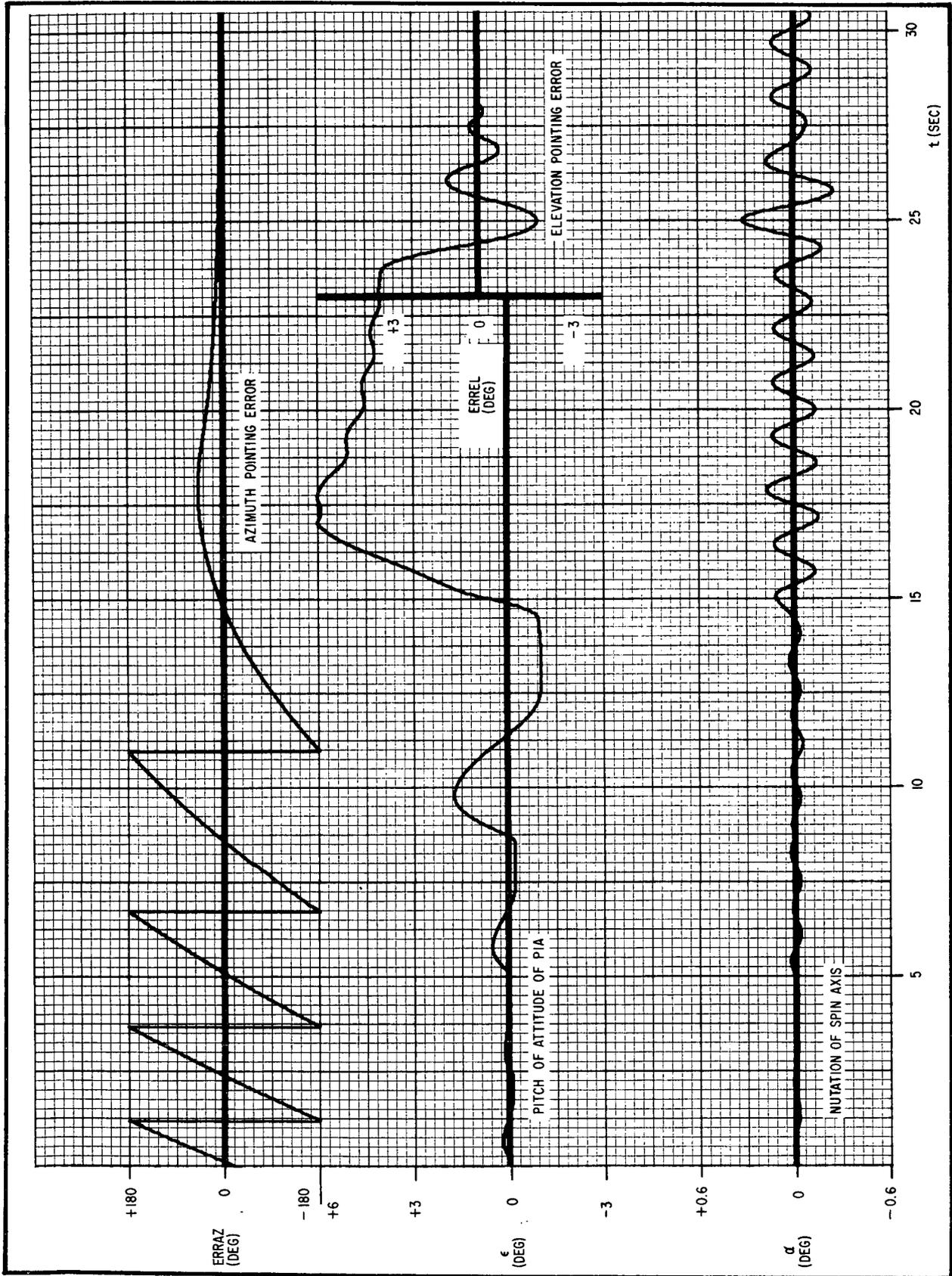
Fig. 6-5 Nutation Generated During Normal Solar Acquisition (no Nutation Damper)

PIA always seeks its dynamic balance position at night. The four parameters plotted in Fig. 6-5 are: azimuth pointing error (ERRAZ); the pitch attitude of the PIA (ϵ) with respect to the wheel plane; elevation pointing error (ERREL) with respect to the sun; and nutation amplitude of the spin axis (α). The azimuth component of pointing error (ERRAZ) is also shown so that the reader can follow the coarse azimuth acquisition and thus know when the sun is in the field of view of the elevation fine control eyes.

During the early phase of solar acquisition, when the sail has a high spin rate, the elevation servo produces torque for brief intervals when the sun is in the field of view of the elevation fine control eyes. At first this torque has little effect on nutation since it lasts only a fraction of a second (note the small perturbations in the α trace during the first 10 to 15 seconds). These short bursts of drive torque do, however, ultimately cause the PIA to begin oscillating about the elevation axis (note the ϵ trace during first 15 seconds) as the bursts get longer.

The centrifugal force due to the sail high spin rate during the early phases at acquisition acts as a spring and thus prevents the PIA from being displaced very much when the elevation torque bursts first occur. As the sail spin rate decreases, the centrifugal force decreases and the amplitude of PIA motion begins to increase (this can be noticed at about t = 17 seconds).

As the sail spin rate approaches zero, the elevation servo begins to null the elevation pointing error. The torque produced by the elevation servo during this nulling phase fluctuates at a frequency determined by the response characteristics of the servo. The response characteristics of the servo are a function of:

(1) The magnitude of the initial error

(2) The nonlinear response characteristics of the elevation fine eyes

(3) The torque capability of the elevation torque motor

(4) The servo lead time constant

The phase relationship between servo torque and spin axis nutation is a major factor in determining the nutation amplitude generated as the error is nulled. That is, large nutation amplitudes are generated when the response frequency of the servo is near the nutation frequency and in phase with it (this can be seen at t = 25 seconds in Fig. 6-5).

• *Interaction Between Elevation Servo and Nutation when the PIA Hits the Elevation Stops During Acquisition.* The PIA can be driven into the elevation stops* if either of two conditions exist during acquisition:

(1) The initial pitch error is greater than about two degrees and thus one of the stops is closer to the solar direction than the other.

(2) The PIA is driven hard by a relatively long torque burst as the sail swings past the sun in azimuth, the PIA will coast into one of the elevation stops due to its momentum.

---

* The elevation stops limit the travel of the PIA to about $\pm 6°$ relative to the solar direction when the pitch error is zero.

In Fig. 6-6, good examples of both of these situations are shown and we have included the elevation torque in this figure to simplify the interpretation of the interactions. A good example of condition (2) above, is shown at t = 15 seconds. Here, the PIA is driven in the positive direction by a relatively long burst of elevation torque. The sail then passes out of view of the elevation control eyes just as the PIA acquires a high angular rate about the elevation axis. At this time the restoring centrifugal force is low due to the sail's low spin rate and the only retarding force is the friction in the elevation bearings. This friction, however, is not enough to prevent the PIA from striking the upper elevation stop (at about t = 16 sec.).

An example of condition (1) above, is shown at t = 25 seconds. Here, the PIA is driven against the upper stop because this stop is closer to the solar null than the lower stop (i.e., the upper stop is only three degrees from the solar null while the lower stop is about nine degrees from the null).

The violent solar acquisition just described and shown in Fig. 6-6 is a rare occurrence in the actual operation of OSO because the PIA usually stays within two degrees of the solar direction during orbit-night. However, it is possible for this kind of acquisition to occur if the pitch attitude is near one of the $\pm3°$ pitch limits and if the dynamic balance point of the PIA (at night) is of the opposite sense.

● *How the Nutation Damper Dissipates Nutation*. The damper used on the OSO consists of a bob of dense material attached to the end of a length of piano wire. This bob is immersed in a viscous fluid which applies retarding

Fig. 6-6 Nutation Generated During Solar Acquisition when the PIA Strikes the Stops (no Nutation Damper)
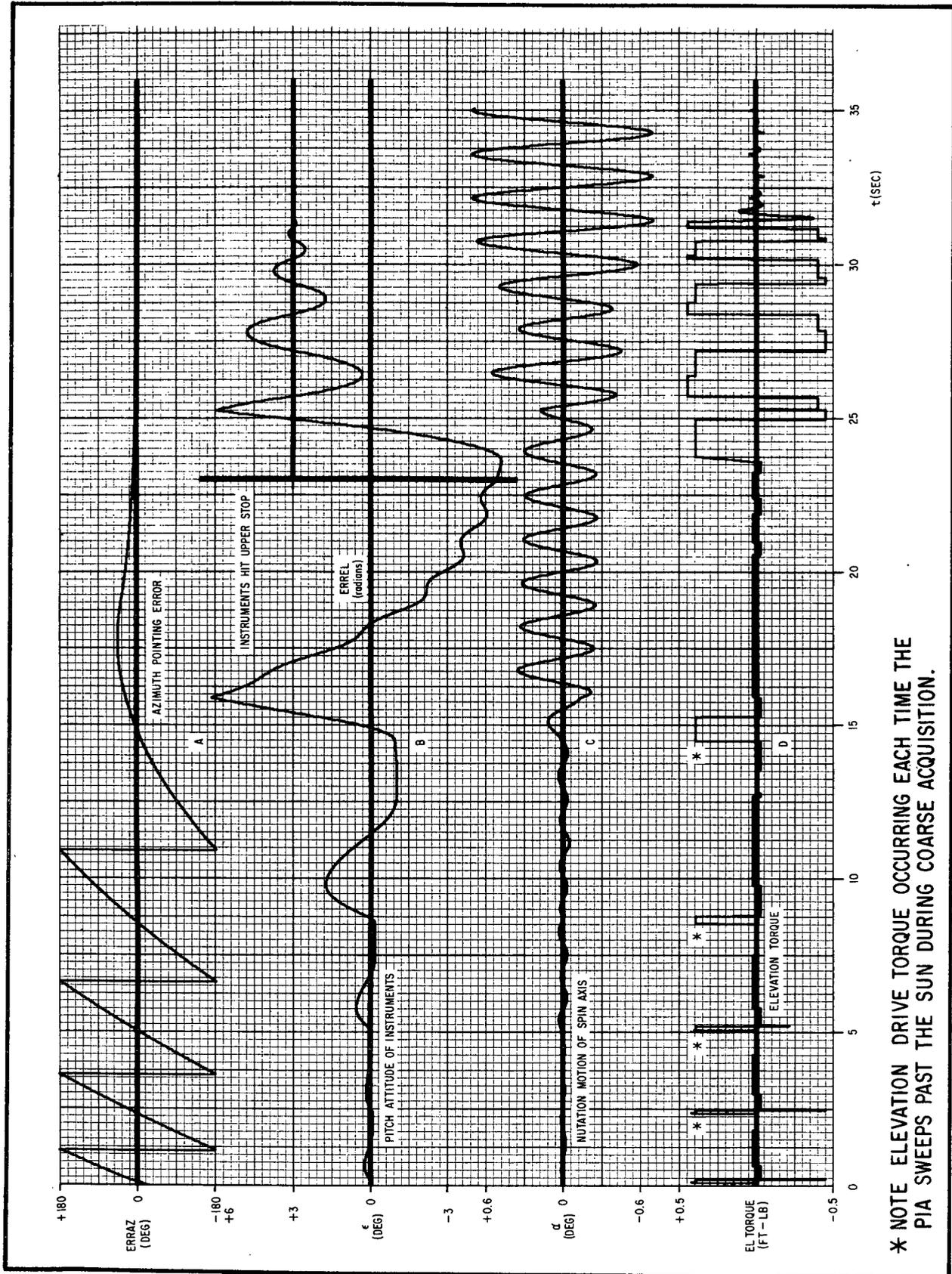
* NOTE ELEVATION DRIVE TORQUE OCCURRING EACH TIME THE PIA SWEEPS PAST THE SUN DURING COARSE ACQUISITION.

forces to the bob. The bob and damping fluid are
contained in a compact case which restricts the amplitude
of bob motion. In the unrestricted range of bob motion,
this damper dissipates the kinetic energy associated with
nutation by converting this energy into heat.

The total rotational kinetic energy of a nutating OSO
can be expressed as the vector sum of the spin component
of kinetic energy ($T_s = 1/2\ I_s \omega_s^2$) and the cross-spin
kinetic energy ($T_1 = 1/2\ I_1\ \omega_1^2$). Thus, the total kinetic
energy is:

$$T = 1/2\ I_s\omega_s^2 + 1/2\ I_1\omega_1^2 \qquad (6\text{-}12)$$

The cross-spin kinetic energy is associated only with the
coning motion (nutation) of the spin axis about the spin
angular momentum vector ($\overline{H}_s$), and must decay to zero if
the nutation cone is to collapse onto the angular momentum
vector. It can also be said that the total angular momentum
($\overline{H}*$) is the sun of its spin and cross-spin components, or

$$\overline{H}* = \overline{H}_s + \overline{H}_1 \qquad (6\text{-}13)$$

If it is assumed that the nutation cone collapses strictly
as the result of dissipation of energy by the nutation
damper, then the total kinetic energy of the spacecraft
must change; but its angular momentum, however, remains
constant (since no external torque is involved). Because
the angular momentum vector is constant, the final spin rate
of the wheel (after dissipating the nutation energy) can be
calculated as follows:

$$H*^2 = (I_s\omega_s)^2 + (I_1\omega_1)^2 \text{ (from Pythagoras' theorem}$$
for right triangles)

The $(I_1\omega)^2$ term in the above equation is reduced to zero as the cross-spin component is dissipated by the nutation damper and thus eventually:

$$H*^2 = (I_s\omega_s)^2 \qquad (6\text{-}14)$$

However, since $H*$ and I_s are both constants, ω_s must increase and the final spin rate (ω_s^*) can be calculated as follows:

$$H*^2 = (I_s\omega_s^*)^2 = (I_s\omega_s)^2 + (I_1\omega_1)^2 \qquad (6\text{-}15)$$

$$\text{or} \quad \omega_s^{*2} = \omega_s^2 + \left(\frac{I_1}{I_s}\right)^2 \omega_1^2$$

The final kinetic energy of OSO after dissipation of the nutation energy is

$$T* = T_s^* = 1/2\ I_s\omega_s^{*2} = 1/2\ I_s\omega_s^2$$
$$+ 1/2\left(\frac{I_1}{I_s}\right)^2 \omega_1^2 \qquad (6\text{-}16)$$

The net change in kinetic energy (ΔT) during collapse of the nutation cone is the difference between the kinetic energy expressions in Eqs. 6-12 and 6-16, thus

$$\Delta T = T - T* = 1/2 I_1 \left(1 - \frac{I_1}{I_s}\right) \omega_1^2 \qquad (6\text{-}17)$$

$$\text{or} \quad \Delta T = \left(1 - \frac{I_1}{I_s}\right) T_1$$

The conclusions that can be drawn from the above analysis are that:

- H_1 and T_1 have decayed to zero

- T_s has grown to T_s^*

- The total kinetic energy of OSO must decrease
  if $I_s > I_1$.

The best explanation for why the wheel spin rate increased
slightly when the nutational energy is dissipated is by
what is called "the modified energy-sink argument".
This argument accounts for the increase in spin kinetic
energy by assuming that the cross-spin energy is
dissipated directly by the nutation damper, and that the
azimuth servo adds the increase in the spin kinetic energy
(this energy is actually supplied by the solar array).

- *OSO Nutation Damper Performance.* The OSO nutation damper
  (with a seismic mass of 1/600 of the spacecraft mass)
  causes a nutation amplitude of less than 1/2 degree of
  decay to about 1/3 of its initial value in 15 seconds
  at the nominal spin rate (larger amplitudes force the
  damper bob against the case). A time interval of 15
  seconds corresponds to 10 nutation cycles or about 7
  revolutions of the wheel.

Section 7
CONTROL SYSTEMS FUNCTIONAL DESCRIPTION

7.1   INTRODUCTION

This section presents detailed functional descriptions of the five OSO-6 control systems. General descriptions of these systems and their components are presented in Section 1. The five basic control systems are described briefly below:

(1)   <u>Pitch control system</u> - Provides automatic pitch control of the observatory by maintaining the spin axis within 4* degrees of normal to the solar vector; automatically switches to manual control mode in case of malfunction; and provides manual control of pitch attitude by ground station command when necessary. A magnetic bias coil (pitch coil) was used to augment the pneumatic control system. The pitch coil provided a means by which the magnetic dipole moment of the spacecraft could be varied by ground commands.

(2)   <u>Spin control system</u> - Provides automatic spin-rate control of wheel between the nominal limits of 0.45 to 0.66 rps; automatically switches to manual-control mode in case of a malfunction in which the spin rate goes outside the limits of 0.39 to 0.70 rps; and provides manual spin-rate control by ground station command when necessary.

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*This is the maximum value allowed by the OSO model specification. The nominal is closer to ±3 degrees.

(3) <u>Pointing control system</u> - Provides automatic solar acquisition of sail each orbit morning and pointing within 1 arc minute of center of sun. Alternate pointing control modes include:

(a) Large Raster -- a 64-line sun-centered raster covering 46 x 46 arc minutes;

(b) Small Raster -- a 16-line raster covering 7.0 arc minutes in azimuth, and 7.5 arc minutes in elevation, centered on any point in the offset grid; and

(c) Offset Grid -- an ability to point at or small raster about any point in a 128 point by 128 point grid 46 x 46 arc minutes square.

(4) <u>Launch sequence control system</u> - Automatically changes observatory from launch to orbit condition by initiating certain control functions at prescribed times; system operation is started by separation of the observatory from the launch vehicle.

(5) <u>Nutation control system</u> - Damps observatory nutational motion acquired during launch and while in orbit.

7.2   CONTROL SYSTEMS INTERACTION

Basically, the five OSO control systems function independently of each other but there are interactions that should be considered. Some of the more important of these interactions are described below.

7.2.1   Pitch Control System

Operation of this system is affected by the operation of three
other systems: First, the pitch control system is not activated
until the spin control system has turned day-power on. Also,
the spin control system controls wheel spin rate, which in turn
governs the angular momentum of the wheel and, therefore, the
rate of pitch drift. Although changes in angular momentum will
not affect the total amount of pitch gas consumed during the life
of the observatory, it will affect the frequency of the pitch
corrections. As in several earlier OSO missions, a magnetic bias
coil was used to reduce consumption of pitch control gas.

Second, a small amount of nutation is produced each time a pitch
correction is made and when the pointed instruments acquire each
morning. This motion will modulate the pitch angle unless it is
damped by the nutation damping system.

And third, the auto pitch limiter circuits are disabled until
coarse-fine switching has occurred. This function is provided
by the pointing control system.

7.2.2   Spin Control System

Operation of this system can be affected by the operation of
two other systems: the spin pulse amplifiers and the day-power
turn-on circuits in the spin control system are first activated
when orbit power is turned on by the launch-sequence-control system
during the launch phase. These circuits then provide day-power to
the auto-spin circuits so that the observatory can be despun from
the high spin rate acquired during third stage spinup. Also, the
launch sequence system causes the three extendible arms to deploy,
thereby decreasing the wheel spin rate from about 120 to 90 rpm
while providing a longer lever arm through which the spin jets
can act.

Second, the action of the pointing control system during normal orbital operation causes the sail to be despun each morning, and azimuth bearing friction causes the sail to spinup to the wheel spin rate each night. This spinning and despinning of the sail results in a corresponding transfer of sail angular momentum to and from the wheel and, therefore, changes the wheel spin rate. The change in wheel spin rate could cause the spin control system to operate if the spin rate is near the upper or lower limits.

Other factors that define the most desirable spin rate range are as follows:

(1) The data resolution of certain wheel experiments defines the lower and upper spin rate limits between which data can be most successfully recovered from these experiments. These limits are approximately 0.45 to 0.60 rps.

(2) The gyroscopic stability of the observatory is proportional to spin rate and thus defines the lowest spin rate at which stability will be acceptable. This limit is approximately 0.05 rps.

(3) The effective damping range of the tuned nutation damper defines the lower and upper spin rate limits between which the most efficient damping will occur. These limits are approximately 0.44 to 0.57 rps.

7.2.3   The Nutation Control System

Operation of this system can be affected by the operation of the spin control system (because the nutation damper is tuned for a specific spin rate) and by the operation of the launch sequence system. The latter provides the signal that uncages the nutation

damper bob during the launch phase. Also, the pitch and pointing control systems have a significant effect because they generate nutation-producing impulses during their operation.

7.2.4   The Pointing Control System

Operation of this system can be affected by the wheel spin rate and by the pitch attitude of the observatory, i.e., wheel spin rate affects the time required for despinning the sail and also, the pitch plane must be within approximately ±15 degrees (the vertical view angle of the coarse eyes) of normal to the solar vector before azimuth acquisition can be accomplished.

7.2.5   The Launch Sequence Control System

Operation of this system can be affected only by the separation of the observatory from the launch vehicle.

7.3   CONTROL SYSTEMS OPERATION DURING LAUNCH PHASE

Perhaps the most critical time in the operation of the OSO control systems is the 20-minute period between second-stage spinup and initial solar acquisition shortly after injection into orbit. During this period, the control systems must perform in unison to change the observatory from the launch to the orbit configuration as well as to accomplish initial solar acquisition. The following paragraphs describe some of the launch and early-orbit events.

7.3.1   Launch Requirements

The second-stage spin motors ignite shortly after the second-stage engine provides the final velocity increment to insert the spacecraft into a nearly circular orbit at an altitude of 350 nm. At injection, the spacecraft's spin axis should coincide with the

second-stage velocity vector which in turn should be parallel to the surface of the earth.

The injection point is determined primarily by the launch vehicle capability, the lift-off location, and the orbital parameters desired. Therefore, the injection point is fixed with respect to the earth's surface for the OSO mission.

For successful acquisition of the sun, the spacecraft must be injected into orbit through a time "launch window" determined by the relationship of the fixed injection point and the declination of the sun on the launch day. Proper injection timing is important because the sun must be in the field of view of both the spin eyes and the coarse control eyes mounted on the perimeter of the wheel and on the front and rear of the sail, respectively. The field of view for these sensors is approximately 15 degrees on either side of the wheel plane and 360 degrees in the wheel plane. This $\pm$15 degree field of view will allow about a $\pm$60 minute "launch window", but for reliability reasons this is reduced to about $\pm$20 minutes for OSO. The spacecraft events that occur during the launch and early orbit phases are discussed in the following paragraphs.

7.3.2   Spacecraft Control Systems Operation

At observatory separation, the spacecraft separation switches close, causing the spacecraft arms to deploy and starting the launch sequence timers. Arm deployment causes the spacecraft spin rate to decrease roughly twenty percent from the spin rate achieved during second-stage spinup. This is caused by the increase in wheel moment-of-inertia at arm deployment.

At T + 330 seconds, the spacecraft timers supply a signal that causes "orbit power" to be applied to the spacecraft. The turn-on of orbit power activates the spin eye amplifiers and level detector, and the "day-power" turn-on circuits. The application of

power to these circuits causes "day-power" to be applied to the spin control, pointing control and the pitch control systems.

When day-power is applied to the spin control system, an over-spin condition is immediately sensed. Subsequently, spin gas is released in bursts from the despin nozzles until the spacecraft wheel is despun to a spin rate of about 0.62 rps.

The spin eyes and their associated level-detector circuits have a high turn-on threshold before day-power is turned on and a much lower turn-on threshold after day-power is on. This prevents day-power "chattering" during orbit morning and evening because of varying light conditions.

With day-power turn-on, the pointing control system senses the position of the sun with the azimuth coarse eyes and produces a despin torque that despins the sail. The time required for initial sail despin is approximately 1-1/2 minutes, which is less than the approximately 3-1/2 minutes required by the spin control system to initially despin the wheel to 0.62 rps. Therefore, wheel despin is still progressing after sail despin has ended.

After the sail has been despun and is pointing within ± 3 degrees in azimuth, the target eye (after an eight-second delay) activates the fine pointing circuits, and the sail and pointed instruments are accurately aligned within 1 arc minute (in azimuth only) of the sun's center. Note that at this time, the elevation frame is still locked and, therefore, there can be no elevation pointing. This means that there will be high elevation motor current for approximately seven minutes prior to elevation frame unlatching which occurs at T + 800 seconds. Another possibility is that if the pitch angle was greater than the ± 6 degrees vertical view of the target eye, coarse-fine switching could not occur until the pitch control system corrected the pitch to within the range of the target eye.

With the application of day-power, the pitch control system is also activated and, for the first time, the spacecraft's pitch attitude is sensed. If this attitude is outside the range of about ± 3 degrees, this system automatically expels nitrogen gas from the appropriate pitch nozzle to correct the pitch angle to within 1 degree of the solar vector.

At T + 600 seconds, the launch sequence timers produce a signal that releases the nutation damper bob. The nutation damper then absorbs nutation energy originating at separation from the launch vehicle or from the ensuing spacecraft control systems operation; within a few minutes any cross-spin momentum of the spacecraft is reduced to a negligible value.

At T + 800 seconds, the timers produce a signal that unlocks the elevation frame supporting the pointed instruments. The pointed instruments then acquire the sun in elevation by action of the pointing control system. Also, a signal is applied to the auto pitch limiter (APL) and the auto spin limiter (ASL) enabling relays, which enable the auto pitch limiter and auto spin limiter circuits, respectively. At T + 1250 seconds, the timers repeat all of the launch sequence control signals to ensure the occurrence of the launch events.

7.4   CONTROL SYSTEMS IN-ORBIT OPERATION

Control system in-orbit operation is somewhat different from that at launch. As OSO comes into sunlight each orbit morning, the sail and wheel are spinning together because of azimuth bearing friction. Consequently, the spin rate has been reduced to about 87 percent of that for the previous day because of the transfer of a small amount of wheel momentum to the sail. The spin axis attitude at orbit-morning is very nearly the same as it was at the preceding orbit-evening because of the gyroscopic stability of the spacecraft, although it is possible that the very

slow pitch drift during the night might cause a pitch attitude in excess of the ±3 degrees correcting threshold. At the most, this drift should amount to no more than a few tenths of a degree.

Another spacecraft condition different from launch conditions is that both the nutation damper and the elevation frame are unlocked, the latter assuming a "night" position (because of dynamic unbalance that may exist in the pointed instruments) with the elevation angle being within a few degrees of zero. Also significant is the sun's position at sunrise with respect to OSO. The sun is viewed each orbit-morning as it rises from behind the limb of the earth through the diffused light band (air glow) along the horizon. The presence of this "extra" light, consisting of air glow and albedo, is favorable to battery charging but has little effect on solar acquisition because of the dominating sun intensity.

Control systems operation each orbit-morning will normally occur in the following sequence: The spin eyes sense the sun as it appears above the horizon and "day-power" is subsequently turned on. The low night spin rate is sensed by the spin control system before any appreciable increase in wheel spin rate can occur because of sail despin. Spin-up jets will be activated for a four-second burst if the spin rate is less than the low limit of 0.45 rps. Simultaneously, the coarse control eyes sense the position of the sun, and sail despin is initiated by the pointing control system. The time required for sail despin is approximately 60 seconds (this compares to 90 seconds for initial despin during launch phase caused by the high initial wheel spin rate). The target eye signal causes coarse-fine switching after despin terminates and the pointed instruments are aligned within 1 arc minute of the sun's center. Simultaneous with day-power turn-on, the pitch control system senses the pitch attitude of the spacecraft. If the pitch error exceeds the limits of about ±3 degrees, the appropriate pitch jets are activated to reduce the error to less than 1 degree. The nutation damper then damps any nutation produced in the above operations.

The above paragraphs have presented a general functional description of each primary control system on OSO during the launch and orbit phases. Each system will now be discussed in detail.

7.5   PITCH CONTROL SYSTEM FUNCTIONAL DESCRIPTION

7.5.1   General Description

The OSO-6 pneumatic pitch control system, like those of earlier OSO's, has two modes of operation: (1) the automatic mode in which the pitch attitude is automatically maintained nearly normal to the solar vector, and (2) the manual mode which is used as a backup in the event of auto-mode failure or when ground station control of the pitch attitude is desired. The auto-mode requires no attention from ground operations other than the routine reduction of the telemetered pitch data to determine the instantaneous pitch attitude and, subsequently, the pitch-drift rate of the spacecraft.

In addition to the pneumatic pitch control system, OSO-6 was equipped with a magnetic bias coil (pitch coil). This pitch coil could be controlled by ground commands to vary the magnetic dipole of the observatory, thereby augmenting the pneumatic system.

7.5.2   Automatic Pitch Control

In the auto-mode, four pitch-angle sensors are used; two for pitch-up control and two for pitch-down control (Figs. 7-1 and 7-2). In Fig. 7-2, these sensors are shown as A-down, B-down and A-up, B-up. The crosshatched areas indicate the turn-on thresholds of the sensors and their amplifiers. Zero pitch error is defined as the condition which exists when the spin axis is normal to the solar direction.

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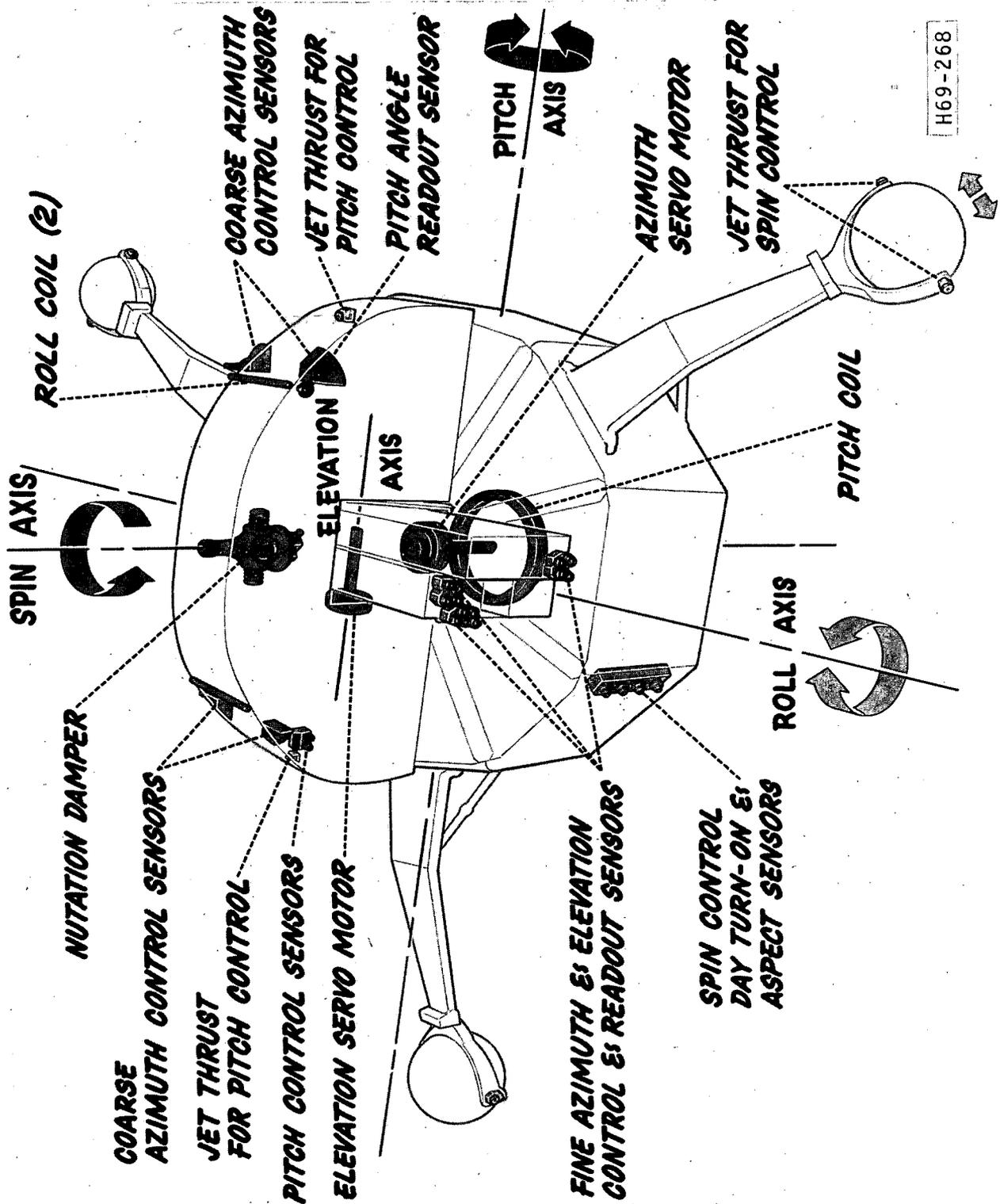
Fig. 7-1   Control Systems Components
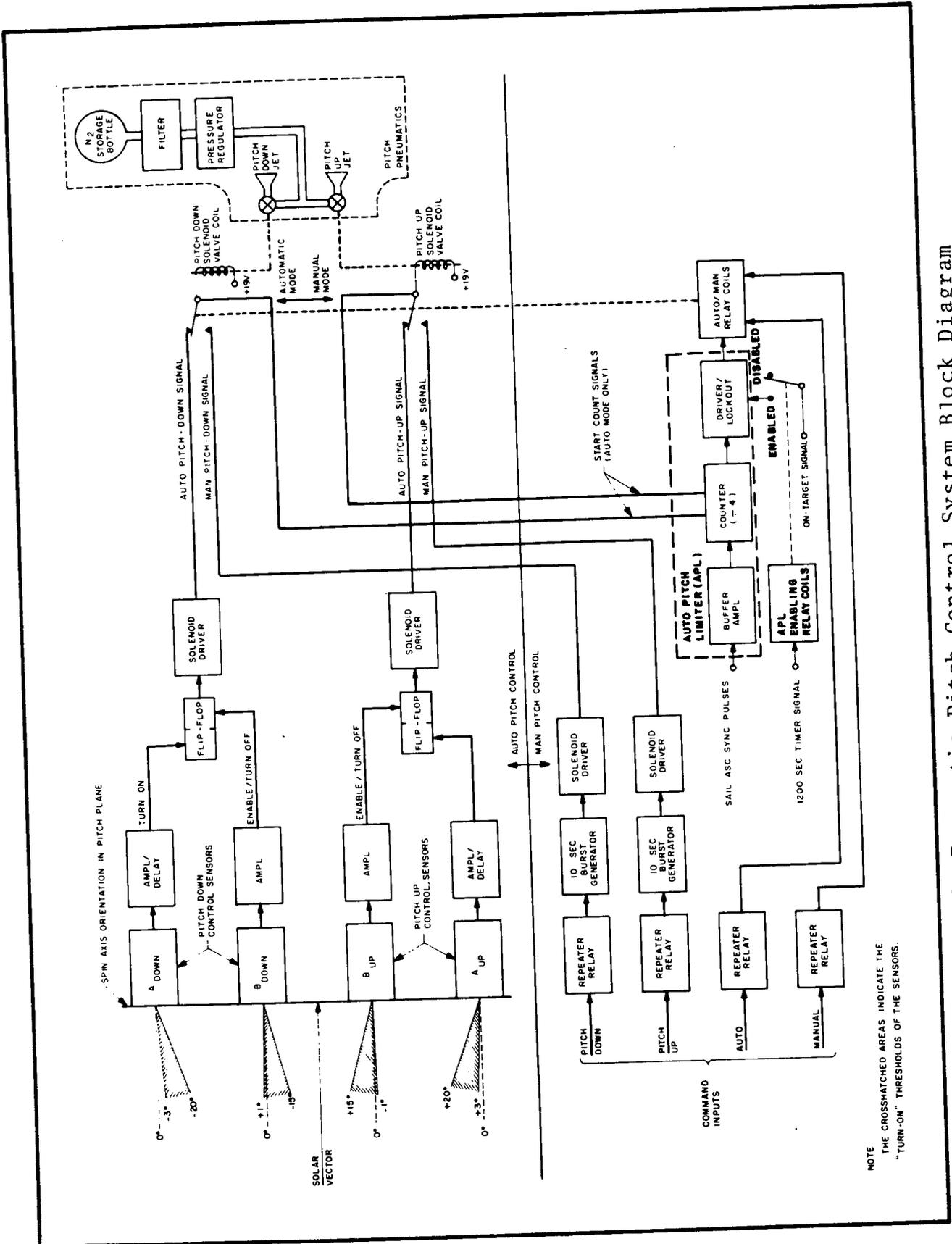
Fig. 7-2 Pneumatic Pitch Control System Block Diagram

First, consider a pitch-up drift of the spacecraft in which the spin axis rotates three degrees clockwise (pitch-up motion) starting from the zero position (Fig. 7-2). The pitch-down flip-flop is initially enabled because the B-down sensor is 1 degree past its turn-on threshold. When the pitch angle has advanced to -3 degrees, the A-down turn-on threshold is reached (assuming that the threshold is adjusted for 3 degrees) and after a delay of approximately 8 seconds, the pitch-down bi-stable flip-flop is set to the "on" state. The output of this flip-flop then drives the solenoid driver, which in turn energizes the pitch-down solenoid valve and pitch gas is exhausted through the pitch-down jet.

As the spacecraft starts to precess counterclockwise (pitch-down motion), the A-down sensor output immediately falls below the turn-on threshold, but the pitch-down flip-flop remains "on" until the B-down sensor output falls below its turn-on threshold. This occurs at +1 degree at which time the flip-flop is reset, and the pitch-down solenoid valve is subsequently de-energized.

The important point to remember in the above discussion is that the purpose of the "B" sensor is to enable and reset the pitch-down, bi-stable flip-flop. That is, when the "B" sensor is "on", the flip-flop is enabled and the "A" sensor can then set it to the "on" state when a pitch error of -3 degrees is sensed. Conversely, when the "B" sensor is "off", the flip-flop is reset; note that this condition normally occurs only at +1 degree because the extreme limit of -15 degrees for this sensor should never be reached.

Automatic correction of a 3 degree pitch-down error occurs in the same manner as does the error correction just presented, except that the pitch-up elements are employed.

The corrective pitch maneuvers just discussed for automatic
mode control, as well as those to be considered in pitch manual
mode operation below, are actually precessional motions of the
entire spacecraft and are governed by gyroscopic principles.
A slight amount of spin axis nutation is probable after a pitch
correction is made. This motion is damped by the nutation damper,
a control system assembly which is discussed later in this section.

7.5.3   Manual Pitch Control

The pitch manual mode of operation is automatically selected by
the automatic-pitch limiter (APL) circuits if either solenoid
valve is held open under automatic control for a period spanning
108 to 122 seconds. The automatic pitching period, which will cause
a switch to manual control, falls at random in the 108 to 122
second range because the switch-over occurs at the moment that
eight sail analog subcommutator frame sync pulses have been counted
from the time a valve is energized. This starting point may fall
anywhere within the subcommutator 15.36-second frame period.

Figure 7-2 also shows the pitch manual control functions. The
automatic switch-to-manual functions are shown on the lower
right. The sail ASC sync pulses, which occur every 15.36 seconds,
are fed to the APL circuits where they are buffered and fed to a
divide-by-eight counter. The "start count" signals to this counter
are supplied directly from each solenoid valve coil circuit whenever
an auto-pitch solenoid driver is turned on. The driver-lockout
circuit switches the auto-manual latching relay to the manual
position when the counter output provides a four-count switching
signal.

During the launch phase, the APL enabling relay supplies no
power to the driver-lockout circuit until both the 800 second launch
sequence timer signal and the on-target +19 v signal have occurred.
This lockout is necessary because orbital injection pitch error can

be as much as ±5 degrees for normal launch vehicle performance, and the automatic pitching time to correct this error would be a duration that could cause a switch to manual mode and therefore inhibit auto pitch operation. Automatic mode correction of abnormal pitch error at injection can be performed within the pitch control sensor's ±18 degree viewing angle during the lockout period.

Figure 7-2 also shows the command control functions of manual pitch operation. When manual control is selected, ground stations can command either pitch-down or pitch-up as required. An analog pitch monitor is used to detect the pitch angle and these data are supplied to the telemetry system. For every manual pitch command, the decoder command pulse operates the appropriate repeater relay, which in turn triggers a ten-second monostable multivibrator circuit (called a burst generator). The monostable output signal controls the appropriate pitch valve through a solenoid driver.

The pitch-manual and pitch-auto command inputs are direct switching commands for changing the pitch control mode to either manual or automatic mode. These commands also operate through repeater relays, which in turn directly control the auto-manual latching relay.

7.5.4   Magnetic Pitch, Spin, and Roll Control

Occasionally, the attitude or the spin rate of the observatory must be changed quickly. This takes large torques, so the gas jets must be used. But most of the time, only slow changes are needed; therefore, smaller torques are adequate and they can be produced by magnetic dipoles reacting against the earth's magnetic field. This magnetic torquing conserves gas.

A dipole along the spin axis can produce torque along the roll axis, causing pitch motion (because the observatory acts as a gyroscope) and along the pitch axis, causing roll motion. A dipole along the roll axis can produce torque along the pitch axis, causing roll motion, or along the spin axis, which changes the spin rate. Thus, dipoles along only two axes can produce control torques along all three. And with selective programming, the two dipoles usually can produce the needed torque in the desired direction without producing too much torque in some other direction.

Pitch Coil. The spin-axis dipole is produced by a coil of wire called the "pitch coil". It consists of copper magnet wire wound on an aluminum hoop. Its DC resistance is about 1000 ohms. This coil is mounted in the wheel hub with its axis along the spin axis.

Commands are used to turn the coil on or off, to select full or half power, and select positive or negative polarity. This means that any one of five spin-axis dipole moments can be chosen. The state of the coil is telemetered to the ground.

Roll Coils. The roll-axis dipole is produced by a pair of coils called "roll coils". The coils are mounted on the sail with their axes parallel to the roll axis. They are well separated so they won't interact much. Their mounting location is shown in Fig. 7-1. Different roll-axis dipole moments can be selected by command. The state of each coil is telemetered to the ground.

It would be possible to switch the coils several times each orbit to take advantage of the rapid changes in geomagnetic field direction. That would give strong control and allow fairly fast maneuvers, but it isn't necessary. Working with the orbit-average magnetic fields and switching every few days is good enough. The best coil settings for various times of the mission can be found using a computer program (COIL).

The effects that can be achieved with this system depend strongly on the attitude of the observatory, the position of the ascending node of the orbit, the season of the year, and the maneuver wanted. But they are generally large enough to be useful. This control system can hold the spin rate within its proper limits at all times, and can produce pitch and roll rates roughly on the order of one degree per day most of the time. Careful use of the coils can conserve the gas supply enough to last for several years.

7.6   SPIN CONTROL SYSTEM FUNCTIONAL DESCRIPTION

7.6.1   General

The spin control system, like pitch control, has a primary or automatic control mode of operation as well as a manual or backup control mode. The latter is normally used as a backup in the event of primary control failure, but may be used to manually adjust the spin rate to some specific value between the automatic control limits of 0.45 to 0.66 rps. A block diagram of the spin control system is shown in Fig. 7-3.

The concept of automatic spin control is to electronically measure the time required for each revolution of the wheel. When this time reaches a value of about 2.2 seconds (0.45 rps) as the spin rate drifts downward, a four-second burst of nitrogen gas is expelled from the spin-up nozzles, thus increasing the spin rate an increment of about 0.0245 rps. Similarily, when the time for a revolution reaches a value of about 1.52 seconds (0.66 rps) as the spin rate drifts upward, a four-second burst of gas is expelled from the spin-down nozzles thus decreasing the spin rate an increment of about 0.0245 rps.
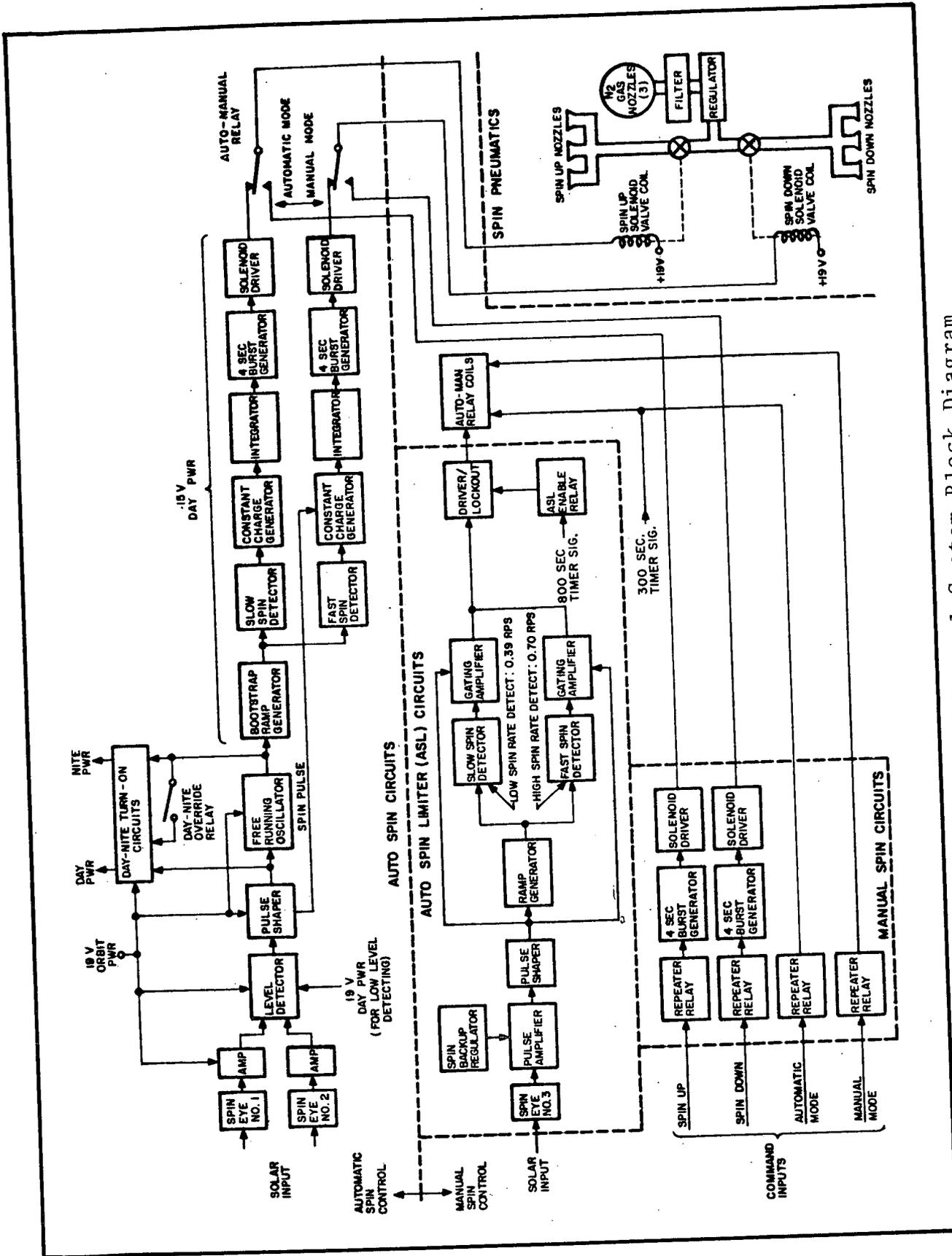
Fig. 7-3 Spin Control System Block Diagram

The manual mode is automatically selected electronically when
the time for one revolution falls outside the nominal range of
1.43 seconds (0.70 rps) to 2.56 seconds (0.39 rps). When either
of these limits is reached, a failure in the automatic spin
control circuits or a stuck solenoid valve is indicated. Manual
control can also be selected by ground station command. When
the spin rate is corrected manually, a four-second burst of gas
is expelled for each spin-up or spin-down command received by
the spacecraft.

7.6.2   Automatic Spin Control

Spin Eyes and Amplifiers. The auto-spin control mode uses two
identical and redundant solar sensors (spin eyes Nos. 1 and 2),
mounted on the perimeter of the spacecraft wheel. As these spin
eyes sweep past the sun, they each produce a pseudo-squarewave
current pulse (spin pulse) of about 20 milliseconds duration.
These two pulses are fed to two current-to-voltage amplifiers
where they are amplified and mixed before being fed to a level
detector.

Level Detector. The level detector has two modes of operation,
i.e., (1) high-level (low-sensitivity) detecting, and (2) low-
level (high-sensitivity) detecting. The high-level mode is
used during orbit morning when the spin eyes first begin to see
the sun. As the sun rises above the limb of the earth, the
light intensity varies because of albedo effects. When the
instantaneous output level of the spin eyes is sufficient to
overcome the high-level detecting threshold of the level detector,
a pulse is fed through the pulse shaper circuit and on to the
day-night turn-on circuits. These circuits then cause day-power
to be applied to the spacecraft during this upward excursion
of light intensity. One destination of this day-power is the
spin box 15-volt regulator. The output of this regulator is
then fed back to the level detector bias circuit and causes the

detector to operate in the low-level detecting mode. This action effectively "latches" the level detector on, once its high-detecting threshold has been overcome and any varying light levels are thereby prevented from causing intermittent application (chattering) of day-power.

Pulse Shaper. As previously mentioned, the output of the level detector is fed to a pulse shaper circuit. Here the spin pulses, which are roughly rectangular, are inverted twice and, in the process, become more square. After the first inversion, the positive-going trailing edge (originally a negative-going trailing edge) is used to turn day power on (this will be discussed in a later paragraph) and also to synchronize a free-running oscillator. After the second inversion, the positive-going leading edge is used as gating pulses in the constant charge generator in the overspin loop. The importance of this double inverting is that the free running oscillator is synchronized by the trailing edge of the spin pulses and the inhibitor constant charge generator circuit is gated by the leading edge.

Free-running Oscillator. The free-running oscillator has a free period of about 2.9 seconds which corresponds to 0.345 rps. This rate is much slower that the normal wheel spin rate expected (0.45 to 0.66 rps) and, therefore, the spin pulses will synchronize and simultaneously increase the oscillator's free-running frequency to match the spin rate. The oscillator output is fed to the day turn-on circuits and to a bootstrap ramp generator.

Bootstrap Ramp Generator. The bootstrap ramp generator develops a sawtooth voltage waveform that increases linearly with time. The period of this sawtooth (or ramp) is identical to that of the spinning wheel since the ramp generator is controlled by the oscillator, which in turn is synchronized by the trailing edge of the spin pulses. The peak voltage reached by the ramp

is then directly proportional to the wheel period. That is, the higher the value of this peak voltage, the slower the spin rate.

This ramp voltage is fed to two detector circuits - an overspin detector and an underspin detector.

Overspin Detector. The overspin detector functions as an under-voltage detector. As the ramp (Fig. 7-4a) increases from zero volts up to the level at which it is reset by the trailing edge of the spin pulse (Fig. 7-4b), the overspin detector produces a positive output pulse (Fig. 7-4c) that starts at a preset point on the ramp (called the overspin threshold) and ends when the ramp is reset. The overspin threshold is preset for a nominal wheel spin period of 1.52 seconds (0.66 rps). This means that for spin rates above 0.66 rps (Fig. 7-4e), the output from the overspin detector is at ground level. (Fig. 7-4g).

When the detector output is positive, a storage capacitor is charged and then slowly discharged. This produces the decay function shown in the detector output (Fig. 7-4c). When the output from the detector is zero, the charge on the storage capacitor drops to zero as shown in Fig. 7-4g. This output from the overspin detector is used as an input to the constant charge generator.

Overspin Constant Charge Generator. The purpose of the constant charge generator is to generate a fixed amount of electrical charge (coulombs) each time an overspin condition is sensed by the overspin detector.

When the overspin detector output is positive (corresponding to a normal spin rate condition, Fig. 7-4c), there is no output from the constant charge generator (Fig. 7-4d). However, when the detector output is zero (indicating an overspin condition, Fig. 7-4g), the constant charge generator generates a pulse of current
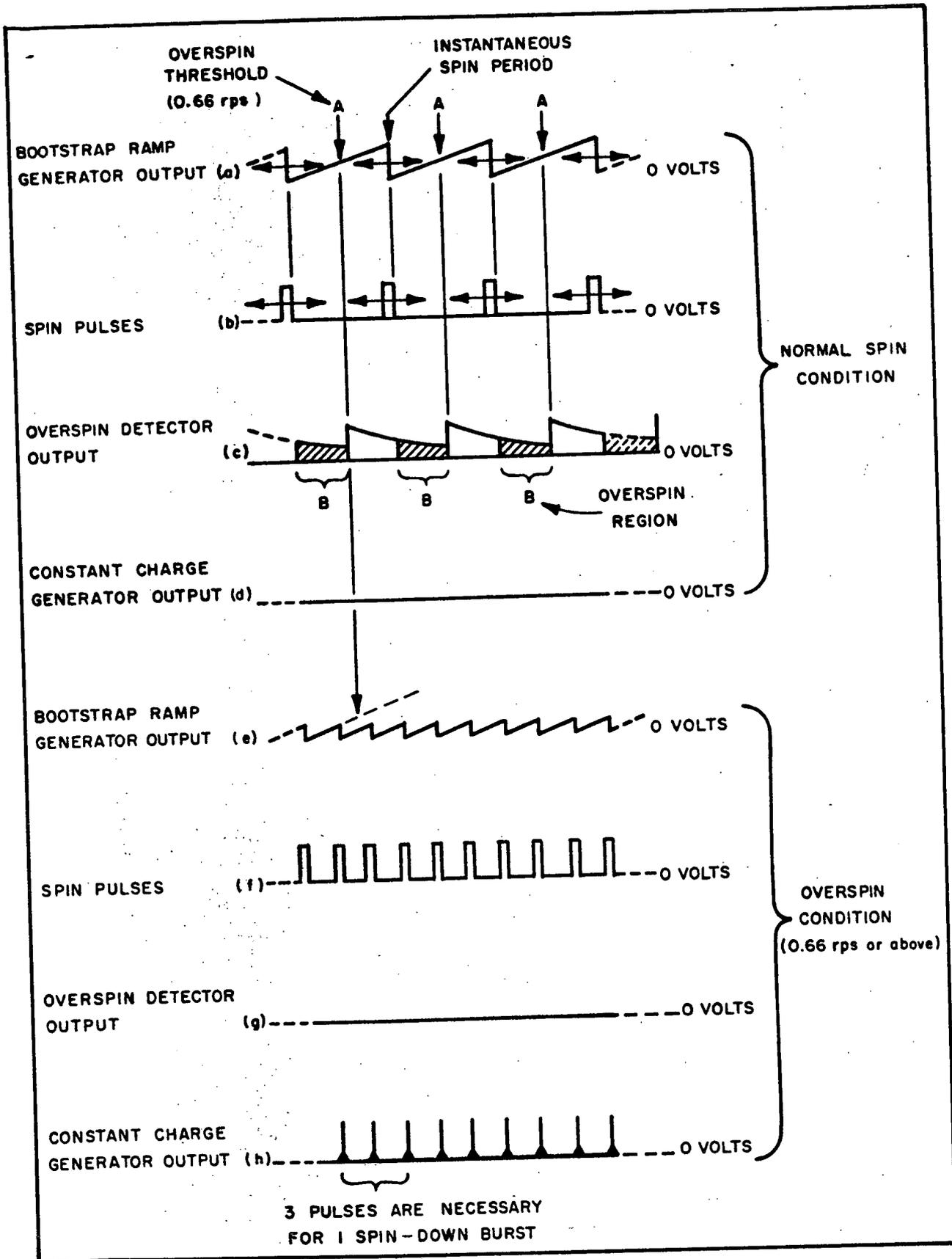
Fig. 7-4 Spin-Down Loop Timing Diagram

for a fixed period of time (Fig. 7-4h). The length of this pulse
is governed by the differentiated leading edge of the spin pulse
(Fig. 7-4f) which is used as a gating pulse.

The timing relationships of the ramp, spin pulses, overspin
thresholds, and constant charge generator outputs are shown in
Fig. 7-4.

Integrator, Spin-Down Burst Generator, and Solenoid Driver. The
output pulses from the constant charge generator are fed to an
integrator circuit where they are stored on a capacitor. This
summing action continues for three consecutive pulses at which
time the charge is sufficient to forward bias a uni-junction
switch. The output pulse from the uni-junction circuit is then
fed to the spin-down burst generator.

The spin-down burst generator is a mono-stable multivibrator
with a four-second time constant. For each integrator pulse,
this circuit generates a four-second rectangular wave output.
This output is fed to a solenoid driver circuit which provides
a ground path for the spin-down solenoid. A four-second burst
of gas is then released from the spin-down nozzles.

Underspin Detector, Constant Charge Generator, Integrator, Spinup
Burst Generator, and Solenoid Driver. The other destination of the
ramp discussed in the Bootstrap Ramp Generator subsection, is the
underspin detector in the underspin loop. This detector functions
as an overvoltage detector.

As the ramp voltage increases (signifying a decreasing spin rate),
the turn-on threshold of the underspin detector is ultimately reached.
This turn-on threshold corresponds to a spin rate of about 0.45
rps.

The output of the underspin detector is used to trigger another constant charge generator. This generator differs from the overspin constant charge generator in that the latter was gated by the spin pulses and this one consists of a mono-stable multi-vibrator and, therefore, provides its own timing. The purpose, however, is the same in both cases.

As in the overspin loop, the output of the constant charge generator is stored by a capacitor in the integrator circuit. After three pulses are received, the integrator triggers the spinup burst generator and a four-second pulse is fed to the spinup solenoid driver. A ground path is then provided for the spinup solenoid and a four-second burst of gas is released from the spinup nozzles.

The timing relationships of the ramp, underspin detector, and the constant charge generator are shown in Fig. 7-5.

Initial Despin. The automatic spin control circuits also provide the initial despinning of the wheel from about 99 rpm (1.67 rps) to 0.62 rps during the launch phase.

7.6.3   Manual Spin Control

Manual spin control can have two separate functional operations: (1) if a spin rate outside the limit of 0.39 to 0.70 rps is detected, the auto spin limiter (ASL) circuits automatically disable the automatic spin circuits to prevent spin rate run-away, and (2) a ground command control capability permits spin rate correction when the manual mode is selected.

Detection of the ASL overspin and underspin rates is similar to spin rate sensing in automatic control operation. Spin detector No. 3 (Fig. 7-3) provides a spin pulse to the pulse amplifier
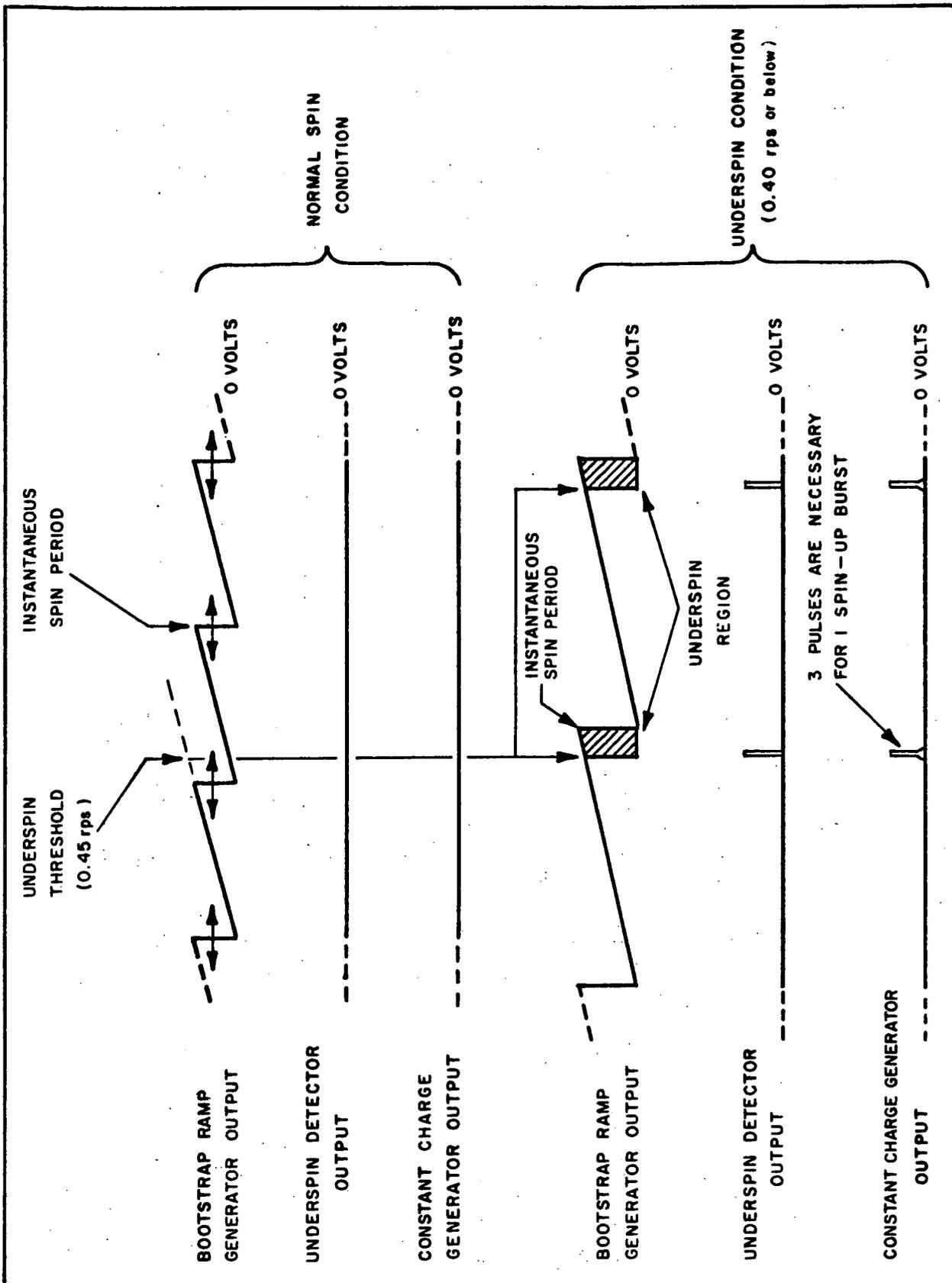
Fig. 7-5 Spin-Up Loop Timing Diagram

once every wheel revolution. The amplifier output is fed
through a pulse shaper and then to a ramp generator whose out-
put voltage rises linearly with time. The ramp generator out-
put reaches a peak level which is a measure of the wheel
revolution period. Both overvoltage and undervoltage detectors
are then used to sense the low and high triggering rates of
0.39 and 0.70 rps, respectively.

The undervoltage (overspin) detector is very similar to that
used in the auto spin-down circuits. When the spin rate goes
above 0.70 rps, the output from the overspin detectors drops to
zero. When this happens, the first unsuppressed spin pulse
energizes the gate amplifier, which in turn energizes the driver-
lockout circuit and latches the auto-manual relay to the manual
position. The underspin detector responds to the high ramp
voltage peaks occurring with wheel spin rates of 0.39 rps or
less, and also causes the auto-manual relay to switch to the
manual position.

The ASL circuits are disabled during the launch sequence phase
so that the auto spin control circuits can perform the initial
despin. Disabling of the ASL circuits is accomplished by
latching the ASL enabling relay in its disabled position during
launch. This prevents power from being applied to the driver-
lockout circuit. At the 800 second timer signal, the ASL
enabling relay is latched in the enabled position, thereby
enabling the ASL circuits. The despin operation will have
normally been completed before the 800 seconds timer signal
occurs.

When the manual control mode is in effect, ground commands must
be sent to initiate four-second spin-up or spin-down bursts.
Repeater relays drive their respective burst generators and
solenoid drivers when commanded. The required command functions

for switching to either manual or automatic operation are also
shown in Fig. 7-3.

7.6.4   Day Power Turn-On

As mentioned, the spin pulses are also used to turn day-power
on and off at sunrise and sunset, respectively. A block diagram
of the day-night circuits is shown in Fig. 7-6.
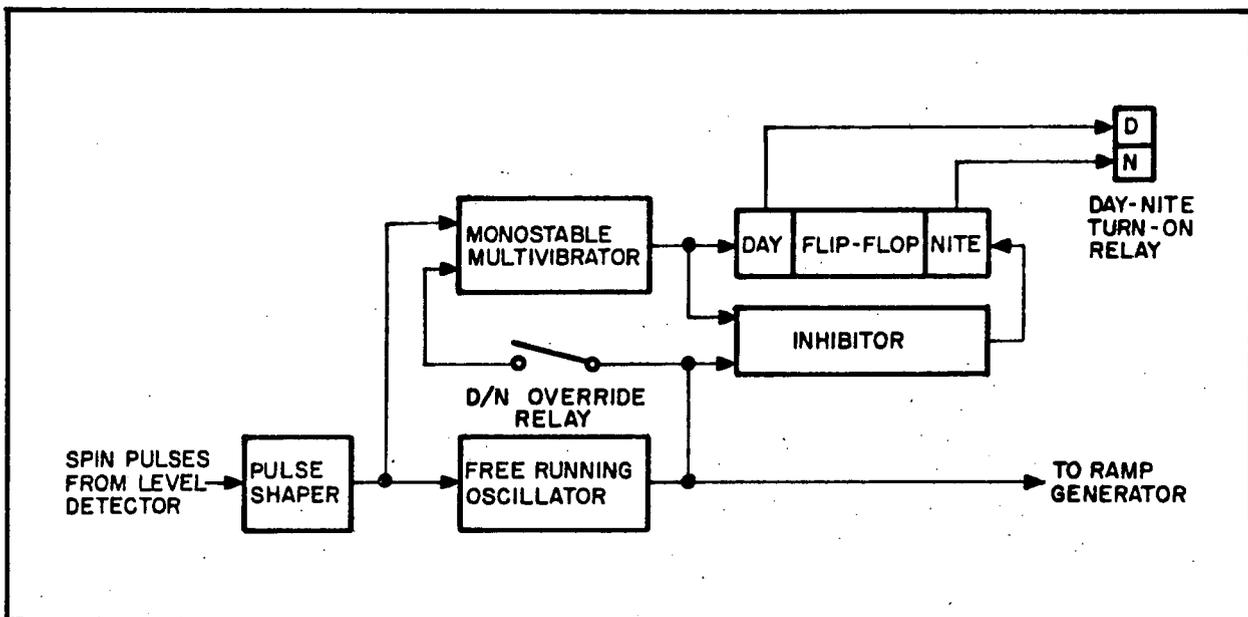
Fig. 7-6   Day Power Turn-On Circuits

First, consider the night condition, where there are no spin pulses, and, thus, the free-running oscillator is operating at its low rate. Also, the mono-stable multivibrator is not producing an inhibiting signal to the inhibitor because there are no spin pulses. Therefore, the output of the oscillator is not inhibited and it keeps the day-night flip-flop set to the night state.

When the spin pulses are present (during day only), the free-running oscillator is synchronized with the wheel spin rate. Also, the mono-stable multivibrator is triggered by the spin pulses and it outputs a pulse about 0.2 seconds long. This pulse is used to set the day-night flip-flop in the day state and also to inhibit the oscillator output from resetting the flip-flop. This action continues until the spin pulses are no longer present, which signifies night, and the flip-flop is again set to the night state by the oscillator pulses.

A means has also been provided by which day-power, during test operations, can be turned on when there are no spin pulses. This is done by feeding the oscillator output through the day-night eye override relay (which can be commanded from the test console only), and using this signal as an input to the mono-stable multivibrator. In this case, the output of the multivibrator is used to set the day-night flip-flop to the day state and also to inhibit itself from resetting the flip-flop.

It is important to note that this function can be used only during ground testing because the D/N eye override relay cannot be operated through the spacecraft command system. However, another relay is provided (the day-night bypass relay), which can be used to bypass the auto day-night turn-on circuits. This relay is commandable through the spacecraft command system.

7.7 OFFSET POINTING CONTROL FUNCTIONAL DESCRIPTION

The OSO-6 pointing control system can be operated in any of four modes:

      (1)   normal center-point mode

      (2)   large raster mode

      (3)   offset point mode

      (4)   small raster mode

In the normal center-point mode, the optical axes of the pointed experiments are pointed at the center of the sun with an accuracy better than ± 1 arc minute. In the large raster mode, the pointed experiments are caused to scan the sun with a 64-line (46 by 46 arc minute) raster pattern centered about the sun. In the offset point mode, the pointed experiments can be pointed at any one of 16,384 points (128 by 128 grid) within the large raster pattern. In the small raster mode, the pointed experiments can be directed to raster about any of the offset points with a raster pattern 7 arc minutes in azimuth and 7.5 arc minutes in elevation.

7.7.1 Large Raster Mode

In the large raster mode, the pointed experiments scan at the rate of one line every 24 main frames, or 0.32 sec/MF x 24 MF = 7.68 seconds per line. The period for a complete large raster pattern is then 7.68 sec/line x 64 lines = 491.52 seconds or 8.2 minutes. The nominal size of the large raster pattern is 46 by 46 arc minutes which means that each line is separated by 46 ÷ 63 spaces = 0.73 arc minutes. The size of the large raster is affected by changes in solar intensity due to the elliptic orbit of the earth. This effect is approximately six percent over a period of one year.

Figure 7-7 illustrates the OSO-6 raster pattern together with the offset point grid coordinates. Point A represents the on-point position (center of the solar disc). The large raster pattern centers about this point.

When the large raster mode is selected, a start-of-raster (SOR) pulse is generated by the raster synchronizing generator which causes the raster generator to begin the large raster at the center of raster line 1. Thereafter, a SOR pulse is generated each time a new raster pattern is begun (each 8.2 min). Also, a start-of-line (SOL) pulse is generated each time a new line is begun (every 7.68 sec).

7.7.2   Small Raster Mode

Point B (Fig. 7-7) illustrates the small raster mode. The size of this raster is 7.5 arc minutes in elevation and 7 arc minutes in azimuth and consists of 16 equally spaced lines. This means that each line is separated by 7.5 ÷ 15 spaces = 0.5 arc minutes. The total time for each raster is one main frame or 30.72 seconds. The sweep time for each line is then 30.72 ÷ 16 = 1.92 seconds.

The small raster is always centered around whatever offset point coordinate is programmed into the offset point generator.

7.7.3   Offset Point Mode

By means of the offset point mode, the pointed instruments can be pointed at any one of 16,384 points within a square pattern about the size of the large raster pattern.

In Fig. 7-7, the numbers 1 through 64 (in both azimuth and elevation) define the offset pointing grid (excluding the vernier offset). Any coordinate point on this grid can be selected by sending the proper combination of offset commands.

Fig. 7-7 OSO-6 Raster Patterns

In addition, the 64 by 64 offset point grid can be expanded to 128 by 128 by using the azimuth and elevation vernier offset commands. The azimuth vernier command will move the pointed experiments one-half step (or 1/2 bit) to the right of whatever point the experiments were previously pointing at. The elevation vernier command will move the experiments 1/2 bit up.

Note: When the vernier offset commands are used, the actual size of the offset grid has been enlarged by 1/2 bit up and 1/2 bit to the right. This means that the center of this new grid is 1/4 bit up and 1/4 bit to the right of the center of the large raster.

7.7.4 · OSO-6 Offset Circuit

Incorporation of the offset point and small raster into the OSO-6 scan system required the following:

(1) Addition of a new PC board, containing the circuitry for the selection of offset points.

(2) Modification of the existing large raster circuitry to incorporate the small raster electronics and also to update the design.

(3) Modification of timing circuits to incorporate small raster and also to enable the offset system to be versatile.

(4) Addition of monitors to allow verification of offset mode and offset point selected.

The above changes were incorporated into the existing power amplifier and servo amplifier assemblies.

Figures 7-8 and 7-9 are functional block diagrams of the offset point/offset scan system.

Voltage levels, corresponding to the offset points on the offset grid pattern, are generated by changing the gain of an operational amplifier in discrete steps. The op-amp gain is changed on ground command by setting a flip-flop which turns on a field effect transistor (FET), connecting a selected resistor to the summing junction, thus changing the input impedance of the op-amp. All the offset point flip-flops are reset by one ground command. Although this adds slight complexity to ground operation, it decreases the number of commands required, thereby increasing system reliability. The offset point generators (azimuth and elevation) are identical and are summed at the input to the azimuth and elevation output operational amplifier by ground command.

Generation of a raster pattern requires a highly-linear triangular voltage waveform (azimuth) and a staircase function of uniform steps (elevation). The triangular wave form is produced by the azimuth integrator, with the peak-to-peak voltage and period dependent upon the pattern selected (large or small). The staircase waveform is produced by summing currents which result from six binary-weighted positioning currents that are controlled by their respective counter stages. The period of the clock input to the counter is dependent upon whether large or small raster is selected.

Azimuth and elevation signal generator outputs are summed at the input to their respective scale amplifier along with the offset point generator output. In addition to providing a summing point for the offset point and scanning signals, the scalar amplifiers permit fine adjustment of the amplitudes to obtain precise pattern sizes. The outputs of the scalar amplifiers are connected to their respective current generators where the scalar output
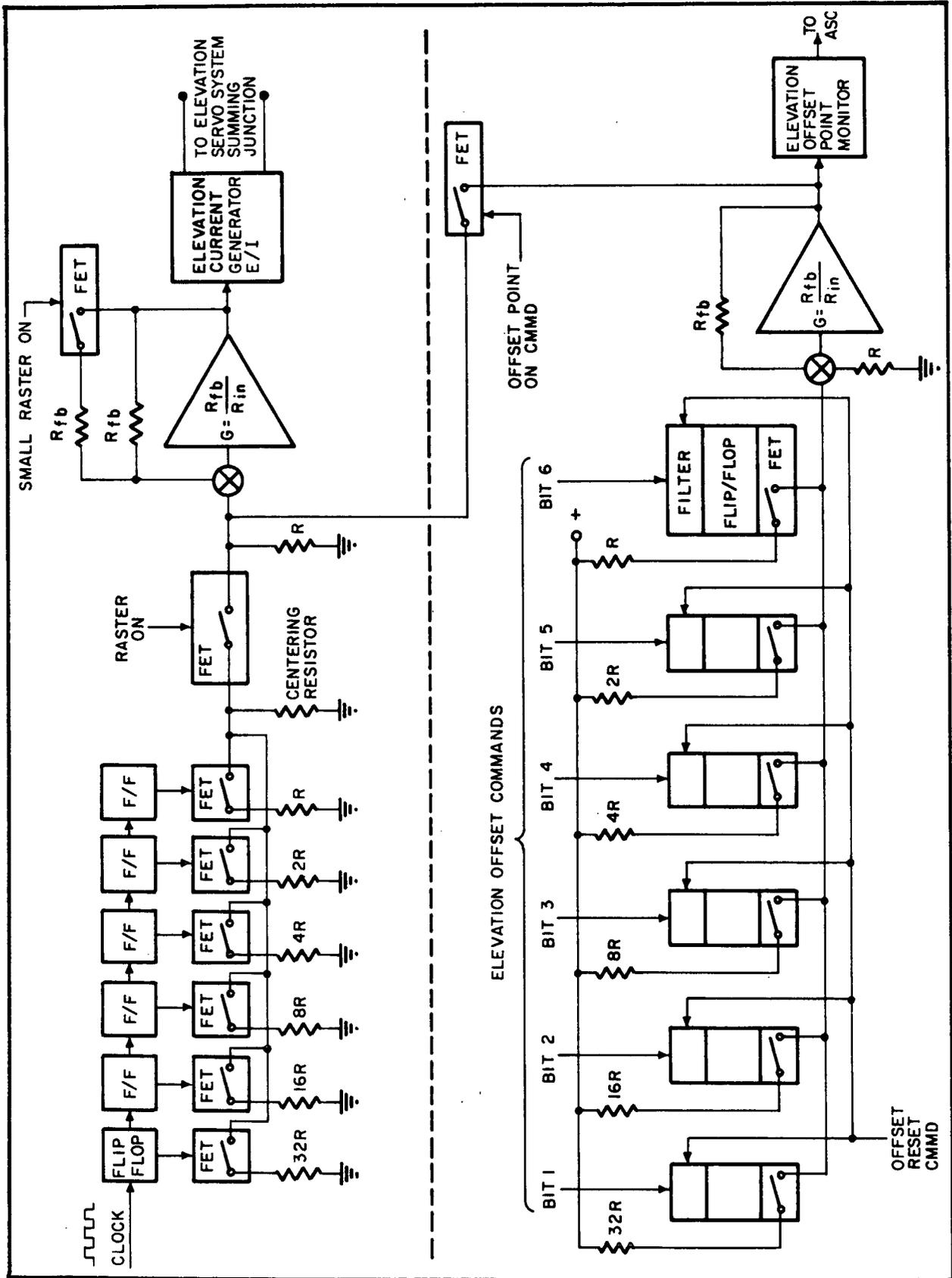
Fig. 7-8 Elevation Offset Channel

C.5

Fig. 7-9 Azimuth Offset Channel

voltages are transformed into proportional current signals.
The current signals are then summed with the servo control sun
sensor signals for offset mode operation.

7.8   COARSE SERVO FUNCTIONAL DESCRIPTION

The pointing control system provides automatic solar acquisition
and pointing of the sail and pointed experiments beginning each
orbit morning. The system also provides offset-mode operation
as described in Paragraph 7.7. The pointing servo consists of
three servo loops; namely, the coarse azimuth, fine azimuth,
and fine elevation. These servo loops consist of three separate
control sensor assemblies (control eyes), electronic control
circuits contained in the servo amplifier and power amplifier
assemblies, and the azimuth and elevation torque motors.

The azimuth and elevation servo loops differ in operational
concepts in only a few respects. The most important difference
is that the azimuth control requires two solar sensing functions
(one for coarse sensing and one for fine sensing) while the
elevation control requires only fine solar sensing because the
pitch control system performs the coarse elevation pointing
function. The dual azimuth sensing design requires an additional
preamplifier and compensation network as well as an on-target
relay that switches azimuth servo control from coarse to fine
sensing circuitry when on-target pointing is achieved.

Each orbit morning, day power is applied to the pointing control
circuits by the day-night sensing circuits in the power distribution
system. The coarse azimuth servo loop then despins the sail from
the wheel spin rate to within 3 degrees of stationary azimuth
pointing. Control is then transferred to the azimuth fine loop by
the coarse/fine control (on target) relay. The azimuth fine and
elevation fine loops then complete the acquisition, aligning the
pointed instruments to within one arc minute of the center of the
sun.

In steady-state pointing, the azimuth servo drives against the unidirectional friction drag of the azimuth bearings contained in the wheel hub. Thus, the azimuth servo must develop a drive torque to counteract this ever-present, and nearly-constant, drag. The elevation servo, on the other hand, must overcome the elevation bearing friction only when a correction in elevation is required. Because of the extremely slow drift rate of the observatory in pitch, no elevation drive is necessary unless there is spin axis nutation or wobble present or if the pointed instruments are dynamically unbalanced so that they assume a different position at night, which would then require elevation drive in the morning to bring them back to the on-point position. Also, a small amount of elevation drive may result from unsymmetrical loading by the flex-cables that provide electrical connection between the pointed instruments and the sail electronics.

Because the functional differences between the azimuth and elevation servos are minor, a detailed functional description of the azimuth servo loop is first presented, followed by a brief discussion of the elevation servo. A block diagram of the OSO-6 pointing control system is shown in Fig. 7-10.

The main difference between the OSO-5 and OSO-6 pointing control systems consists of the offset mode capability described in Paragraph 7.7.

7.8.1   Coarse Azimuth Servo Sensors

The coarse azimuth servo despins the observatory upper structure from the night spin rate each orbit dawn. It aligns the pointed instruments with the sun to within three· degrees in azimuth. The on-target sensor then enables the fine azimuth servo. This discussion of the coarse azimuth servo will cover the operating characteristics of the coarse sensors, circuits unique to the
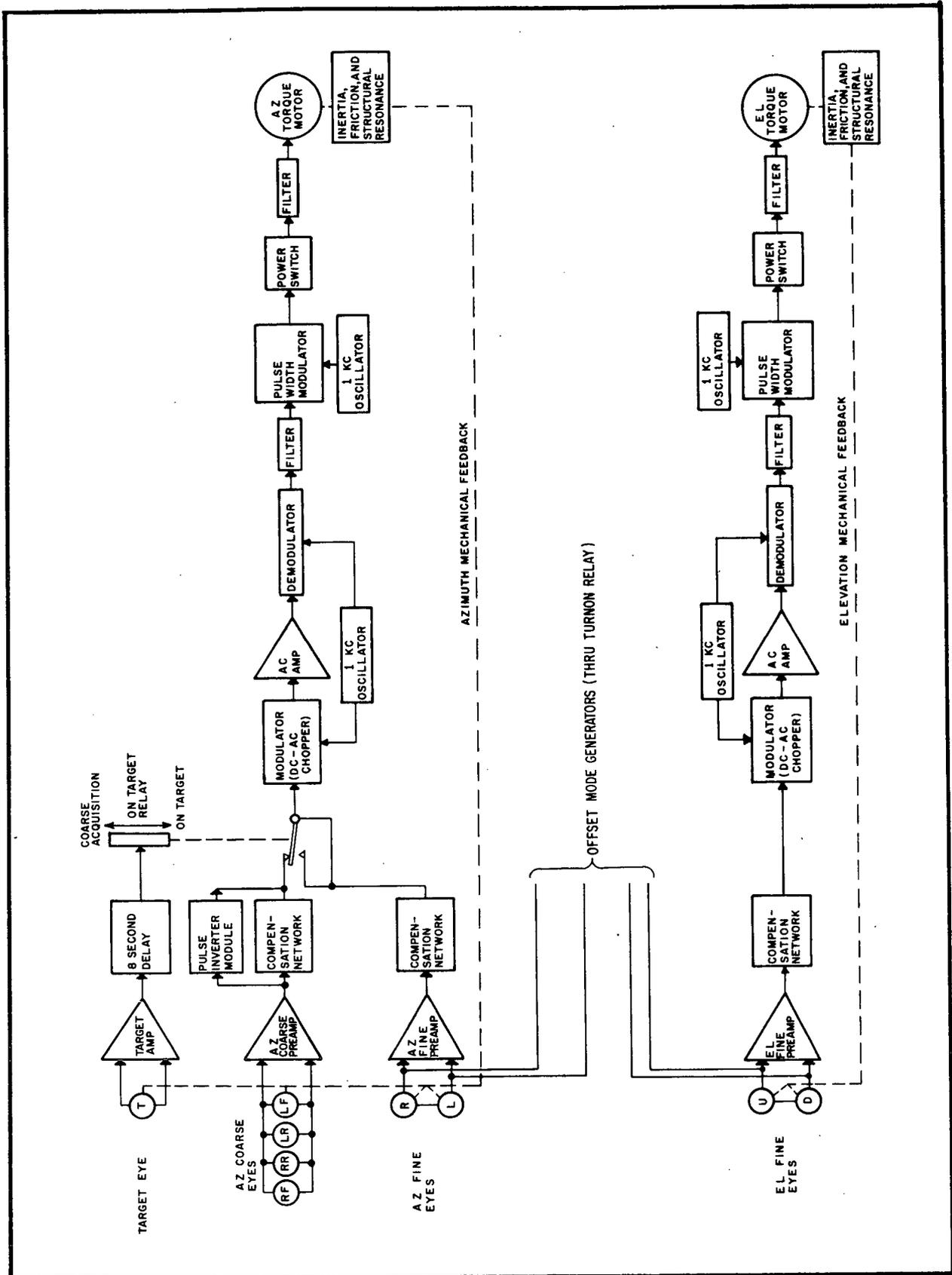
Fig. 7-10 OSO-6 Pointing Control Block Diagram

coarse-acquisition mode, circuits common to the coarse and fine modes, and the functions of the target eye and its circuits.

Each of the four azimuth-coarse eyes consists of a silicon photo-electric sensor mounted to an aluminum block and covered by a Corning No. 2600 red glass filter. Each eye assembly is protected from radiation damage by a 0.1-inch quartz cover slip installed over the filter. The eyes are mounted in holders which establish the desired view angles. The completed eye assemblies are shown in Fig. 7-11.

The vertical field-of-view of the eyes is from 15 degrees above to 15 degrees below the wheel plane. This restricted field-of-view prevents the eyes being affected by light reflected from the top of the wheel or from earth albedo. It also prevents solar sensing and servo drive when the spin axis is not perpendicular to the solar vector within 15 degrees. The composite field of the eyes in azimuth is 360 degrees. The static view angle, however, is about 110 degrees for each front eye, and 95 degrees for each rear eye (see Fig. 7-12).

As the sail rotates, the coarse eyes produce a current waveform (error signal) shown in Fig. 7-12. The figure shows the location of each eye and the output waveform of each as the sail rotates. The four individual outputs are summed as shown on their way to the azimuth coarse preamplifier.

FINE POINTING
SOLAR
DIRECTION

ALUMINUM
BLOCK (TYP)

LEFT FRONT   RIGHT FRONT

CELL
NORMAL
VECTOR
(4 PLACES)

LEFT REAR   RIGHT REAR

NOTE: THE CELL NORMAL VECTORS ARE PERPENDICULAR
TO THE SPIN AXIS.
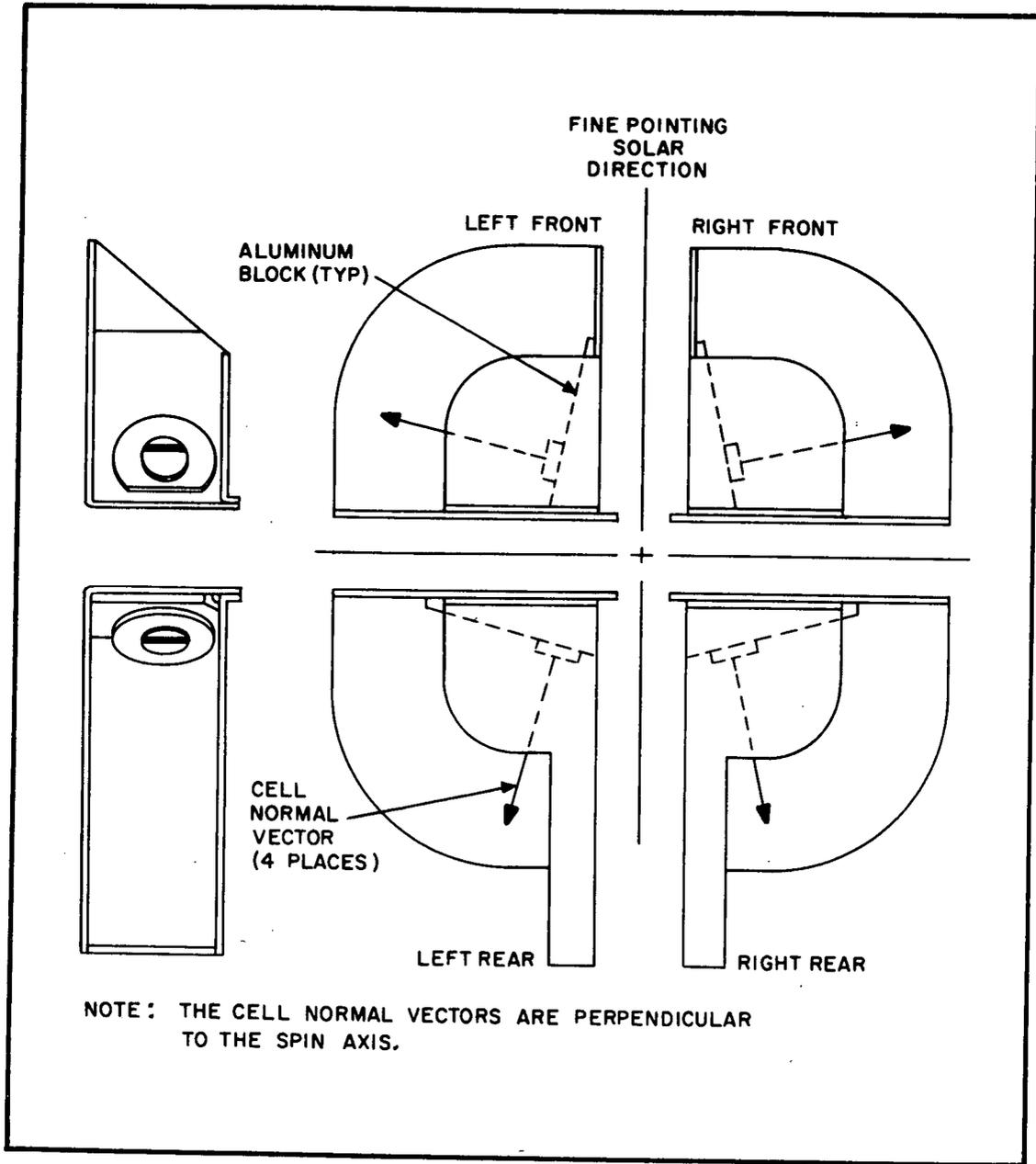
Fig. 7-11   Coarse Sensor Assemblies

POSITIVE SIGNAL
(DESPIN DRIVE)

NEGATIVE SIGNAL
(SPIN-UP DRIVE)

+360° ①
+260°
+180° ③
+100° ②
0° ①
-100° ④
-180° ③
-250° ②
-360° ①

.2 ma

0 ma

COARSE SENSOR OUTPUTS
RIGHT FRONT
RIGHT REAR
LEFT REAR
LEFT FRONT

NOTES

THE AZIMUTH DRIVE SATURATES AT ABOUT 0.2 MA EYE-CURRENT SIGNAL.

INDIVIDUAL EYE CURRENTS ACTUALLY RETURN TO ZERO BUT ARE SHOWN SEPARATED FOR PURPOSE OF ILLUSTRATION.

THE CURVE REPRESENTED ⊖⊖⊖⊖⊖⊖⊖⊖⊖ IS THE COMPOSITE SIGNAL AT THE PREAMP INPUT.

THE NULL AT POINTS MARKED ③ IS THE UNSTABLE NULL WHICH OCCURS WHEN THE REAR OF THE SAIL POINTS TOWARD THE SUN.

THE POINTS MARKED ① REPRESENT THE STABLE NULL OCCURRING WHEN THE POINTED INSTRUMENTS ARE AIMED TOWARD THE SUN.

THE COMPOSITE SIGNAL PEAKS AT ABOUT 1.75 MA. INDIVIDUAL EYE SIGNALS PEAK AT ABOUT 1.3 MA.

② RIGHT END

RF 110°

RR 95°

SAIL ROTATION

FRONT
TOP
REAR

SHADOW SHIELD

LF 110°

LR 95°

①

③

④ LEFT END

COARSE EYE FIELD-OF-VIEW

Fig. 7-12 Coarse Sensor Signal

The composite signal from the coarse eyes has two important terms: a position term and a time-rate-of-change term. The position term is the instantaneous amplitude of the signal. The rate term reflects the rate at which the position term changes. Each of these terms plays an important role in the coarse-acquisition function.

The polarity of the position term is such that the positive area of the signal (shown on Fig. 7-12) causes the sail to be driven in the despin direction. This portion of the signal is produced by the right-front (RF) and right-rear (RR) eyes. The negative area of the signal causes the sail to be driven in the spinup direction. This portion of the signal is produced by the left-rear (LR) and left-front (LF) eyes.

Note that the drive produced by signals in the region of the unstable null (③ on the curve) are away from the null, while those near the stable null (① on the curve) are toward the null. This assures that the sail will only stop at the stable null, which is the desired position.

The rate term dominates while the sail rotates quickly. A rate network and pulse-inverter circuit (which will be discussed later) convert most of the negative (spinup) signal into a positive signal. This action continues until the spin rate is reduced to about 0.07 rps. The position term then becomes dominant and remains so throughout the rest of the acquisition. Thus, the rate term is dominant at high spin rates and the position term at slow spin rates.

7.8.2   Coarse-Servo Circuits

The circuits described in this paragraph are those peculiar to the coarse-mode operation of the azimuth pointing control.

The coarse preamplifier converts the coarse-eye output (Fig. 7-12) from a current waveform to a voltage waveform. The preamp is a differential, low drift amplifier and incorporates automatic compensation circuits for changes in its own gain. It provides a low-impedance load for the eyes and a high-impedance output to the rate network. The preamp is designed for unsaturated operation between +1.5 ma of input eye current.

The output of the azimuth coarse preamp is fed into a compensating network and a pulse-inverter module (PIM) as shown in Fig. 7-10. Rate sensing and the conversion of spinup signals are accomplished in these circuits.

The compensating network consists of a resistor-capacitor coupling arrangement which provides dc coupling for low-spin-rate eye signals and predominantly-despin signals for higher spin rates. This network provides for a positive (despin) drive signal when the eye signal is positive-going and spin rate is greater than about 0.07 rps. Figure 7-13(a) and (b) show the relationship between the eye signal and the ideal output of the compensating network. Notice how this compensation turns the symmetrical waveform into a predominantly positive signal.

Part of the despin signal is lost, however, due to differentiation of the ideal output. The resultant waveform is shown in Fig. 7-13(c). The negative portion of this waveform represents spinup drive and therefore decreases net despin efficiency. The effect of the negative signal is substantially reduced by the addition of the PIM circuit.
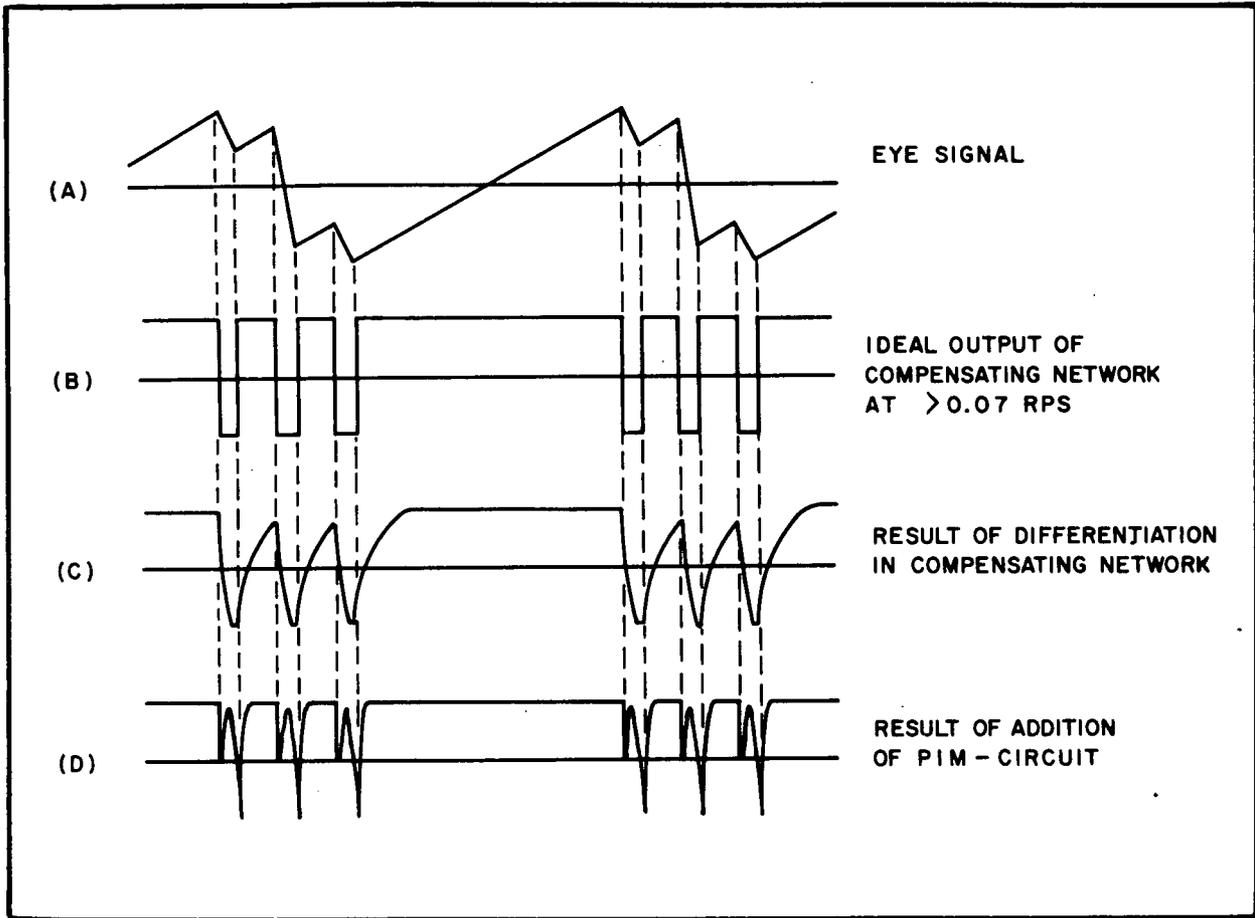
Fig. 7-13   Rate Signal Compensation

The pulse-inverter module (PIM) tends to superimpose a positive signal on the negative output of the compensating network. It does this through a time constant which prevents it from interfering with spinup drive in the final phase of solar acquisition. The PIM also peaks up the leading edge of the positive output of the compensating network. Both of these actions of the PIM increase net despin drive efficiency as shown in Fig. 7-13(d).

When the sail rotation is reduced to about 0.07 rps, the compensating network and PIM give way to direct dc coupling of the eye signal to the modulator (Fig. 7-10). The position term of the

error signal then becomes dominant and controls the final stages of coarse azimuth acquisition.

The net efficiency of the azimuth coarse servo is expressed in a term known as net despin drive. It is found by subtracting the percent of time the drive is negative (spinup) from the percent of time drive is positive (despin) during a revolution of the sail. Net despin drive is expressed as a percent and plotted as a function of sail spin rate.

The output of the coarse-servo circuits is fed to the circuits which are common to the azimuth servo in both the coarse-acquisition and fine-pointing modes (Para. 7.8.3).

7.8.3   Common Azimuth Circuits

The characteristics of the azimuth modulator, ac amplifier, demodulator, pulse-width modulator, power switch, and torque motor are described in this paragraph. These are circuits which are common to the azimuth fine and coarse servo loops.

The output of the rate network is fed through the on-target relay to the modulator.

The modulator is a dc-to-ac chopper, which samples the differential output of the coarse preamplifier and rate network at a 1,000 Hz rate. The output of the modulator is a square wave with a peak-to-peak amplitude equal to the differential input signal. This output is fed to the azimuth ac amplifier.

The ac amplifier saturates with an input corresponding to coarse-eye currents of about ±0.2 ma. Thus, during acquisition, this amplifier is saturated most of the time, providing a clipped output. This output goes to the demodulator for reconditioning.

The <u>demodulator</u> is driven by the same oscillator that drives the modulator. It reconditions the output of the ac amplifier as a dc signal. The output of the demodulator is an amplified and clipped reconstruction of the error signal, except that it is referenced to a 7.5 v bias level, and has 1,000 Hz transients in it. The response curve for the coarse-preamp/ac-amplifier combination is shown in Fig. 7-14.

The output of the demodulator is fed through a filter which removes the 1,000 Hz transients and any other system noise down to about 50 Hz. This filtered signal then goes to the PWM circuit.
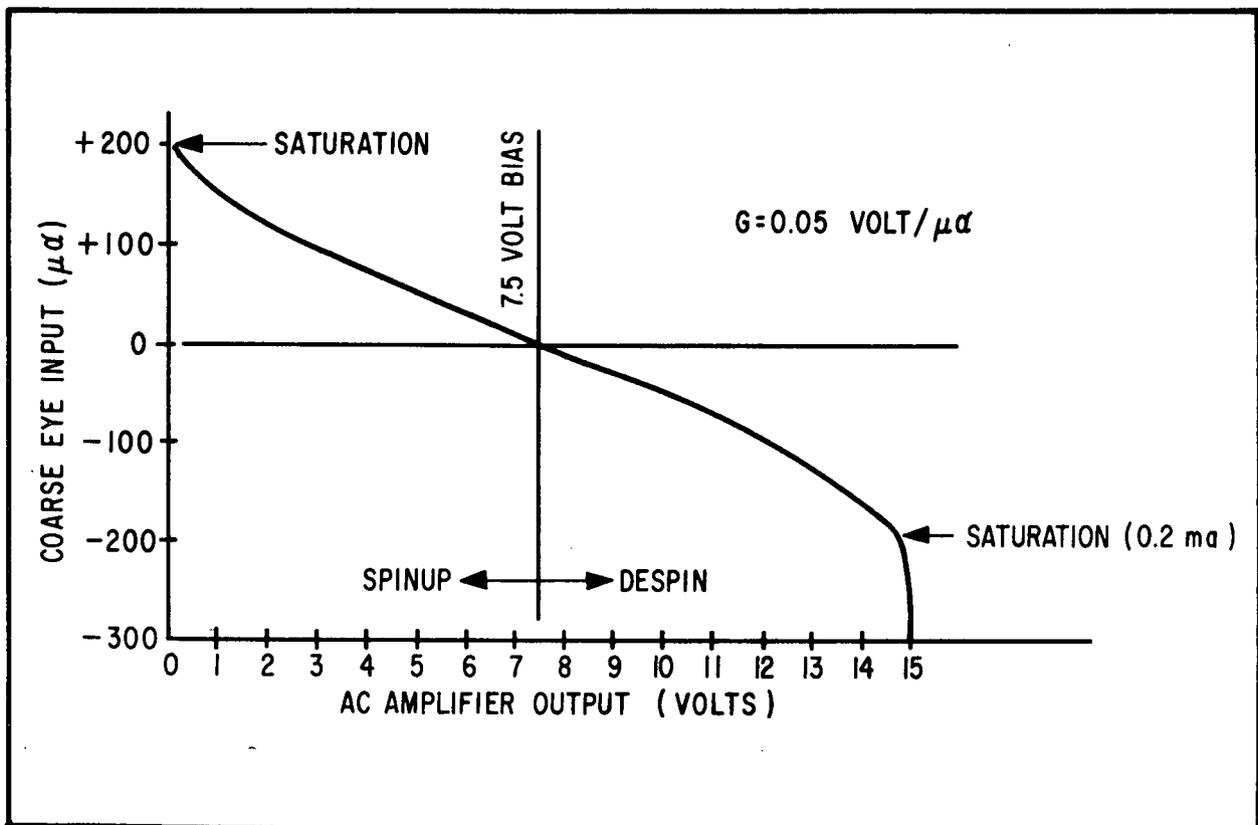
Fig. 7-14 Typical Coarse Azimuth Preamp/AC Amp Response

The pulse-width modulator (PWM) converts the amplitude modulated error signal into a constant-amplitude 1,000 Hz waveform in which pulse width (or duration) is proportional to the amplitude of the dc error signal. The principle reason for using a PWM design is to reduce power dissipation in the power switch. This eliminates a potential component-reliability problem in these circuits.

The PWM is designed to produce an output only when the input varies above or below the 7.5 v bias level. This variation is typically between about +2 and +13 v as the sail rotates. The PWM has a dead zone (no output) at about 7.5 $\pm$ 0.25 v. The dead zone has no effect on the operation of the azimuth loop since the wheel always spins in the same direction, holding azimuth-loop operation on one side of the dead zone (except during acquisition). The PWM dead zone is shown in Fig. 7-15.

The PWM actually consists of two separate PWMs that work together. One converts despin signal (>7.5 v) into pulse-width waveform and the other converts spinup signal (<7.5 v). Outputs of both PWM circuits are positive. Each output is fed to one side of the power switch for power amplification. Obviously, outputs from both PWM circuits cannot occur simultaneously since the inputs are separated at 7.5 v.

The power switch consists of two pairs of saturable transistor switches. One pair of transistors is controlled by the despin PWM and drives the azimuth torque motor in the despin direction. The other pair is controlled by the spinup PWM and drives the motor in the spinup direction. The outputs from the power switch are filtered before being applied to the torque motor to reduce power losses in the motor windings. The drive current is averaged by the filter.
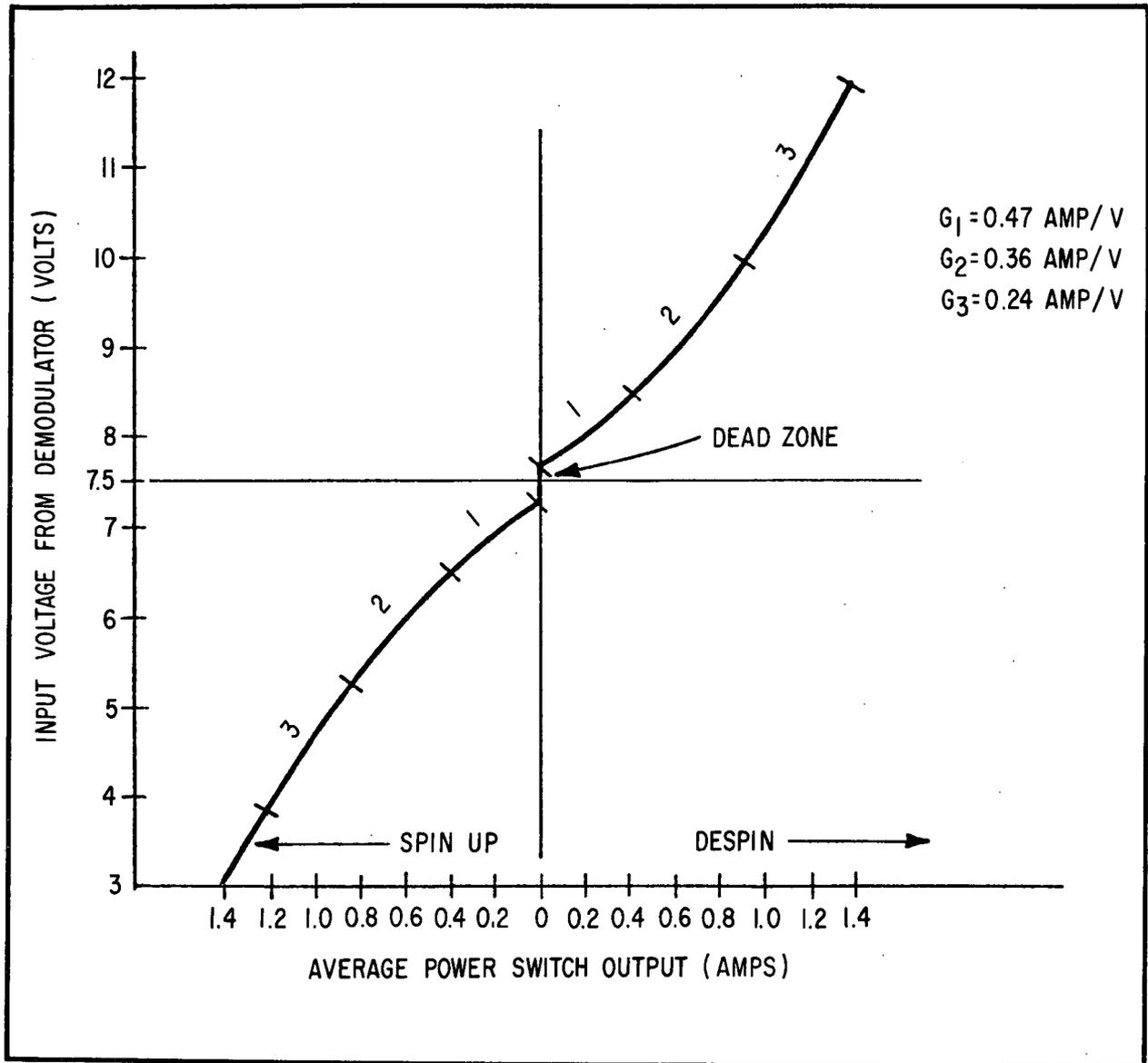
Fig. 7-15 Typical PWM/Power-Switch Response

Operation of the power switches in the region of saturation and cut-off avoids the heat dissipation and power waste inherent in other areas of the transistor load line.

Figure 7-15 shows a typical overall response of the PWM and power-switch circuits to inputs from the demodulator.

The azimuth torque motor consists of a permanent-magnet stator attached to the azimuth shaft (fixed to the sail), and a rotor attached inside the wheel hub. The drive power from the power switch (located in the power box in the sail) is fed through a brush assembly to the commutator of the rotor.

7.8.4   Target Eye and Coarse/Fine Switch

The target eye and coarse/fine switch are the link between the azimuth coarse and fine servo loops. When the coarse loop has brought the pointed instruments within 3 degrees of the solar direction, the target-eye senses the sun. After about an eight-second delay, coarse/fine switching occurs. The input to the azimuth ac amplifier is then developed in the fine-mode sensors and circuits.

The target eye is mounted in the fine-control-eye block, presented in subsection 7.9. The eye produces about a 2 ma current when the pointed instruments are aimed within the capability of the coarse pointing control. The elevation field-of-view of the target eye is about ±6 degrees. Operation of the elevation servo is, therefore, not required for coarse/fine switching. The field-of-view of the azimuth fine-control eyes is greater than ±8 degrees, providing the necessary overlap for smooth coarse/fine transition. The output of the target eye is fed through the target-eye amplifier to the coarse/fine relay driver and the telemetered target-intensity monitor.

The azimuth coarse/fine switch is often called the "on-target relay" for brevity. After the target-eye has been illuminated for the 8-second delay period, the on-target switch is released to the "fine" position. The switch is a momentary-operation relay and is energized in the coarse mode. This conserves power, since the servo operates in the fine mode for more than 98 percent of the time powered.

The on-target relay switches out the signal from the azimuth coarse eyes and completes the azimuth fine-control servo loop. This function is illustrated in Fig. 7-10. In addition, the relay generates an "on-target" signal which is monitored by telemetry, and enables the auto-pitch limiter (APL) circuits as described earlier in this section.

The target-intensity monitor is a telemetry sampling of the output of the target-eye amplifier. The linear characteristic of this amplifier provides for in-flight evaluation of target intensity and sensor degradation.

7.9   FINE AZIMUTH AND ELEVATION SERVO LOOPS

Fine azimuth and elevation servo control begins at the completion of coarse acquisition (coarse-fine switching). These servo loops function very much like the coarse azimuth servo loop, with the following exceptions:

(1)   The error signals are produced by the azimuth and elevation fine eyes.

(2)   A different azimuth preamplifier is used.

(3)   The rate-sensing and PIM circuits are replaced by compensation networks designed to adjust servo stability and response characteristics to other servo parameters such as static gain and structural resonances.

(4)   The elevation torque motor produces about one-half the torque that the azimuth motor produces.

The <u>fine-control eyes</u> for the azimuth and elevation servos each
consist of a pair of BBRC FE-5 fine sensors. Each FE-5 sensor has
a focusing lens, a red-glass filter, and a silicon solar cell.
One half of each red filter disc is masked by a thin aluminum film.
The output of each cell is a function of the direction it is
pointing, as shown in Fig. 7-16.

The electrical leads from each sensor are connected in opposing
polarity to the other half of the sensor pair. The sensors of each
pair point slightly away from each other so that, on point, each
sensor produces about ten percent of its maximum output. The two
opposing currents produce a stable null in the on-point direction.

An azimuth or elevation error signal results when the sensor
assembly is rotated from the on-point position. Figure 7-16 shows
the output curve for a typical fine-eye pair. The figure also
shows the front view of the eyeblock (including the Target eye),
the aluminum shield orientation, and the behavior of individual
eye outputs near the null region.

The <u>fine-servo circuits</u> are similar, in many ways, to those of the
coarse azimuth loop already discussed. The areas presented here
are those which are peculiar to the fine-control mode. The fine
azimuth and elevation servo loops seek a precise null position,
and maintain pointing accuracy within ± 1 arc minute.

As can be seen in Fig. 7-10, the fine-servo loops differ from the
azimuth coarse loop primarily in the development of the sensor
signals. The fine azimuth and elevation preamplifiers convert
the current signal from the eyes to a voltage signal for the
ac amplifiers.
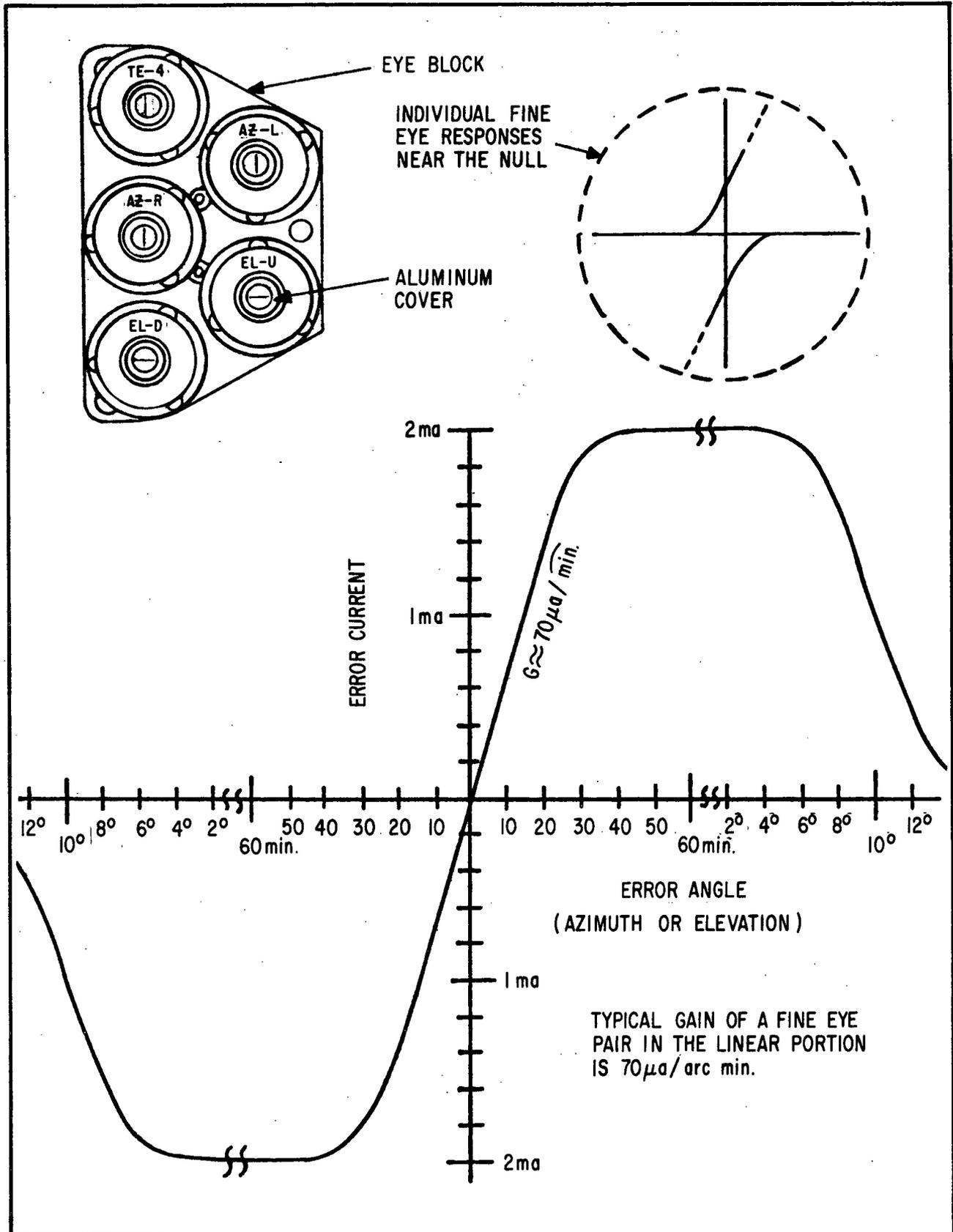
Fig. 7-16 Typical Fine-Eye Response

The circuits of the azimuth fine control loop are common to the
coarse loop beyond the preamp stage. The elevation loop has its
own modulator, ac amplifier, demodulator, PWM, power switch, and
torque motor, similar to azimuth.

The _fine servo performance_ characteristics are effected by such
factors as friction, motor back-emf in azimuth, and flex-cable
loading in elevation.

The azimuth fine eyes are aligned to produce zero pointing error
when driving against the normal friction of the azimuth bearings.
A 0.5 in-lb increase in shaft friction will cause a pointing change
in the direction of spin of about 2 arc seconds. Similarly, a 0.2
rps increase in spin rate would cause a shift in the direction of
spin of about 2.7 arc seconds. These examples show that the azimuth
servo can tolerate reasonable friction and rate changes. Normal
friction of the azimuth bearings is about 1.1 inch-pounds.

The PWM dead zone is reflected in the fine-servo gain curve shown
in Fig. 7-17. This dead zone has no effect on azimuth operation
since wheel spin always holds the sail to the side of the zone in
the direction of spin.

Appreciable pointing errors in the fine elevation servo loop
can be caused by elevation motor shaft friction, flex cable loading,
and electronic and mechanical alignment drift. The friction errors
would be the result of wobble or nutation of the spin axis. These
motions tend to displace the elevation gimbal above and below its
average pointing position and the pointing error would appear as
an approximate squarewave. If wobble were causing the motion, the
frequency would be identical with the wheel spin rate. If nutation
were causing the motion, the frequency would be a function of
wheel spin rate. Note that the spin rate/back-emf changes that
can cause pointing errors in azimuth do not apply to the elevation
servo.

SERVO-LOOP
SATURATION AT
$\approx 42 \mu \alpha$ INPUT

IN-LBS.

PWM
DEAD ZONE

-2   -1   0   1   2

ARC SECS.

MOTOR TORQUE

0

-0.7   0   0.7

FINE EYE INPUT (MINUTES OF ARC)
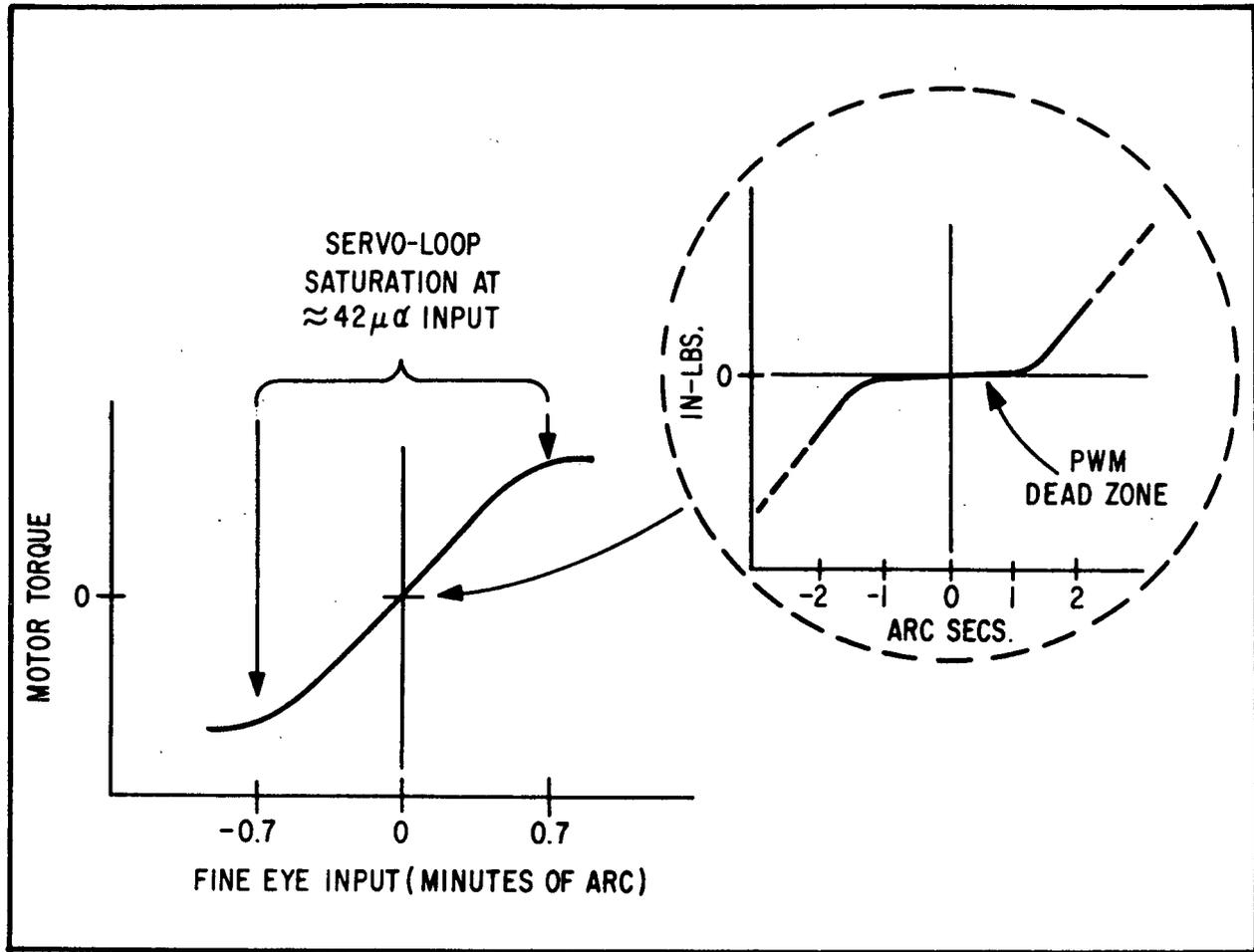
Fig. 7-17   Overall Fine Servo Gain Showing Dead Zone

The PWM dead zone can cause a 2.5 arc second instability in
elevation pointing. This instability is small compared to the
desired 1 arc minute pointing accuracy. Rather than raising the
gain of the loop and risking oscillation, the dead-zone is tolerated
as it is.

The combined friction and flex-cable-loading torques were determined
to introduce negligible error contribution.

During booster operation at launch, a relative alignment change
between the right and left pointed instruments could occur. Such
a change would cause a pointing error in the left instrument. The
error would show in the telemetered pointing data since the fine

control and pointing readout eye blocks are mounted on the right
instrument and the raster readout block is on the left instrument.

After considering all the disturbing drifts, the steady-state azi-
muth pointing accuracy is well within the ±1 arc minute requirement.

7.10   POINTING CONTROL INTERACTION WITH OTHER SYSTEMS

The pointing control system influences the operation of other
control systems. Other systems, likewise, produce an effect on
the operation of the pointing control. These interactions of the
pointing control are herein presented briefly.

The spin-control system normally makes spinup corrections at orbit
dawn because the night spin rate is slower than the day rate.
Spin-control circuits are not powered in the night condition.
During orbit night, azimuth-bearing friction brings the sail up
to the wheel spin rate. Due to the conservation of angular
momentum, the larger spinning moment of inertia at night results
in a decreased wheel spin rate. When day power is first applied,
before the angular momentum of the sail is imparted to the wheel
through the azimuth servo, the spin control system senses its
lowest spin rate. If a spinup correction is needed, it will occur
at this time.

Spin rate has an effect on coarse-acquisition time since it takes
the servo longer to slow down the sail from higher spin rates.
The motor back-emf is an increasing function of spin rate,
causing a shift in azimuth in the spin direction for higher spin
rates. This can result in pointing error and increased azimuth-
drive power consumption.

Pitch corrections by the pneumatic pitch control can only be
triggered when steady-state pointing is achieved. The pitch-eye
block is directional and must be aimed by the pointing control for

at least eight seconds before an excessive pitch error can be detected. The pneumatic jets must likewise be held at a fixed angular orientation to produce a pitch correction.

Pitch error must be less than 15 degrees for azimuth acquisition, limited by the field-of-view of the coarse eyes. Error must be less than 5 degrees for elevation pointing, limited by the freedom of traverse of the elevation gimbal. Pitch attitude within the 5-degree limit will influence the effect of flex-cable rigidity on pointing accuracy in elevation and the resultant drive power consumed.

The action of the Auto-Pitch Limiter circuit is disabled in the azimuth-servo coarse mode.

Nutation is caused by elevation acquisition at orbit dawn, by the torques produced in the elevation motor in the scan mode, and by pneumatic pitch corrections. Such nutations have a minimal effect on the pointing control, since servo gain is high enough to overcome them. Nutation energy is quickly absorbed by the nutation damper.

7.11   POINTING PARAMETERS AND MONITORS

The servo parameters and performance monitors used to evaluate the pointing control both before and during flight are described in this subsection. Other monitors in the pointing-control system are the "target intensity" and "on-target" monitors presented in earlier paragraphs. The conversion tables for translating tele-metered binary-coded data into engineering units are contained in BBRC document TN68-16.

7.11.1   Position Readout

There is a set of position-readout eyes mounted to the front of
each pointed instrument. These eyes are the same type fine-eye
pairs used for fine-pointing control. Their response is about
the same as that shown in Fig. 7-16. In addition to their use in
analyzing servo performance, the two sets of eyes provide for
evaluation of relative displacement of the pointed instruments.
Such displacements may occur due to vibration during the launch
sequence, or thermal warpage in orbit.

The "point" position readout eyes are mounted on one pointed
instrument and the "raster" readout eyes on the other. The
difference between the eyeblocks, other than their location, is
their gain.

The pointing readout eye amplifiers' gains are set for about ± 2 arc
minutes full-scale telemetry readout. These provide for a high-
resolution of pointing accuracy readout. The raster eye amplifiers,
however, are adjusted for a gain to provide accurate position data
in the range of the servo raster-mode operation. All eye amplifiers
are of an extremely stable type.

The azimuth and elevation position data are read out, each assigned
a single channel, through the sail analog subcommutator.

7.11.2   Error-Signal Monitors

The dc-voltage levels of the inputs to the azimuth and elevation
PWMs are each read out through a channel of the sail subcommutator.
The conversion tables (TN68-16) translate the binary data into
error-voltage readout. In the event of excessive in-flight pointing
errors, analysis of error-signal voltage data would aid in determining
the causes of the problem.

7.11.3   Motor-Current Monitors

The average currents supplied to the azimuth and elevation torque motors are each read out through a channel of the sail subcommutator. The data from these monitors are used to study the long-term in-flight performance of the azimuth and elevation bearings, and miscellaneous torques operating against the servo system. These may be friction, balance, flex-cable, or oscillatory anomalies.

7.12   LAUNCH SEQUENCE SYSTEM FUNCTIONAL DESCRIPTION

The OSO launch sequence system performs the following:

(1)   Sequences the unlatching of mechanical assemblies, which are stowed or locked during launch

(2)   Supplies control signals which activate electrical and control functions required during observatory orbital injection

Figure 7-18 is a block diagram of the launch sequence system. With zero-time starting at observatory separation from the launch vehicle, the spacecraft controlled launch sequence functions occur in the order indicated on the figure. These launch sequence functions were discussed in subsection 7.3.

Several requirements are placed on the Delta launch vehicle before the OSO launch sequence events can occur. First, the final spin rate achieved by the second-stage/payload combination before separation must be greater than 80 rpm. Second, the launch vehicle must sequence its separation from OSO after spinup. Finally, the OSO spin axis must lie within five degrees of normal to the solar direction at separation and the OSO center of gravity should be translating in a circular orbit of about 556 km (300 nm) altitude and a 33-degree inclination to the equator.
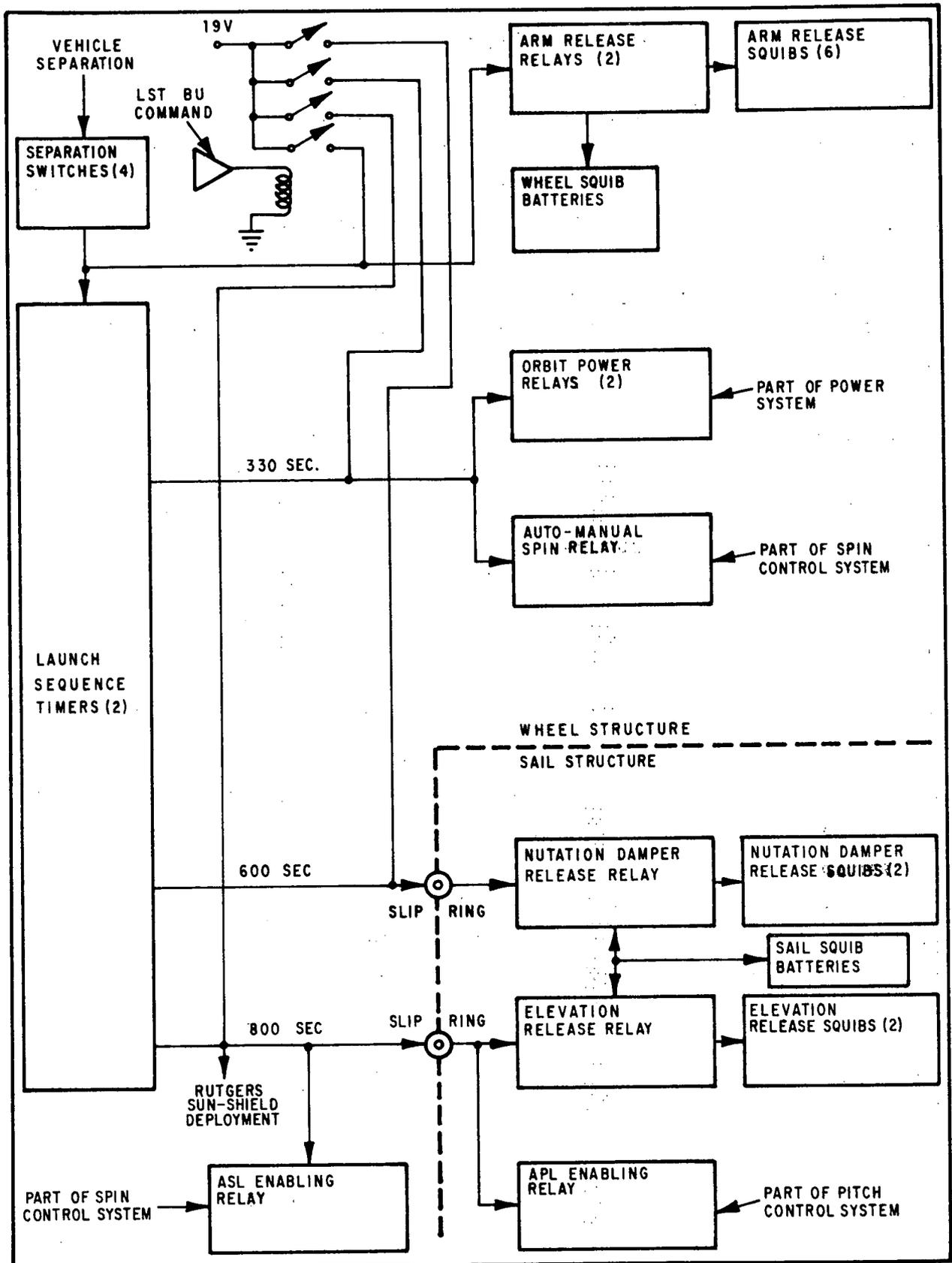
Fig. 7-18   Launch Sequence System Block Diagram

7.12.1   Launch Sequence Timers

Each of the two launch sequence timers consists of a constant
speed drive motor which drives a cam-laden mandrel (Fig. 7-19).
The cams on the mandrel make momentary contact with wire brushes,
thereby providing 15-second closure signals for sequencing events.
A dc voltage regulator contained within the timer drives the dc
motor at a speed of approximately 4,800 rpm. By means of gear
reduction, the motor drives the mandrel through one rotation in
about 1,600 seconds.

Once a separation switch actuates, the timer motors are supplied
power through the timer-run relay (latching type) and the timer-
run contacts internal to the timers. The timer motors then run
until the timer-run contacts open at a nominal 1,450 seconds. The
fastest-running timer then shuts off the power to both timers.
To reset the timer to zero time during ground tests, a timer-reset
contact is provided which supplies power from the test console
to the timers until zero-time is reached.

Each timer motor consumes about 40 ma of current. The tolerance
of the timing sequences is +5 percent. . Due to the large variance
in friction with temperature, it is extremely difficult to main-
tain timer accuracy over a wide temperature range. To partially
compensate for speed change due to temperature, the voltage
regulator is designed to provide a higher voltage to the motor at
low temperatures.
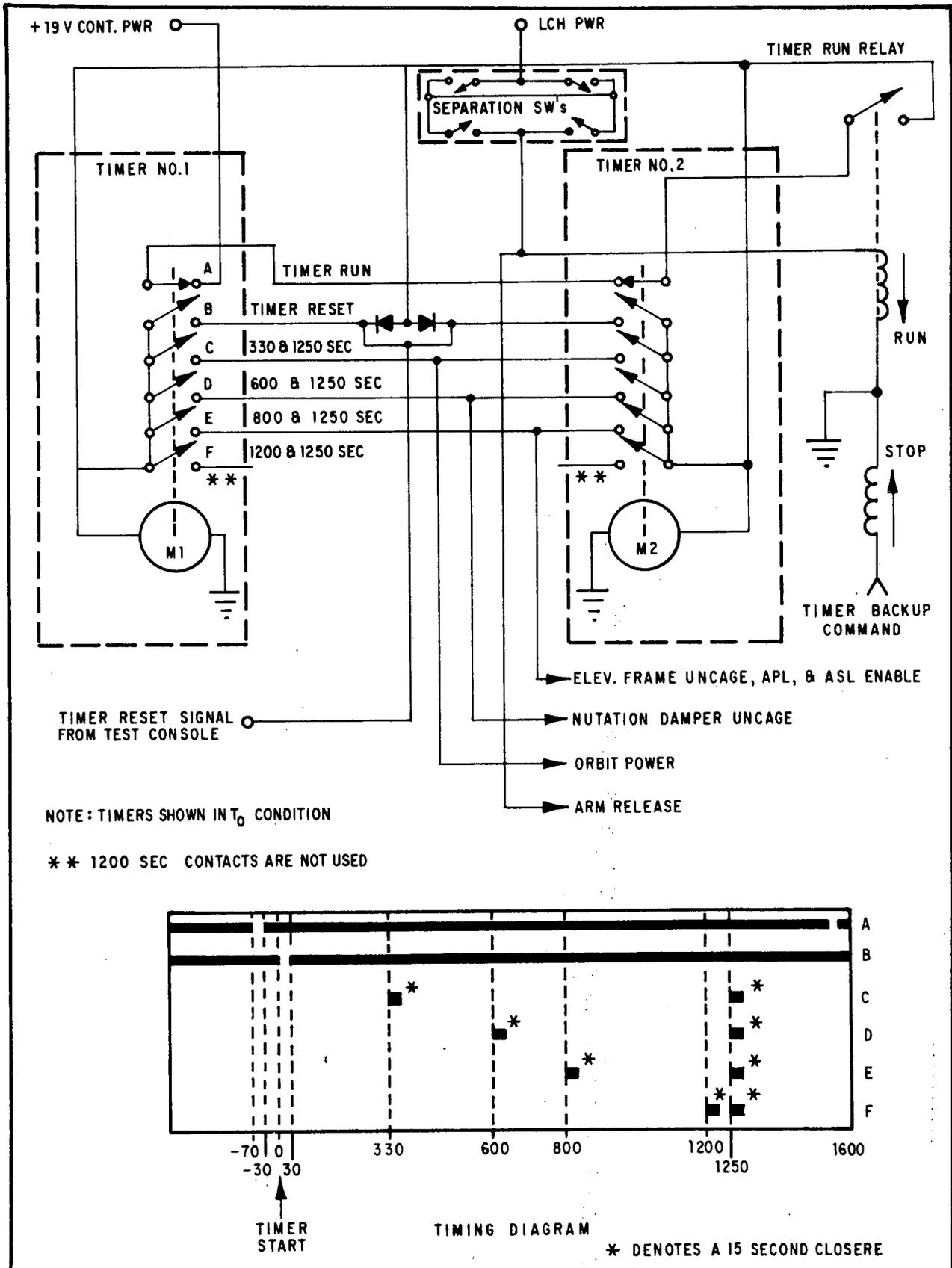
Fig. 7-19 Launch Sequence Timer Schematic Diagram

7.12.2   Pyrotechnic Pin-Pullers

The five pyrotechnic pin-pullers used on the OSO-6 spacecraft are as follows:

(1)   1 through 3 - Arm release (1 on each arm)

(2)   4 - Nutation damper bob release

(3)   5 - Elevation frame release

Each pin-puller consists of a latching block, two pressure cart-ridges (squibs), and a latching pin. The purpose of the pin-pullers is to release the various mechanical components stowed or constrained by latching pins during the launch phase. The detonation signals originate from the separation switches and the launch sequence timers.

Arm Release Pin-pullers. When in the launch state, the space-craft arms are folded down alongside the wheel and a folding linkage extends under the spacecraft. A latching block is used on each arm to pin the linkages to the wheel. The latching pins are pulled simultaneously at observatory separation from the launch vehicle, and the arms are then deployed to their extended or orbit condition by the centrifugal force exerted by the spinning wheel.

Power for the arm release pin-pullers is supplied through two parallel relays with contacts rated at 10 amps. These relays switch power from the two wheel squib batteries to fire the six (two on each arm) arm release squibs. Each squib has two filaments that are fused separately so that short circuits to ground, as a result of firing, do not drop the voltage available to fire the other circuits. Also, the circuit design is such

that normal functioning of either relay, either battery, or either squib on each arm will cause the three arms to deploy. A typical squib firing circuit is shown in Fig. 7-20.

Nutation Damper Bob Pin-Pullers. The nutation damper bob is secured during the launch phase by two spring loaded cones pinned by a latching block. The latching pin is pulled at T + 600 seconds, and the bob is then allowed to move freely.

The T + 600 seconds timer signal (15 seconds duration) is fed to the nutation damper release relay through a slip ring. This relay also has contacts rated at 10 amps and is used to fire two squibs in the nutation damper latch block. In this case, the power is provided by the two sail squib batteries. Again, each of the four filaments (two per squib) are individually fused.

Elevation Frame Release Pin-Puller. The elevation frame is latched to the azimuth frame during the launch phase by the elevation latching block. A T + 800 seconds, the latching pin is pulled, and the pointed instruments are allowed to move in elevation.

The T + 800 second timer signal (15 seconds duration) is also supplied to the sail through a slip ring. In the sail, this signal actuates a relay, which in turn fires the elevation release squibs. The elevation release design is similar to that of the nutation damper described above.
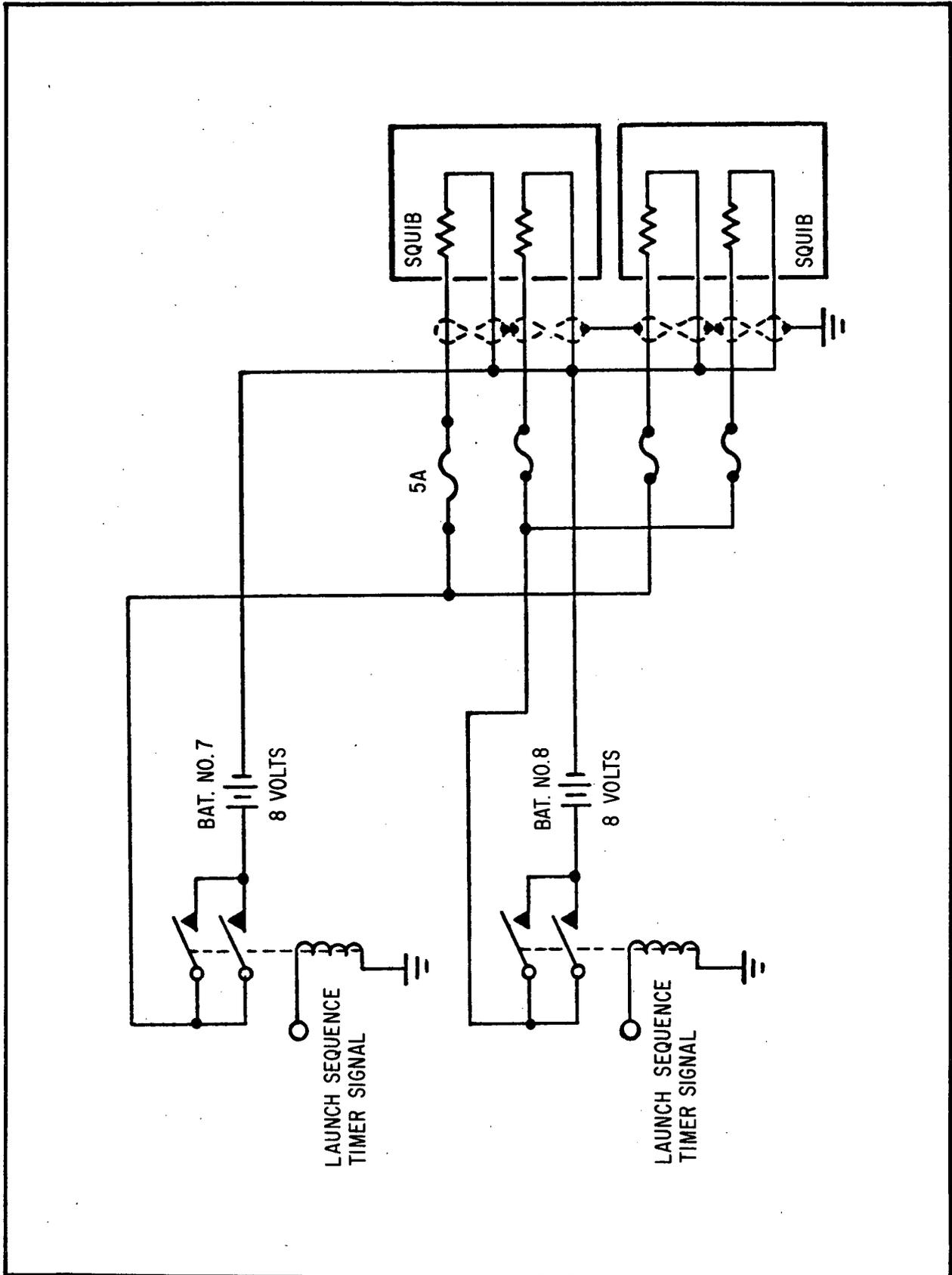
Fig. 7-20 Typical Squib Firing Circuit

7.12.3   Squib Batteries

There are two redundant wheel squib batteries whose only function is to fire the arm release squibs. When connected to the bridge wires of the squibs, each battery is loaded with a 1/4-ohm load.

Each wheel squib battery consists of two parallel strings of 1/2 C-size nickel-cadmium cells with seven cells in series in each string. The battery is capable of supplying 20 amp into a 1/4-ohm load for two seconds.

There are two sail squib batteries; their function is to fire the squibs that uncage the nutation damper and the elevation lock. Each sail squib battery consists of two parallel strings of three 1/2 C-size nickel-cadmium cells connected in each string. At 600 seconds and again at 800 seconds, each sail squib battery is loaded with 1/2-ohm. Each battery is capable of providing 6 amp into a 1/2-ohm load for 2 seconds.

7.13   NUTATION CONTROL SYSTEM FUNCTIONAL DESCRIPTION

During the orbit life of the OSO spacecraft, many external torques act on the spacecraft structure. These torques are caused by the earth's unsymmetrical gravity field, interaction with earth's magnetic field, solar radiation pressure, aero-dynamic drag, elevation drive torques, and pitch corrections by the pitch pneumatic system. All these torques can cause the spacecraft spin axis to precess in pitch and produce nutation of the spin axis. However, only the reaction of the pitch jets and the elevation drive torques have any significant effect because of the high impulses they produce.

Another important nutation source is the initial nutation imparted to the spacecraft when the second-stage motor is separated after

orbit injection. This nutation is attributed to unsymmetrical thrust of the separation spring and to unsymmetrical motion of the arms as they are deployed into position.

Because the primary mission of the spacecraft is to provide a stable platform from which the pointed experiments can operate and also to allow the wheel experiments to uniformly sweep the celestial sphere, a passive nutation control system is used to damp any nutation that the spacecraft may acquire. To summarize, the spacecraft is "put to sleep" after the following events have occurred:

(1) Orbit injection (to dissipate any nutation produced by separation, arm deployment, initial solar acquisition of the pointed experiments and initial pitch corrections)

(2) Orbit morning (to dissipate the nutation produced by elevation motor torque)

(3) Pitch correction (to dissipate the nutation produced by pitch jet activation)

The nutation control system consists of a tuned nutation damper (Fig. 7-21) mounted in the sail. The damper consists of a pendulum bob supported by a wire and immersed in a bath of silicone oil. This type of damper is termed a passive device because it is driven by the nutational motion which it eventually damps out. This means that it has good reliability, simplicity of operation, and requires no external power source.

The nutation damper pendulum consists of a 0.8 pound spherical bob suspended by a steel wire with a length of about 2.9 inches and a diameter of 0.020 inches. The exact length of the pendulum wire is adjusted to suit the mass properties of the spacecraft.

SILICONE OIL

HOUSING

WIRE

CAGING SEAT

BELLOWS

BOB

CAGING PLUNGER

LATCH BLOCK

SQUIB ACTUATED PISTON
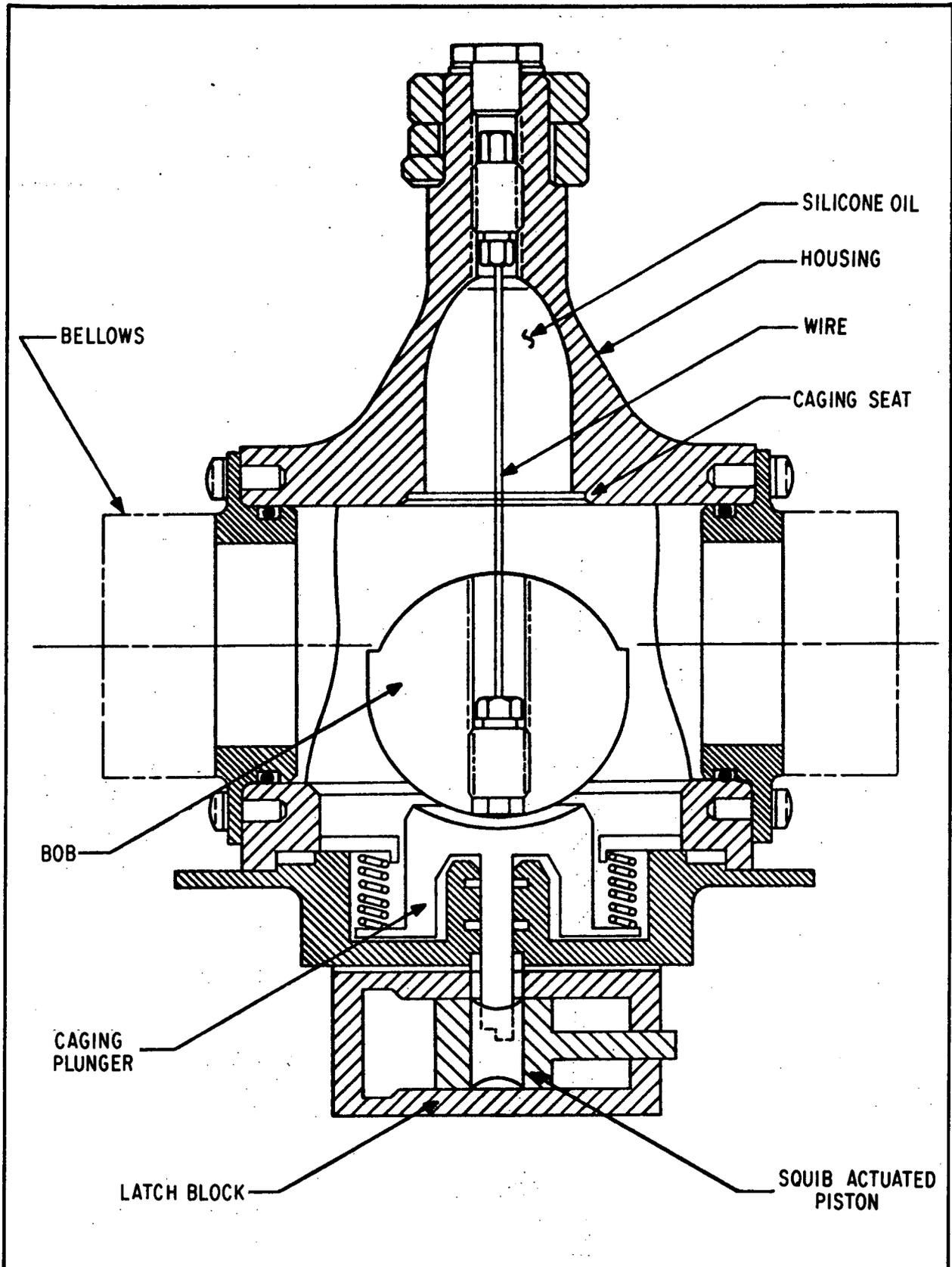
Fig. 7-21   Nutation Damper

The main body of the nutation damper and the bellows contains
silicone oil. The bellows accommodate any volume change
in the oil due to temperature changes, and thereby maintain a
positive pressure upon the oil.

The bob is caged with the squib-actuated caging plunger in the
up position during the test, transportation, and launch phases.
The bob is uncaged at T + 600 seconds (spacecraft time) by the
firing of the squibs in the latch assembly. The spring forces
the caging plunger to the down position and frees the bob. The
bob is then able to oscillate in the oil bath with a 2-degree
freedom of movement.

The nutation damper absorbs nutational energy by converting the
kinetic energy of the pendulum bob into heat energy as the bob
moves in the oil bath. The heat energy is a result of fluid
friction and is dissipated by the nutation damper casting.
Monitoring of nutation damper performance is accomplished by
analyzing pitch readout data.

The design damping range of the nutation damper is approximately
0.45 to 0.55 rps although the damper is effective over a much
wider range than this.

Section 8
POWER AND DISTRIBUTION DESCRIPTION

8.1   INTRODUCTION

The OSO power system consists of the power supply, power control, and distribution circuits. The power supply consists of a solar-cell array assembly and a battery assembly. The power control is provided basically by an undervoltage switch and the day-night turn-on circuits. The power distribution consists of the wiring and relays that connect the electronic equipment to the various power buses.

A simplified functional diagram of the power system is shown in Fig. 8-1. This diagram shows which "black boxes" are powered from the various power bases and also shows the commands that operate the power control relays and the location of the protective fuses.

8.2   POWER SUPPLY

Electrical power for the spacecraft is derived from sunlight by a solar-oriented array of silicon solar cells (Fig. 8-2). Part of this power is used to operate the spacecraft and experiment electrical components during the day part of the orbit. The balance of the solar array power is used to charge a storage battery which stabilizes the voltage, supplies peak demands, and provides power during the dark part of the orbit. The solar array produces about 40 watts which allows a comfortable power margin to meet the observatory load requirements.
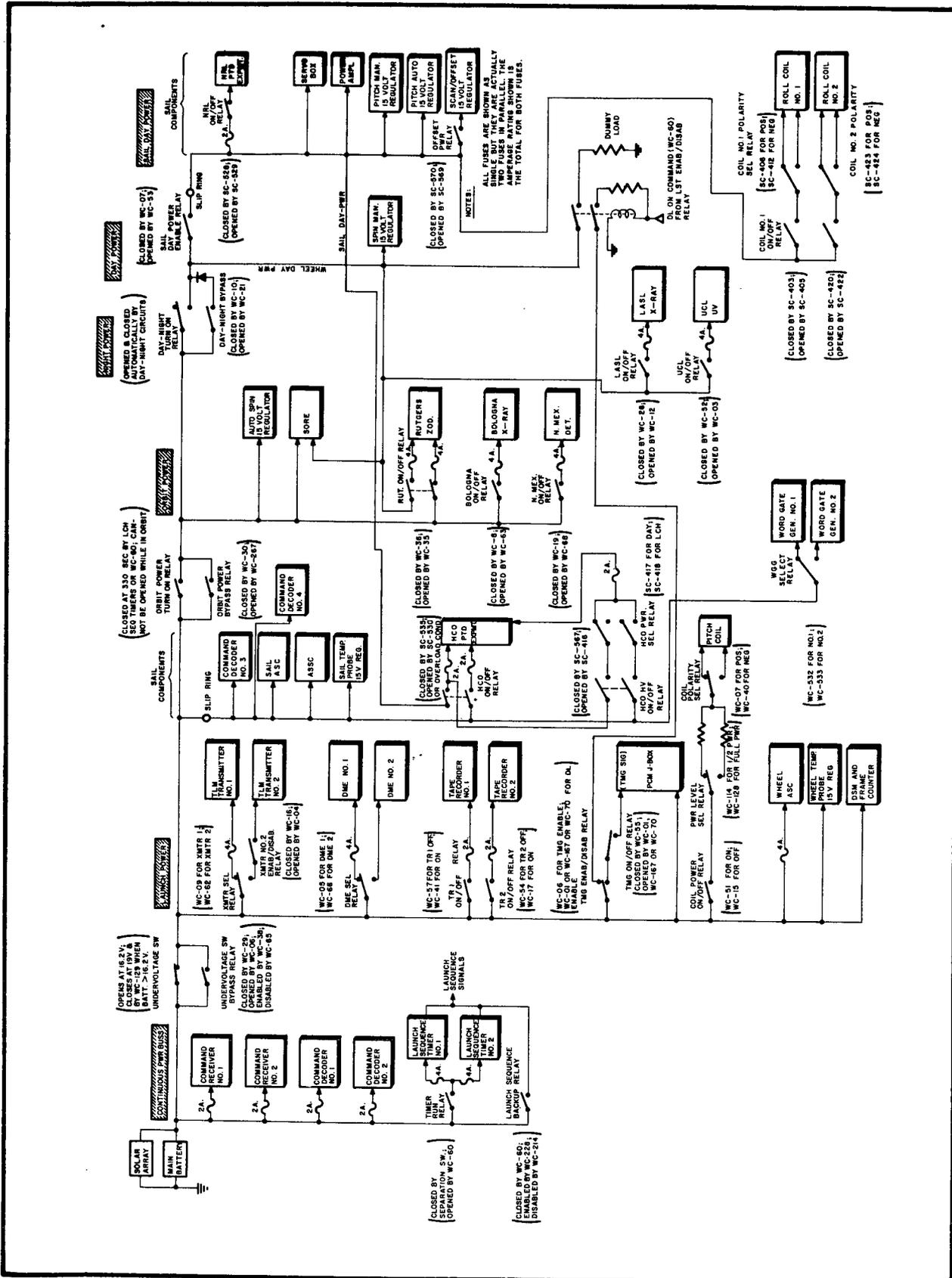
Fig. 8-1   OSO-6 Power Distribution Diagram
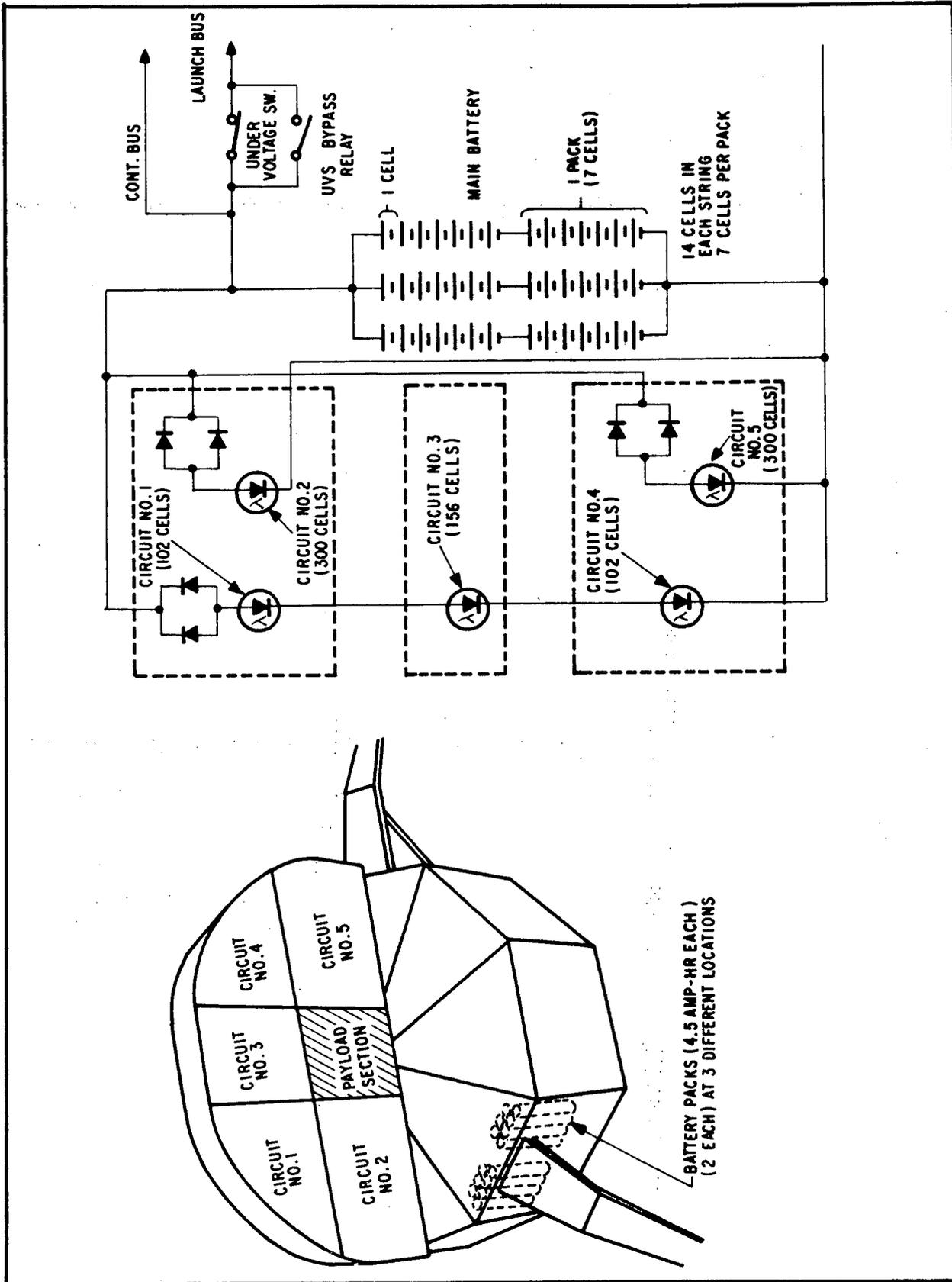
Fig. 8-2 Main Battery and Solar-Cell Array

The greatest demand on the power supply occurs at each satellite dawn. During the 40 to 60-second acquisition period there is nearly a three-ampere load on the power supply.

During the day, the main battery is charged at a rate of about 0.8 amperes. The charge rate is highest immediately after acquisition (about one amp) due to low battery voltage and low temperature of the solar array. During satellite night, the main battery is discharged at a rate of about 0.5 amperes for a maximum of 35 minutes.

8.2.1   Solar-Cell Array

The solar array consists of flat mounted N-P type silicon solar cells mounted on three panels (see Fig. 8-2). Flat mounting of solar cells requires less cells, fewer interconnections, and lower cost. The array output response was designed with margin for errors in temperature predictions. This placed the maximum power point near the higher end of the anticipated range of OSO operating voltages. Thus, a shift in the response curve caused by higher-than-predicted temperatures will not cause output current to fall sharply.

Solar Cells. The Spectrolab N-P type cells (Fig. 8-3) have five tapered solder grids on the sunlight side to collect current with a minimum amount of power loss from resistance. The back side of the cell is prepared to take solder by means of an electrolysis plating method. The response characteristics of the solar cells are shown in Fig. 8-4.

Solar Array Wiring. The 960 cells, most of which measure 2 by 2 centimeters, are arranged in five circuits (see Fig. 8-2). The center panel supports one circuit matrix (circuit No. 3) of six cells in parallel by 26 cells in series. This is connected in series with circuits No.'s 1 and 4 each of which have six cells
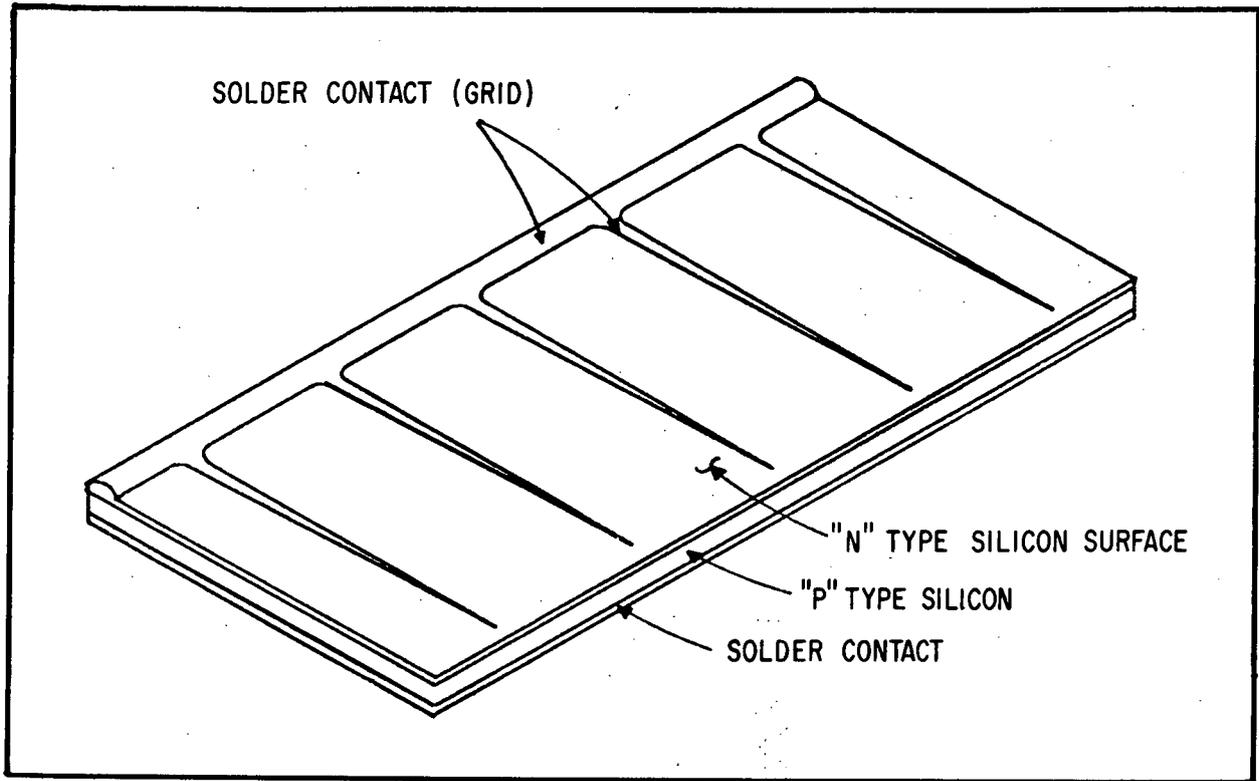
Fig. 8-3 N-P Silicon Solar Cell
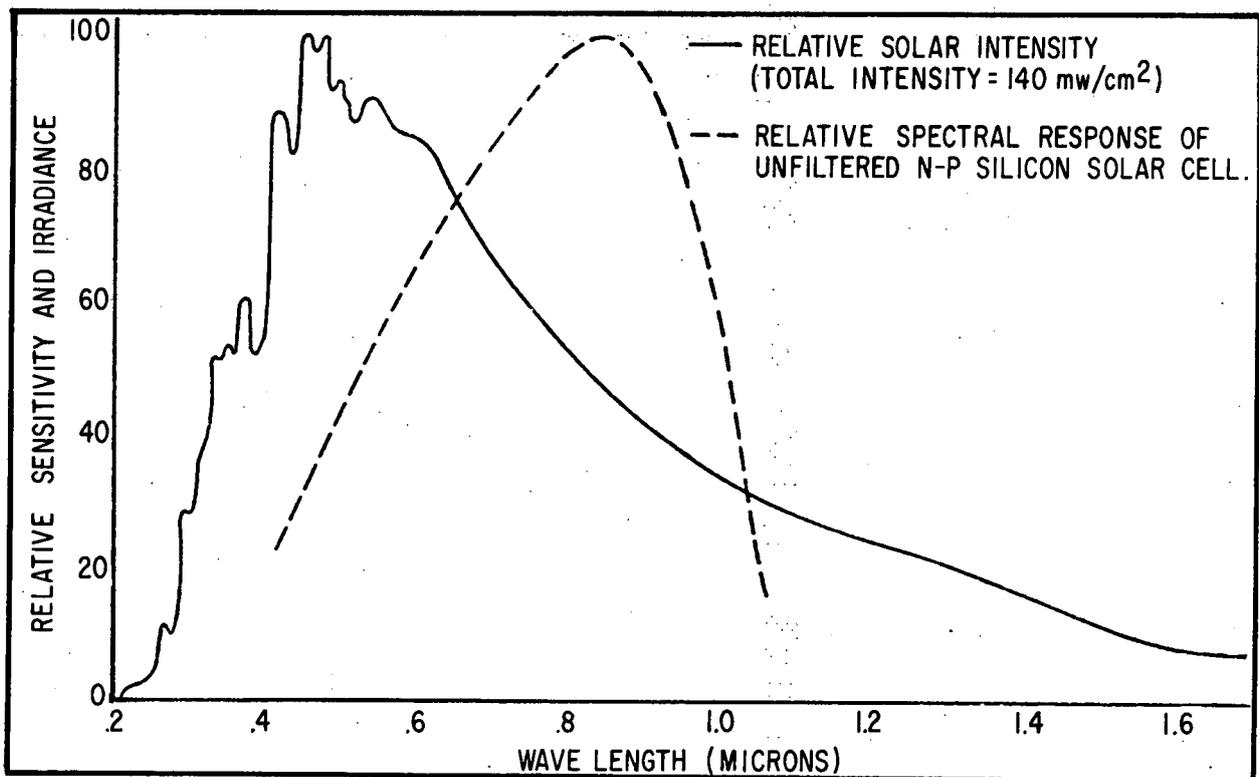
Fig. 8-4 Solar-Cell Response

in parallel and 17 in series. On the bottom of the side panels
there is a circuit matrix of five cells in parallel by 60 cells
in series. Mechanically, the solar cells are flat mounted
modules of five or six cells per module (except along the rounded
edges of the solar array substrate when some of the modules are
broken into smaller blocks.) The modules include the necessary
interconnecting contact strips between cells on the module. Each
series circuit is isolated by means of two blocking diodes con-
nected in parallel.

Cover Glass. Each cell has a 30-mil thick fused silica cover
glass which is cemented to the cell with a silicone-based ad-
hesive. On the face of the cover glass is an anti-reflective
coating and on the side nearest the cell is an ultraviolet re-
flective coating. The ultraviolet reflective coating has a
cutoff point at about 4 microns. This is slightly higher than
the shortest wavelength to which a cell will respond. The anti-
reflective coating has a cutoff point of 0.625 microns. By
means of these coatings and the use of proper thermal coating
on the back side of the panels, the orbital temperature of the
array is kept below 70°C.

Solar Array Tests. The most severe environmental test for the
solar array is a thermal shock test. The greatest stresses
occur during this test between the intercell connections due to
the difference in thermal expansion between the cells and the
panels on which they are mounted. The solar array is, therefore,
subjected to 20 thermal shock cycles. Each cycle consists of
about nine minutes in a cold chamber at 45°C and about nine
minutes in an oven at 80°C. This test is performed before power-
output measurements are made.

Power-output measurements are made on each circuit of the solar array with the panels operating at orbital temperature in order to minimize errors that might occur from extrapolating from a lower temperature.

With a panel oriented so that it is perpendicular to the sun, I-V curves of each of the circuits on that panel are made together with measurements of the solar intensity by means of a standard cell under collimated and uncollimated conditions.

Other tests are also conducted as part of the OSO acceptance test program. One of these tests consists of measuring the current output of the array at different voltages and temperatures, and the results are adjusted for nominal space conditions. The I-V power curve for a typical solar array is shown in Fig. 8-5.

The maximum-power point for the solar array is about 1.95 amp at 20 v. A voltage drop across the solar-array blocking diodes is between 0.7 and 0.8 v and the drop across the charge-current monitor can be as high as 0.3 v. The voltage at the battery terminals, then, is about 19 v. It is this voltage (19 v) which is used in all calculations, and which is plotted in Fig. 8-5.

8.2.2   Main Storage Battery

The main battery is charged by the solar array and effectively limits the maximum supply voltage to 21 v. It also provides power during launch, during night, and for surges during the day.

AREA OF NORMAL OPERATION

AREA OF EXTREME OPERATION

MAX POWER POINT

UVS CUTOFF VOLTAGE

BATTERY OVERCHARGE
VOLTAGE LEVEL
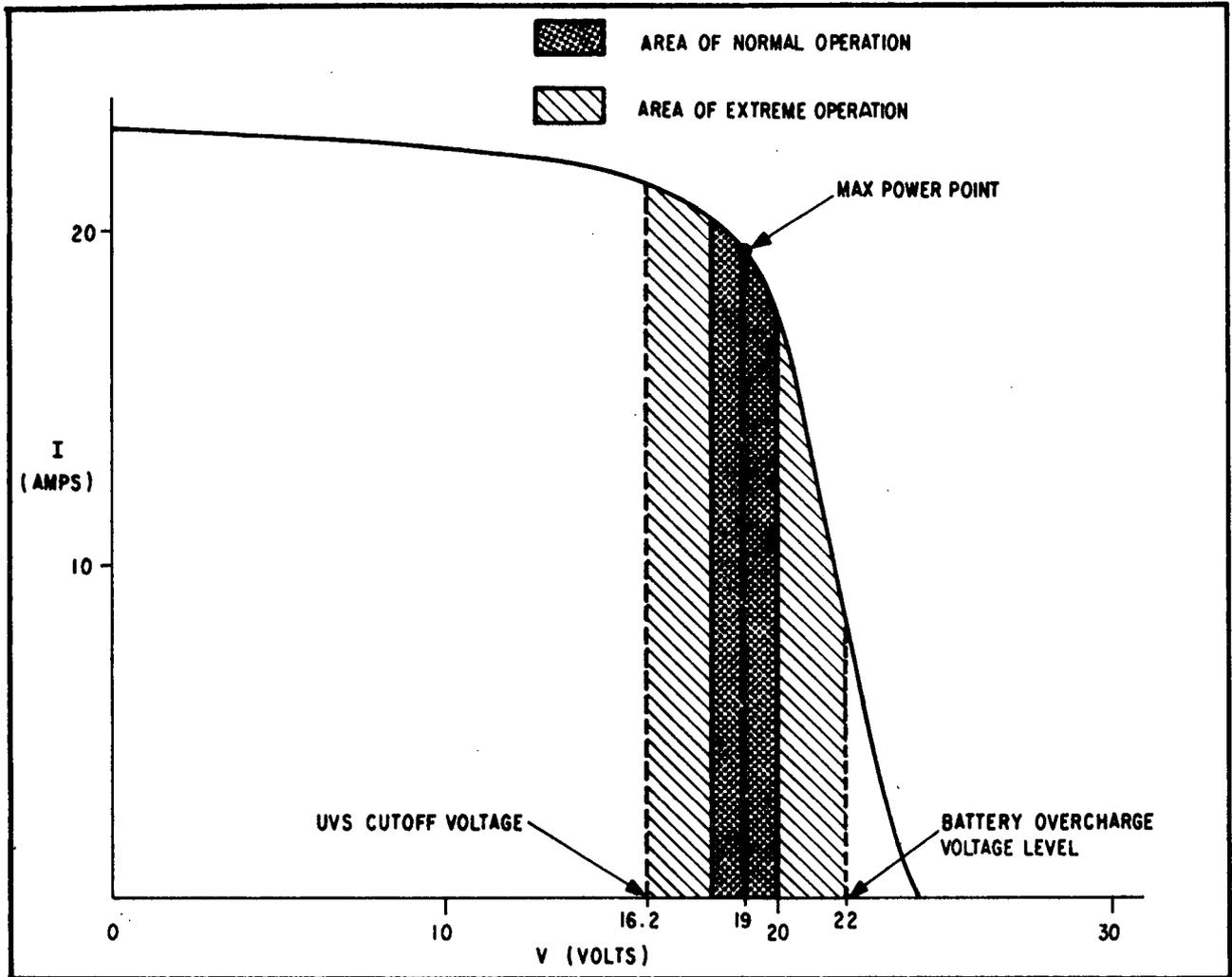
Fig. 8-5   Solar Array Power

The cell selected for use in the main battery is a Sonotone Type
F hermetically sealed nickel-cadmium (Nicad) cell, with two plates
32-mils thick and a poly-propylene separator. This cell has a
measured capacity of at least 4.2 amp hr above 1.2 v at room
temperature. It can stand continuous overcharge of 0.45 amp
with a maximum voltage of 1.48 v.

The battery consists of 42 cells arranged in three parallel
strings of 14 cells each. Each string consists of two packs
of seven cells each. At time of launch, the battery capacity is
about 13 amp hr above 16.8 v. It can stand a continuous

overcharge of at least 1.35 amp. for additional safety margin, however, 0.6 amp was used as a maximum overcharge rate for system-design calculations.

In normal operation, the battery is cycled to a discharge depth of about 3-1/2 percent. If the solar-cell array should fail, the battery alone could power the basic spacecraft for about eight orbits at the time of launch.

8.3   POWER CONTROL AND DISTRIBUTION

The power control and distribution circuits provide for operation of the spacecraft in variously-powered modes, such as the day, night, and undervoltage modes.

As shown in Fig. 8-1, all power control is exercised by ground command, with the exception of the undervoltage switch, the orbit power relay, and day-night switching. Orbit power turnon is normally controlled by the launch-sequence timer (Section 7) and is irreversible in flight. Day-night power switching is incorporated into the spin-control system and is also discussed in Section 7.

OSO has a command-controlled dummy load, which can be used to prevent excessive battery overcharging. This component is described below.

8.3.1   Undervoltage Switch Operation

An undervoltage switch (UVS) is used on OSO to prevent an excessive power consumption problem from depleting the batteries

to the point of no return. This point would be that at which it would be impossible to execute corrective commands. (The voltage threshold of the command decoders has been shown, in engineering tests, to be below 15 volts.)

When the spacecraft is in the undervoltage state, the sail can no longer be solar oriented, and the solar array therefore charges the batteries in a pulsating dc fashion (at about a 25 percent rate) until the voltage rises to about 19.1 v. At this time, power is reapplied to the observatory, and normal functions can be resumed. Such an undervoltage condition could be caused by battery failure--but it is more likely to be caused by some excessive current demand due to an electronic circuit failure.

The undervoltage switch consists of a voltage sensing circuit that automatically opens the UVS relay (Fig. 8-1) when the battery voltage drops to 16.2 v and that closes the relay when the voltage rises to 19 v.

A bypass relay can be commanded to bypass the UVS circuits. Because of the inherent danger of using this bypass capability, it is used only when the automatic undervoltage circuit has failed in an open condition. The danger in closing the UVS bypass when it is not needed is that the protection offered by the UVS is no longer effective. If battery voltage drops below the command-system operation level, there is no way to remove the excessive load from the power supply. The battery, therefore, cannot be recharged to its operating level.

The UVS bypass command closes the undervoltage switch. It may therefore be followed immediately by the "not bypassed" command and if battery voltage stays above 16.2 v, the under-voltage switch will remain closed. In this way, power may be reapplied to QSO safely before the battery is fully recharged.

8.3.2 Dummy Load

The dummy load feature on OSO-6 consists of a 90-ohm load resistor and a parallel circuit consisting of a control relay and a feedback resistor. The total resistance is about 75 ohms and this load is applied or removed by ground commands. The resistor forms a holding loop on the dummy load on/off relay. The dummy load is connected to the day power bus and, therefore, only diverts charge current from the solar array and thus prevents battery overcharging.

Battery overcharging, while not desirable, can be tolerated up to a limit of approximately 600 milliamperes continuously. Therefore, excessive overcharging poses no serious problem unless some of the experiments are turned off. In this event, the batteries may be subjected to overcharging in excess of 600 milliamperes. Charging above this level has the adverse effect of decreasing battery life.

Therefore, the dummy load is normally used early in the flight since the solar-cell array output is high during this time because none of the anticipated degradation factors have yet come into effect. Also, later in the mission it is possible that some of the scientific payload will have been turned off, and the dummy load is then used to bleed off the resultant surplus power.

BATTERY VOLTAGE VERSUS DUMMY LOAD CURRENT AND POWER

Battery Voltage (v)	Load Current (ma)	Power (w)
22	292	6.4
21	278	5.8
20	265	5.3
19	250	4.7
18	238	4.3
17	225	3.8

8.4 SOLAR ARRAY OUTPUT AND OBSERVATORY POWER BUDGET

Table 8-1 shows the predicted solar array output in ma-hr per orbit at 19 volts for 6 month increments of orbit life. The output at launch was calculated to be about 1980 $\frac{ma-hr}{orbit}$ and degradation was predicted to be a maximum of two percent during the first 6 months and a maximum of two percent per year thereafter (the actual in-orbit performance is described in Section 2).

The solar-array output prediction is influenced by many factors, including:

- Preflight measurements have an uncertainty due to varying atmospheric attenuation

- Degradation of the array in flight due to solar-radiation damage

- Solar flux varies from the mean by ± 3.4 percent in the course of a year

- The daylight portion of the orbit was predicted to be between 60.1 and 70.9 minutes. The lower limit was used for power budgeting and thus represents a worst-case situation.

- The earth's albedo increases the array output every orbit morning and evening. The amount of the increase depends, to a degree, on spacecraft aspect

- Output from the solar array varies sharply with array temperature and load voltage

Actual power availability is also influenced by variations in battery-charging efficiency, which can be expected to degrade with age.

Table 8-1

OSO-6 PREDICTED SOLAR ARRAY OUTPUT

Orbit Period	Radiation Degradation (%)	Output Energy ($\frac{ma-hr}{orbit}$)
Launch	0.0	1980.0
6 months	2.0	1940.4
12 months	3.0	1921.0
18 months	4.0	1901.8
24 months	5.0	1882.8

The observatory power budgets for spacecraft systems and the experiments are shown in Tables 8-2 and 8-3, respectively. By using the worst case prediction of 1980 $\frac{ma-hr}{orbit}$ for solar array output at launch, it can be seen from Tables 8-2 and 8-3 that the predicted power pad at launch was about 1980-(1162 + 792) = 26 $\frac{ma-hr}{orbit}$. However, if a sunlight period of 70.9 minutes is used the power pad would be about 358 $\frac{ma-hr}{orbit}$.

The values in Tables 8-2 and 8-3 were computed before launch using the following assumptions and factors:

    (1)   A 96-minute orbit period (1.6 hour per orbit) was used and divided as follows:

$$96 \text{ min. orbit} \begin{cases} 60 \text{ min. day} = 62.5\% \\ 36 \text{ min. night} = 37.5\% \end{cases}$$

(2) The orbit day only energy (E_d) was computed as follows:

$$E_d = (0.625) \left(1.6 \ \frac{hr}{orbit}\right) I_d = I_d \ \frac{ma-hr}{orbit}$$

(3) The orbit night only energy (E_n) was computed as follows:

$$E_n = (0.375)(1.25) \left(1.6 \ \frac{hr}{orbit}\right) I_n$$

$$= 0.75 \ I_n \ \frac{ma-hr}{orbit}$$

where the 1.25 term represents the battery loss factor.

Table 8-2

OSO-6 SPACECRAFT POWER SUMMARY

Subsystem	Assemblies	Day Power		Night Power	
		milliwatts	ma-hr per orbit	milliwatts	ma-hr per orbit
Pointing Controls	Servo Amplifier	1,710.0	90.0	190.0	7.5
	Power Amplifier	3,420.0	180.0	- - - -	- - -
	Pitch Readout	57.0	3.0	- - - -	- - -
	Raster	1,026.0	54.0	380.0	15.0
Spin Control	Spin Box	1,330.0	70.0	855.0	34.0
Attitude	SORE	266.0	14.0	152.0	6.0
Monitor	Magnetometer	209.0	11.0	209.0	8.3
Telemetry	PCM (DME & 2 ASC)	1,235.0	65.0	1,235.0	49.0
	DSM	551.0	29.0	551.0	21.7
	PCM Junction Box	361.0	19.0	361.0	14.8
	Tape Recorder (1)	1,837.0	96.6	1,837.0	72.5
	Transmitter (1)	2,242.0	118.0	2,242.0	88.5
	Terminal Boards	95.0	5.0	95.0	3.8
	ASSC	76.0	4.0	76.0	3.0
	Word Gate Generator	975.0	25.0	975.0	18.8
Command	2 receivers/4 decoders	304.0	16.0	304.0	12.0
Power	Charge monitor current	142.5	7.5	- - - -	- - -
	TOTAL	15,836.5	807.1	9,462.0	354.9

Summary

Night current: 354.9 ma-hr per orbit
Day Current: 807.1 ma-hr per orbit
Total Spacecraft Current: 1162.0 ma-hr per orbit

Table 8-3

OSO-6 FLIGHT INSTRUMENTS ACTUAL POWER BUDGET

| Experiment | Day Bus | | Night Bus | | Orbit Bus | | | Exmt. Current Subtotals |
	pwr (mw)	ma-hr/orbit	power (mw)	ma-hr/orbit	power (mw)	ma-hr/orbit Day	ma-hr/orbit Night	ma-hr/orbit
Pointed								
HCO	2163	113.8	-0-	-0-	182.0	9.6	7.3	130.7
NRL	3460	182.1	-0-	-0-	-0-	-0-	-0-	182.1
Wheel								
Rutgers	779	41.0	-0-	-0-	1482.0	78.0	58.5	177.5
LASL	1129	59.4	-0-	-0-	-0-	-0-	-0-	59.4
Bologna	-0-	-0-	-0-	-0-	832.2	43.8	32.8	76.6
UCL	367	19.3	-0-	-0-	-0-	-0-	-0-	19.3
UNM	-0-	-0-	-0-	-0-	1558.0	83.6	62.7	146.3
Totals	7898	415.6	-0-	-0-	4054.2	215.0	161.3	791.9

8.5   POWER SYSTEM DIFFERENCES

8.5.1   OSO-4/HCO Instrument Conversion

In order to fly the HCO/OSO-4 flight spare instrument on
OSO- 6, an adapter was installed on the HCO instrument to
redistribute the interface wiring. This adapter is called the
"K-Box". In addition to matching the OSO-4 and 6 interfaces,
the K-Box provides the filtering originally contained in the
HCO filter choke module and contains the relay and wiring
required to operate the overload protection (OP) circuit which
connects at the rear of the K-Box.

The OP circuit monitors the HCO high voltage power supply
current. If a high voltage malfunction is indicated by a
change in current, the OP circuit automatically actuates the
"HCO off" command to disable the high voltage power supply.

8.5.2   Flex Cable Modifications

The OSO-5 flex cables consisted of printed copper paths
thermally bonded between layers of mylar with 37 pin connectors
soldered and potted on the ends. The low melting point of
mylar caused these cables to be susceptible to heat damage
during soldering operations. Furthermore, the flat printed
wire was susceptible to damage by handling during routine
spacecraft operations.

The OSO-6 required additional circuits across the elevation
axis. In order to reliably provide the additional circuits
and maintain flexibility across the elevation axis, a different
type of cable was selected.

The OSO-6 flex cables consist of shielded and unshielded stranded wires embedded in a flat ribbon of silicon rubber material with 0.070 in. centers and "twist-pin" connectors. The resulting cable is more flexible and durable than mylar cables and does not have the melting problem during solder operations. Refer to Para. 2.3 of this report for a discussion of the in-orbit performance of these redesigned flex cables.

Section 9

TELEMETRY SYSTEM DESCRIPTION

9.1   INTRODUCTION

The spacecraft and experiment data is sampled, encoded, and telemetered to the GSFC Satellite Tracking and Data Acquisition Network (STADAN) system by means of an on-board pulse code modulation/phase modulation (PCM/PM) telemetry system. This system consists basically of a data handling subsystem and a RF subsystem (Fig. 9-1). The data handling subsystem provides communication (sampling), multiplexing (time sharing), encoding, and on-board storage of the observatory data. The RF subsystem provides RF transmission of both real-time and stored data to the STADAN stations.

The OSO-6 telemetry system is very similar to OSO-5's. The main differences consist of: the addition of the word gate generator, changes in the sail J-box, decreasing the bandwidth of the transmitter spectrum, and changing the transmitter frequency to 136.71 MHZ.

9.2   DATA HANDLING SUBSYSTEM

9.2.1   General

The data handling subsystem provides the interface between all observatory data sources and the RF subsystem. Its primary function is to assemble the observatory data into a form suitable for transmission and to store these data for playback when the observatory is over a selected STADAN station.
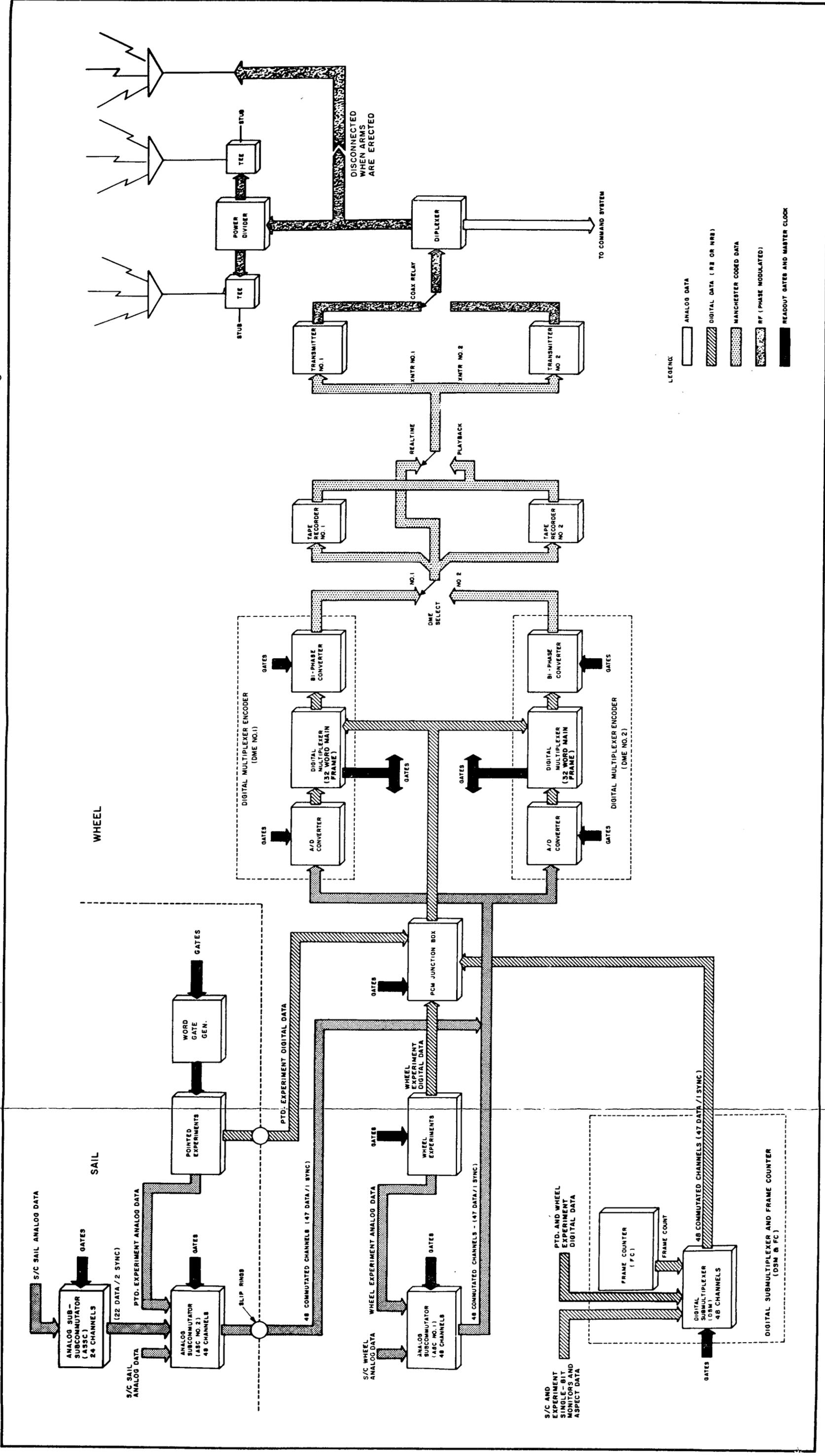
Fig. 9-1 OSO-6 Telmetry System Block Diagram

The data-handling subsystem uses time multiplexing techniques wherein each data source is assigned a time interval (channel) for transmission of the parameter being measured. This source data can be in either digital or analog form.

The OSO-6 telemetry system provides 26 channels of a 32-channel main-frame format for experiment prime (digital) data. Most of the spacecraft status and performance data are in analog form, and the system provides 47 data channels in each of the two 48-channel analog subcommutators (ASC's) and 22 data channels in the 24-channel analog sub-subcommutator (ASSC) for this type of data. Some of the analog channels are used for experiment performance data. A 48-channel digital sub-multiplexer (DSM) provides additional digital channels for aspect, spacecraft housekeeping, and experiment data. There is a total of 196 telemetry channels (184 data, 7 sync, and 5 spares) available on OSO-6. Additional submultiplexing is done internally in some of the experimenter's instruments.

The multiplexed PCM signal has a bit rate of 800 bps. Each of the 32 main-frame channels (or words) contains eight bits. Two words (16 bits) are used for a main frame synchronization code, one word is assigned to each of the two analog sub-commutators (ASC), one word is assigned to the digital sub-multiplexer (DSM), and the remaining 27 words are assigned to the experiment instruments. The two ASC's, the ASSC, and the DSM subframes contain their own synchronization codes.

All signals into the two redundant digital multiplexer encoders (DME) are connected in parallel, and the internal circuit design is such that failure in one will not affect the other. The redundant DME output is selected by the same ground station commands that control DME power switching.

The experiments have self-contained storage shift registers for their digital data. The spacecraft data handling subsystem provides gate pulses to shift the stored data into the telemeter at the assigned time interval. The assignment of words on OSO-6 for both digital and analog channels is shown in Fig. 9-2.

Data are stored by redundant, continuous-loop, magnetic tape recorders which have the capacity to record data for 100 $\pm$ 2 minutes. Either of the two recorders is selectively powered in response to ground station command and, upon receiving a "playback command", the selected recorder reproduces the recorded data at 18 times the realtime rate (14,400 bps). The recorder automatically reverts to the record mode at the end of the playback.

9.2.2 Digital Multiplexer and Encoder

The digital multiplexer and encoder (DME) assembly provides the following five basic functions:

(1) Generates telemetry master clock (800 Hz)

(2) Generates gate pulse signals for multiplexing digital and analog signals

(3) Generates sync codes for main frame and ASC subframes

(4) Encodes the analog data into eight-bit digital words, and

(5) Provides signal conditioning to convert the multiplexed digital data into a biphase (Manchester) code.

TELEMETRY FORMAT

MAIN FRAME

1	NRL SAIL
2	NRL SAIL
3	HCO SAIL
4	HCO SAIL
5	RUTGERS WHEEL
6	BOLOGNA WHEEL
7	BOLOGNA WHEEL
8	NRL SAIL
9	NRL SAIL
10	BOLOGNA WHEEL
11	HCO SAIL
12	HCO SAIL
13	NRL SAIL
14	UNM WHEEL
15	UNM WHEEL
16	UNM WHEEL
17	UNM WHEEL
18	BOLOGNA WHEEL
19	HCO SAIL
20	HCO SAIL
21	LASL WHEEL
22	BOLOGNA WHEEL
23	UCL WHEEL
24	NRL SAIL
25	SAIL ANALOG SUBCOMMUTATOR
26	WHEEL ANALOG SUBCOMMUTATOR
27	HCO SAIL
28	HCO SAIL
29	DIGITAL SUBMULTIPLEXER
30	BOLOGNA WHEEL
31	SYNC
32	SYNC

WHEEL ANALOG SUBCOMMUTATOR (WASC)

WORD	MONITOR
1	UNM COUNT RATE DATA 1
2	UNM COUNT RATE DATA 2
3	UNM COUNT RATE DATA 3
4	UNM VOLTAGE MONITOR
5	UNM VOLTAGE / TEMP MONITOR
6	UNM ASSC ID
7	BOLOGNA HV MONITOR
8	BOLOGNA SPARE
9	BOLOGNA SPARE
10	BOLOGNA LV MONITOR
11	RCVR 1 AGC
12	SPIN RATE
13	BOLOGNA TEMP
14	BOTTOM SKIN TEMP
15	TEMP PROBE 15 V REGULATOR
16	MAIN BATT PACK 3 TEMP
17	MAIN BATT PACK 4 TEMP
18	SPIN GAS BOTTLE TEMP
19	TOP SKIN TEMP
20	UCL LA MONITOR 1
21	RIM SKIN TEMP
22	TR 2 TEMP
23	HUB TEMP
24	SORE 5 V REGULATOR
25	SORE 15 V REGULATOR
26	LASL HV1 MONITOR
27	LASL HV2 MONITOR
28	LASL TEMP
29	SPACECRAFT LOAD CURRENT
30	LCH BUS VOLTAGE
31	UCL SQ DISARM MONITOR
32	UCL LA MONITOR 2 - CALIB OFF
33	SPIN AUTO 15 V REGULATOR
34	SPIN MANUAL 15 V REGULATOR
35	RUTGERS VOLTAGE MONITOR
36	RUTGERS DATA SAMPLE
37	SPIN GAS PRESSURE
38	SPIN BOX TEMP
39	RIB TEMP
40	DECK TEMP
41	SOLAR ARRAY CURRENT
42	UCL TEMP MONITOR - CALIB ON
43	UCL HV 3 MONITOR
44	UCL HV 2 MONITOR
45	RCVR 2 AGC
46	UCL HV1 MONITOR
47	PITCH COIL STATUS
48	SYNC

OSM

WORD	MONITOR
1	UCL MODE AND HV STATUS
2	UNM BIT ERROR CODE
3	SINGLE BIT
4	RUTGERS CMMD-OPTICS STATUS
5	UCL MODE AND HV MONITOR
6	UNM BIT ERROR CODE
7	BOLOGNA TOTAL COUNT
8	BOLOGNA CMMD STATUS
9	SINGLE BIT
10	UNM BIT ERROR CODE
11	UCL MODE AND HV STATUS
12	SORE SPIN RATE
13	SORE SPIN RATE
14	UNM BIT ERROR CODE
15	UCL MODE AND HV STATUS
16	SPACECRAFT SPARE
17	NRL BURST-BACKGROUND
18	UNM BIT ERROR CODE
19	HCO SPARE
20	UCL MODE AND HV STATUS
21	NRL CMMD STATUS
22	UNM BIT ERROR CODE
23	NRL BRAGG POSITION
24	NRL BRAGG POSITION
25	UCL MODE AND HV STATUS
26	UNM BIT ERROR CODE
27	NRL CMMD STATUS
28	NRL BURST-BACKGROUND
29	HCO SPARE
30	UNM BIT ERROR CODE
31	UCL MODE AND HV STATUS
32	SORE COUNT ANGLE
33	SORE COUNT ANGLE
34	UNM BIT ERROR CODE
35	UCL MODE AND HV STATUS
36	FRAME COUNT
37	SINGLE BIT
38	UNM BIT ERROR CODE
39	NRL BURST-BACKGROUND
40	UCL MODE AND HV STATUS
41	SINGLE BIT
42	UNM BIT ERROR CODE
43	HCO SPARE
44	NRL BURST-BACKGROUND
45	UCL MODE AND HV STATUS
46	UNM BIT ERROR CODE
47	SPACECRAFT SPARE
48	SYNC

SAIL ANALOG SUBCOMMUTATOR (SASC)

WORD	MONITOR
1	EL ANGLE, POINT MODE
2	EL ANGLE 1, RSTR-OFST MODE
3	AZ ANGLE, POINT MODE
4	PITCH ANGLE
5	HCO HV MONITOR
6	HCO GRATING POSITION MONITOR
7	AZ ANGLE 1, RSTR-OFST MODE
8	HCO +4.3 V MONITOR / PULSE AMP OUT
9	HCO +4.3 V MONITOR / PULSE AMP OUT
10	RSTR-OFST MODE POSITION EYE TEMP
11	EL SERVO ERROR
12	EL ANGLE 2, RSTR-OFST MODE
13	AZ ANGLE 2, RSTR-OFST MODE
14	EL MTR CURRENT
15	OFST MODE STATUS
16	NRL PM HV MONITOR
17	AZ ANGLE 3, RSTR-OFST MODE
18	NRL PROP HV MONITOR
19	NRL BRAGG HV MONITOR
20	AZ MTR CURRENT
21	NRL BURST HV MONITOR
22	EL ANGLE 3, RSTR-OFST MODE
23	AZ ANGLE 4, RSTR-OFST MODE
24	SOLAR INTENSITY
25	AZ SERVO ERROR
26	NRL RSTR HV MONITOR
27	NRL +9 V MONITOR
28	NRL +12 V MONITOR
29	NRL CRYSTAL TEMP
30	NRL FRONT TEMP
31	NRL +4 V MONITOR
32	WG GEN TEMP
33	SOLAR ARRAY (CENTER) TEMP
34	SOLAR ARRAY (LWR RT) TEMP
35	AZ ANGLE 5, RSTR-OFST MODE
36	EL ANGLE 4, RSTR-OFST MODE
37	AZ ANGLE 6, RSTR-OFST MODE
38	EL OFST GEN STATE
39	AZ OFST GEN STATE
40	SAIL ANALOG SUB-SUBCOM (ASSC)
41	SASC TEMP
42	AZ ANGLE 7, RSTR-OFST MODE
43	DAY PWR / HCO HV LCH AND DAY
44	NRL DAY / HCO LV LCH AND DAY
45	PITCH MODE / APL STATUS
46	AZ ANGLE 8, RSTR-OFST MODE
47	PITCH GAS PRESSURE
48	SYNC

WORD 3 - SINGLE BIT ALLOCATION

BIT NO.	MONITOR
1	DME 1 ON (MSB)
2	DME 2 ON
3	XMTR 1 ON
4	XMTR 2 ON
5	XMTR 2 DISAB
6	TR 1 ON
7	TR 2 ON
8	SEPARATION SWITCH ENAB

WORD 9 - SINGLE BIT ALLOCATION

BIT NO.	MONITOR
1	UVS BYP CMMD DISAB (MSB)
2	UVS NOT BYP
3	ORBIT RELAY NOT BYP
4	D-N RELAY NOT BYP
5	SPIN AUTO
6	ASL DISAB
7	TMG DISAB
8	DL ON

WORD 37 - SINGLE BIT ALLOCATION

BIT NO.	MONITOR
1	ORBIT PWR ON (MSB)
2	WHEEL DAY PWR ON
3	SAIL DAY PWR ON
4	UNM ORBIT PWR ON
5	BOLOGNA ORBIT PWR ON
6	UCL DAY PWR ON
7	LASL DAY PWR ON
8	RUTGERS DAY PWR ON

WORD 41 - SINGLE BIT ALLOCATION

BIT NO.	MONITOR
1	LSBU DISABLE
2	LST RUNNING
3	ARM 1 ERECT
4	ARM 4 ERECT
5	ARM 7 ERECT
6	ARM 1 STOWED
7	ARM 4 STOWED
8	ARM 7 STOWED

ANALOG SUB - SUBCOMMUTATOR

WORD	MONITOR
1	PWM 15 V REGULATOR
2	SERVO 15 V REGULATOR
3	TEMP PROBE 15 V / VERNIER OFST BITS
4	ON TARGET / FINE POINT MODE
5	PITCH MANUAL 15 V REGULATOR
6	HCO ELECTRONICS TEMP
7	HCO DELTA TEMP
8	PITCH AUTO 15 V REGULATOR
9	SERVO AMPLIFIER TEMP
10	AZ CASTING TEMP
11	PWR SWITCH TEMP
12	OFST DAY 15 V REGULATOR
13	WG GEN STATUS
14	RT STRUCTURE TEMP / NUT DAMPER REL
15	OFST LCH 15 V REGULATOR
16	ROLL COIL 1 STATUS
17	HCO -5.5 V / READ-IN GATE
18	HCO ELECTRONICS TEMP
19	HCO DELTA TEMP
20	HCO OP CKT RST / EXCT MONITOR
21	HCO OP CKT ENAB / DISAB MONITOR
22	ROLL COIL 2 STATUS
23	SYNC
24	SYNC

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NOTE:
See Table 10-1 for abbreviations.

Fig. 9-2   OSO-6 Telemetry Format

Figure 9-3 is a functional block diagram of the data-handling subsystem, and 9-4 is a logic diagram of the DME.

The basic timing-signal generator is a crystal-controlled oscillator operating at 102.4 kHz. This basic frequency is divided down through a binary divider consisting of five flip-flops. The output of the 6.4 kHz flip-flop is fed to another binary counter in the bi-phase modulator. The 800 Hz (master clock) output of this divider is used to provide all of the necessary timing signals in the system. The master clock is also brought out to be used as shift pulses in the experiment and spacecraft shift registers.

The main-frame word-gate signals are generated by the word-gate generator consisting of a matrix of AND gates. This circuit is essentially a sequential pulse generator that produces 32 sequential pulses. The timing is such that each pulse is one word-length in duration (10 msec). These 32 pulses are referred to as "word gates". Word gates 31 and 32 are connected internally to the main frame sync-code-generation circuits. During the time of word 31 and 32, a pseudo-random code, "00011110/10110010" (which corresponds to a count of 30 and 178 respectively), is multiplexed into the output for main-frame identification and ground equipment synchronization.

Twenty-eight of the remaining word gates are routed outside the DME to the PCM junction box (J-Box) patch panel where they are interconnected to make up the desired multiplex format for the wheel and pointed experiments and the DSM. Two of the word gates are connected back into the DME at the inputs designated "word A" and "word B" for generation of the analog subcommutator timing signals. These two inputs activate trigger generation circuitry that develops two gates, 156 μsec wide. These gates are routed outside the DME as ASC-1 trigger and ASC-2 trigger.

C 6
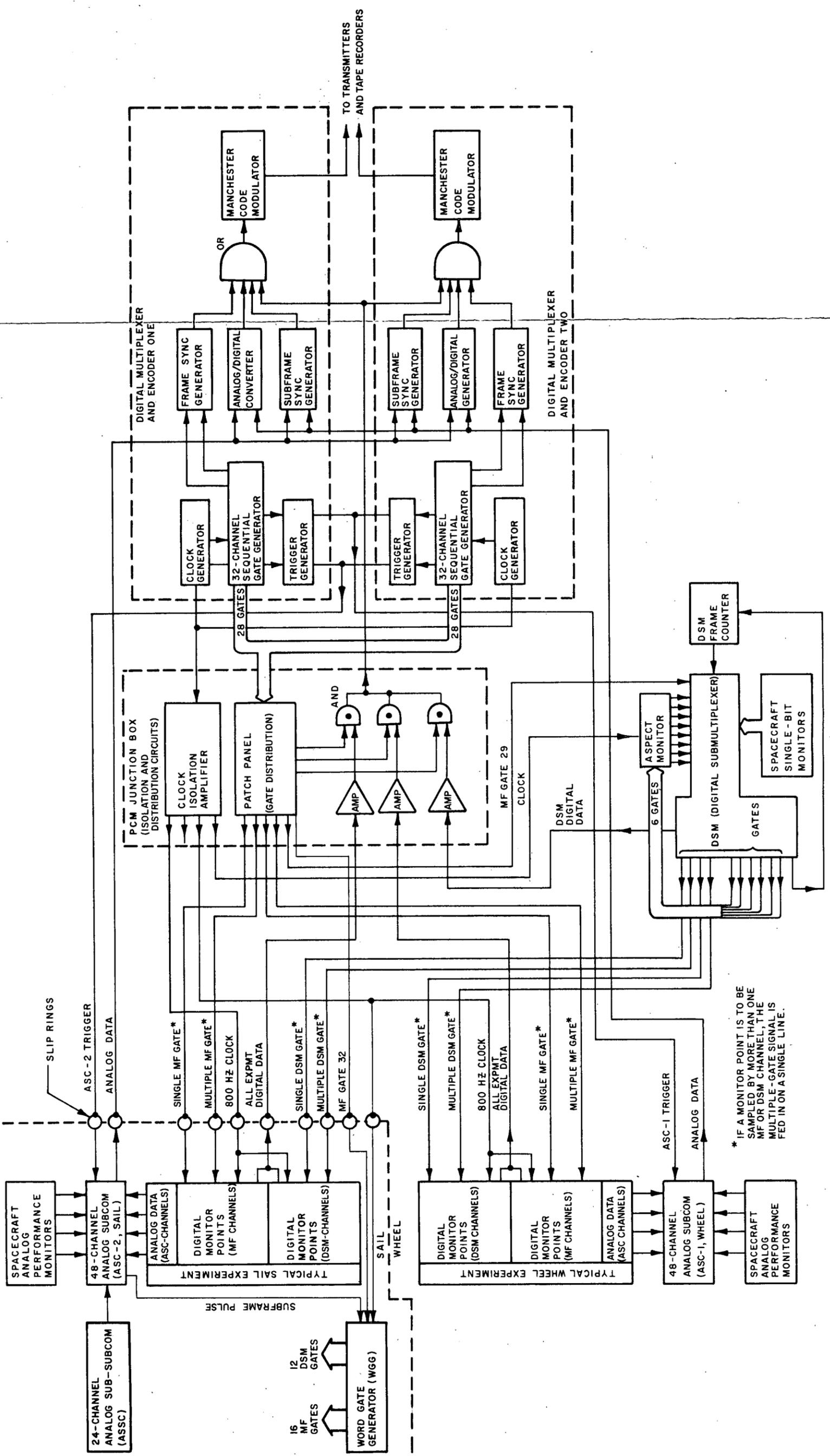
Fig. 9-3 Data-Handling System Functional Diagram

9-7
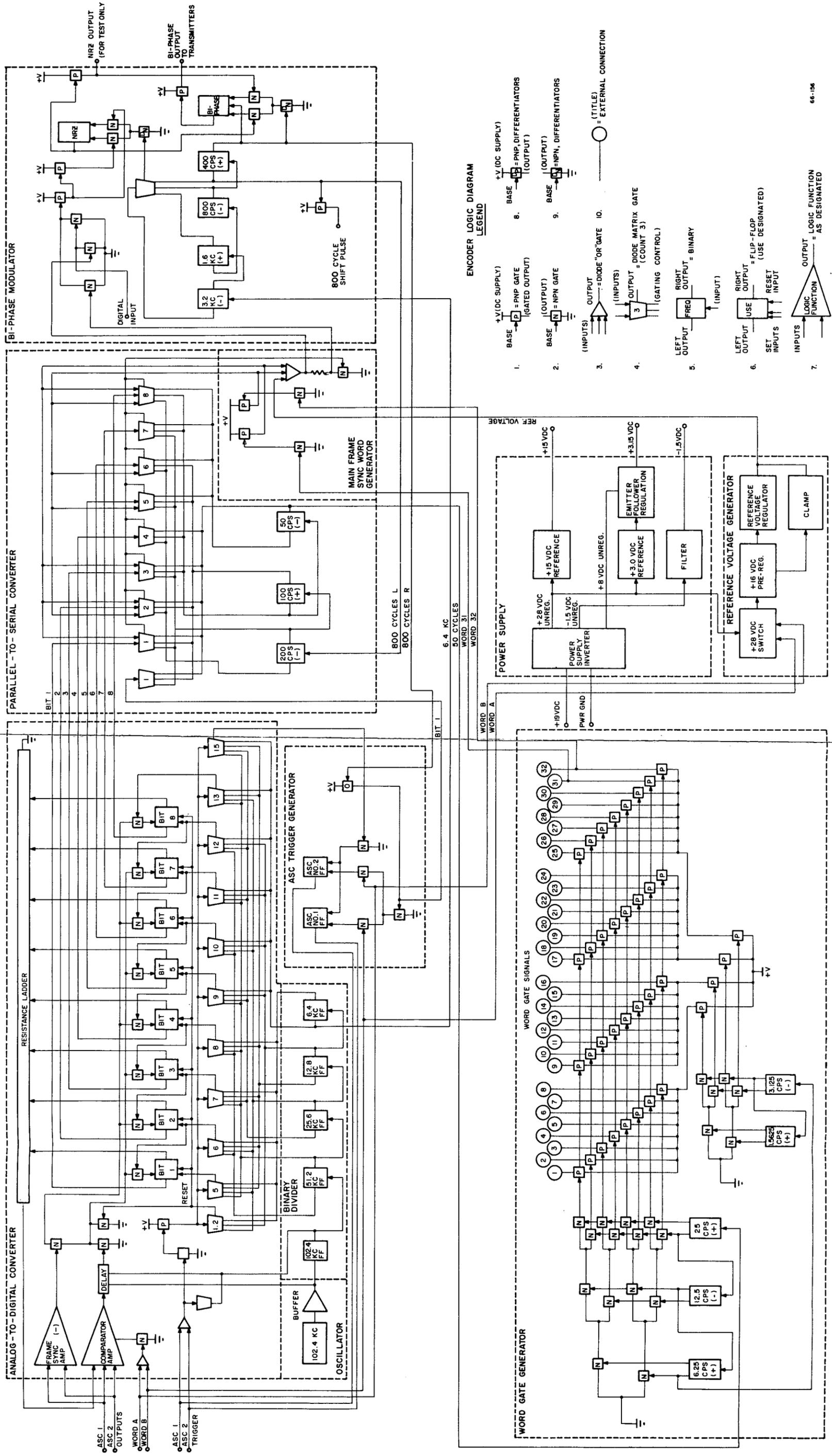
Fig. 9-4 Digital Multiplexer and Encoder Logic Diagram

9-8

The "word A" and "word B" inputs also operate a power switch that applies power to the analog-to-digital converter (ADC) portion of the DME. Although the circuitry required to do the analog-to-digital conversion takes appreciable power, a significant power savings is accomplished by switching the ADC on only when it is required to operate. (See Fig. 9-5 for the DME timing diagram.)

The PCM J-Box also provides gate pulses to the DSM to generate the 48 DSM words, to the individual experiment instruments to shift out their data at the proper time, and to the word gate generator (a new feature on OSO-6) which in turn provides gates to the HCO pointed experiment.

Analog-to-Digital Encoding. The analog-to-digital encoding is accomplished by a successive-approximation method. The analog input from the ASC's is fed to a comparator in the DME's where it is compared against a series of reference voltages and the result of each comparison is stored in a flip-flop. For example, during the first comparison interval, a voltage is applied to the resistance ladder network to develop a one-half-scale voltage reference at the comparator. When the input voltage is more than half scale, a "one" will be set into the first flip-flop and if it is less than half scale, a "zero" is set into the first flip-flop. During the second comparison interval a voltage is applied to the resistance ladder network to develop a one-quarter-scale reference voltage. If a "one" is set into the first storage flip-flop during the first interval, the half-scale reference will remain and is added to the one-quarter-scale reference. If a "zero" is set, then only the one-quarter-scale reference will be retained.
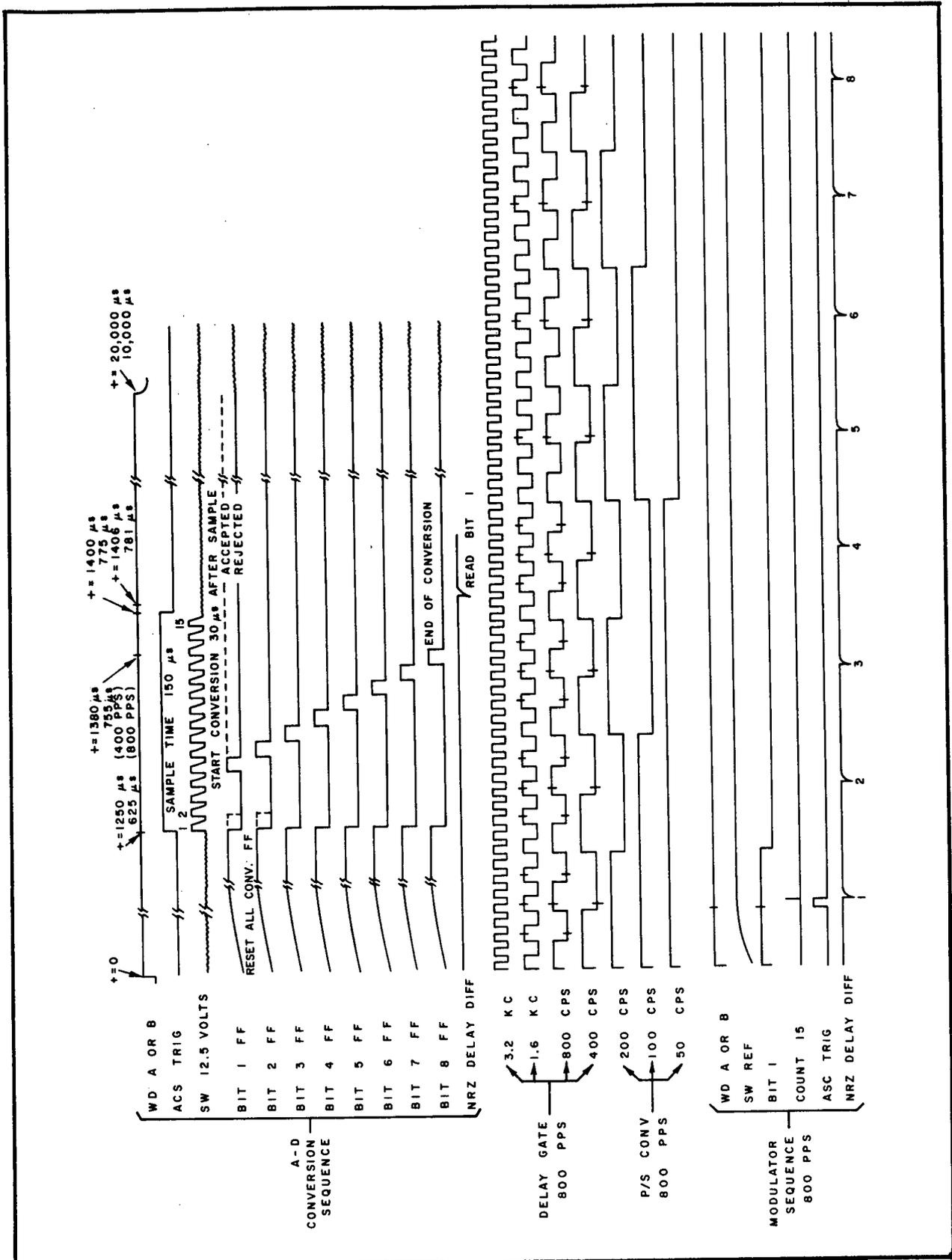
Fig. 9-5 DME Timing Chart

The result of the second comparison is then set into the second storage flip-flop. This process continues through eight successive comparisons, whereby the reference voltage is changed by one-half of its previous incremental change. During the last comparison the reference voltage will have a difference from the input voltage within one part in 256 of the full-scale voltage and a binary code equivalent of the analog input will thus be stored in the eight AND gates. The complete encoding process takes place during each sample interval of the analog subcommutator which is approximately 156 μsec.

The output of each of the flip-flop storage units is applied to the eight AND gates. These AND gates are then sequentially enabled at the 800-Hz bit rate, and the output is a serial pulse train of eight bits which represents the coded analog signal for that particular ASC channel. These serial data are then applied through OR logic where it is time-multiplexed with the digital data from the experiments and DSM. The complete data train is then applied to the Manchester encoding circuitry which develops the biphase (Manchester) coded output signal.

Input and Output Data Formats. The OSO data handling subsystem will accept four types of data formats:

    (1)   0 to 5 V analog

    (2)   RZ (shifted)

    (3)   NRZC (not shifted), and

    (4)   NRZC (shifted).

The RZ (return-to-zero) format is where the presence of a bit signifies a binary "one" and the absence of a bit signifies a

zero. The NRZC (non-return-to-zero, change) format is where
the data train changes state for every change in binary value.
The timing relationships of these formats are illustrated below:
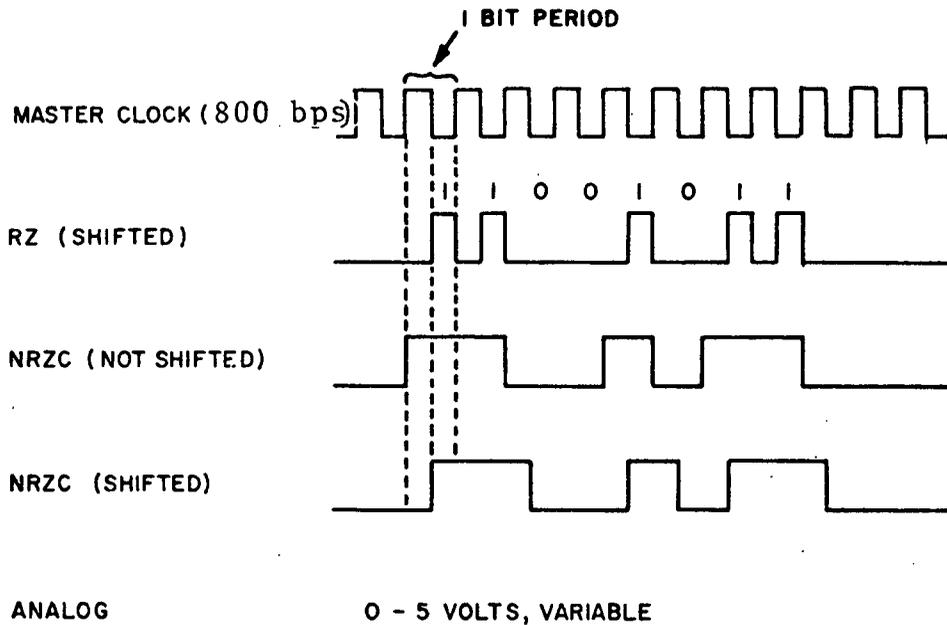
The DME output is biphase (Manchester) format, illustrated below:
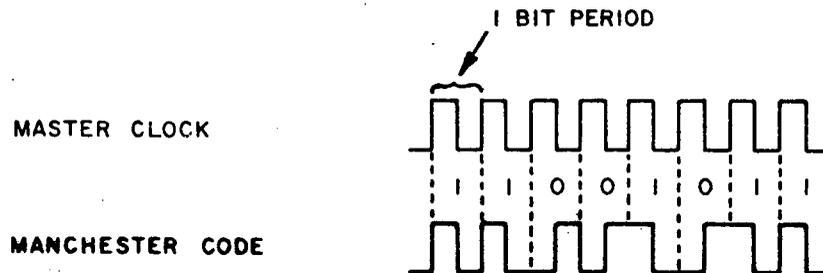
ASC Sync-Code Generation. When the input from the sail or wheel
ASC is a positive-going pulse (ASC channel 48 only), rather than
the normal negative-going pulse, it is detected by the frame sync
amplifier in the DME. The normal action of the ADC is then
inhibited and a "one" is set into every other storage flip-flop.
This alternate "one-zero" code is the synchronization code for
the subcommutator, and is read out as encoded data with a count
value of 085. The value 085 is unique in that all other ASC
channels are inhibited from having this count.

Data Handling Subsystem Synchronization. Main frame synchronization
is determined by ensuring that the proper number of main frame
words exist between the main frame sync words. As previously
stated, these words consist of words 31 and 32 into which a 16
bit pseudo-random code of 30/178 is inserted.

The main-frame sync code is generated by applying pulses to the
appropriate AND gates that are used for sampling the encoder
storage units. This process of sequentially enabling these AND
gates during main-frame words 31 and 32 generates the synchronization
(or ID) code.

The wheel and sail ASCs are mutually synchronized within the 32-
word main frame by using the unique sync code contained in the
48th word of each ASC. This synchronization is accomplished
automatically by circuitry in the DME in the following manner:

> (1) The sync circuits in the DME expect to see both
>   of the ASC sync words at their assigned times
>   within the ASC subframe. The sail ASC sync word
>   is used as the synchronization reference, or
>   "master", and the wheel ASC is "slaved" to this
>   reference.

(2) The presence of the sail sync word and the absence of the wheel sync word in the same main frame results in a synchronization error decision by the sync circuits in the DME. The sync circuits then generate trigger pulses at 1,024 times the normal ASC trigger rate. These trigger pulses are used to advance the wheel ASC during main-frame words 31 and 32 until the wheel ASC sync word appears. The wheel ASC is then mutually synchronized with the sail ASC again. Since the ASC sync words are assigned to channel 48, the next time the ASC's are read out, channel one will be present because the ASC's will have been recycled.

The DSM subframe channels are identified by using the sync word that appears in the first channel. This sync word (or ID word) is a unique code (085) just as that used for the ASC's. Unlike the ASC, the DSM sync word is generated internally, whereas the ASC sync word is generated in the DME.

External Digital Multiplexing. External digital signals (experiment prime data) are time-multiplexed in the PCM J-Box and are fed to the DME's in serial form. These data are then multiplexed with the converted analog data by OR logic circuits, and signal conditioning is accomplished simultaneously with conversion to Manchester code form.

System Characteristics. With the exception of some of the logic in the analog-to-digital converter, all circuitry within the DME is quad-redundant. That is, each logic function is both series and parallel redundant and multiple component failures must occur before the logic function fails. Assembly failures caused by component failures are greatly reduced by this design, but at the sacrifice of some simplicity in fabrication.

9.2.3   Analog Subcommutator

The purpose of the wheel and sail analog subcommutators (ASC-1 and ASC-2) is to time-multiplex 47 wheel and sail analog input signals varying in amplitude from zero to +5V and to provide an ID synchronization signal to facilitate data reduction. A block diagram of the ASC is shown in Fig. 9-6 and a logic diagram in Fig. 9-7.

The input timing signal consists of the ASC trigger pulse (word A or word B) from the DME. This trigger is applied to a divide-by-48 counter that recycles at the end of 48 counts. The counter consists of binary stages two through seven on the logic diagram. A seventh binary (number one on the logic diagram) is triggered "on" by the leading edge of the trigger pulse and "off" by the trailing edge of the trigger pulse. (See Fig. 9-8 for ASC timing chart.)

The outputs of each stage of the binary counter are applied to decoding gates that are interconnected to form two matrices. One matrix has an eight-line output and the other has a six-line output. Each line of the eight-line matrix is connected to six analog gates, which are referred to as first tier gates, and each line of the six-line matrix is connected to a single analog gate which is referred to as a second tier gate.

The type of analog gate used is characterized by high input impedance in the "off" condition and very low impedance in the "on" condition. The input impedance in the "on" condition is essentially the input impedance (100 k ohm) of the buffer amplifier which provides impedance matching. As shown in the logic diagram, there are six groups of eight analog gates where the outputs of each group are connected. The six common lines are then connected
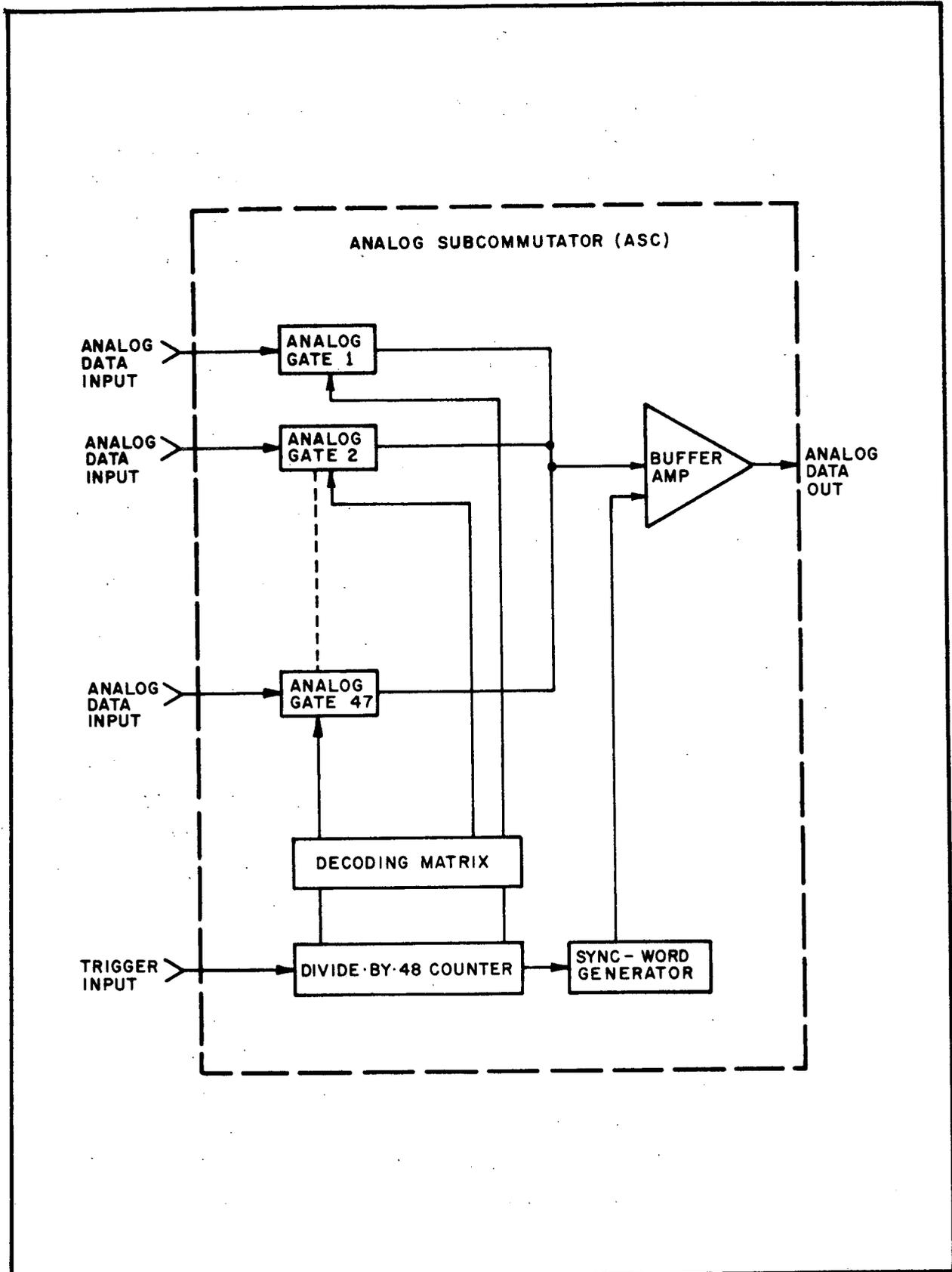
Fig. 9-6 Analog Subcommutator (ASC) Functional Block Diagram

Fig. 9-7 Analog Subcommutator Logic Diagram
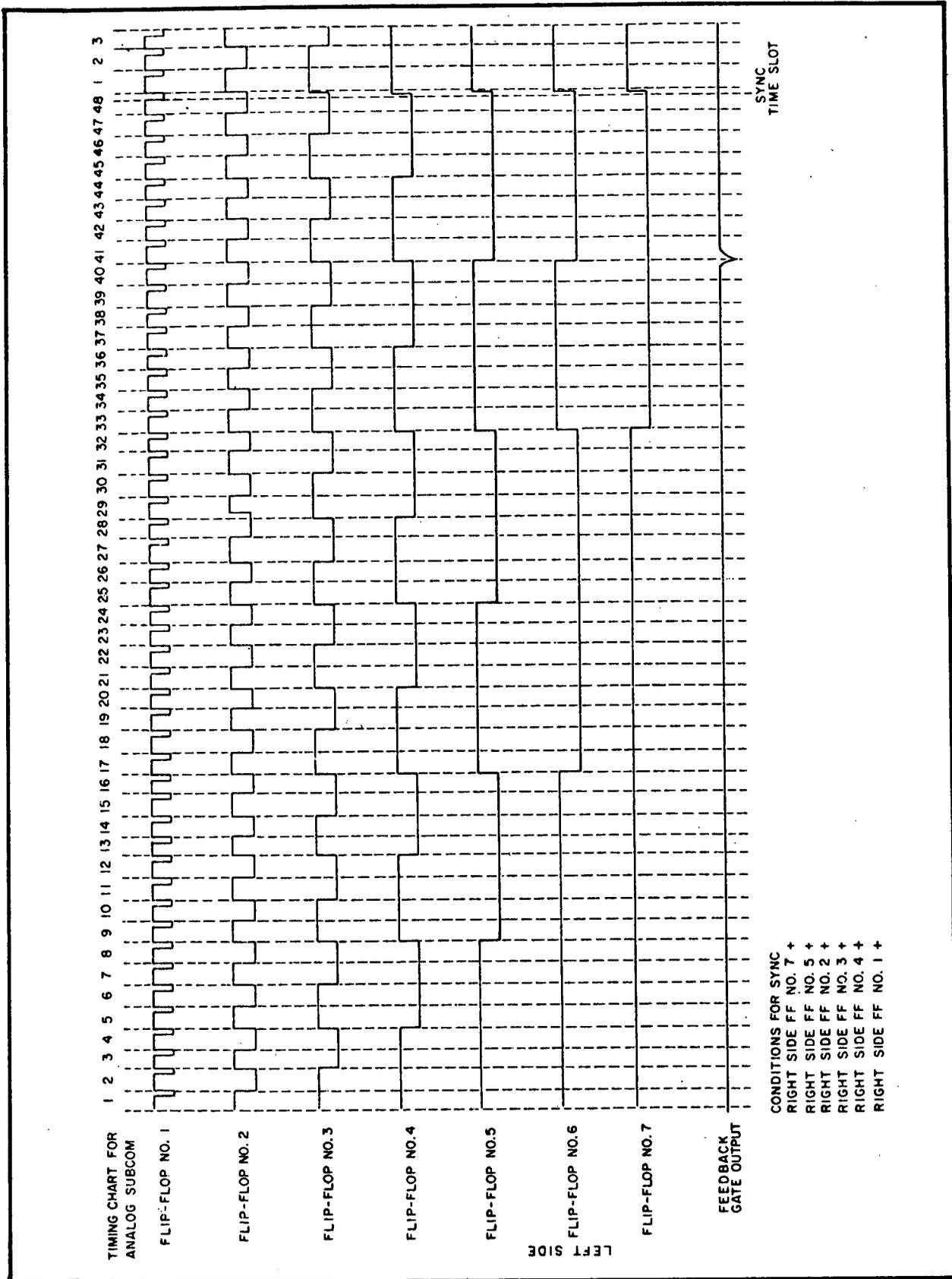
Fig. 9-8 ASC Timing Chart

through the second tier analog gates. The second tier gates
sequence at one-eighth the rate of the first tier gates and,
therefore, only one input is connected to the output at any one
time.

The common connection of the second tier gates is connected to a
buffer amplifier. The purpose of the buffer amplifier is to
provide a constant input impedance for all channels and provide a
low output impedance for the signal going to the encoder in the
DME. The buffer also inverts the input signal and inserts a dc
reference of +8 v.

Since the analog gates are turned on for only 156 μsec (the width
of the trigger pulse), the output signal is a negative-going pulse
that is amplitude modulated over a range of five volts, and has
a zero reference of +8 v.

When the binary counter reaches the forty-eighth count, it
activates logic circuits which apply a negative voltage to the
input of the buffer. Since the buffer inverts, this results in
a positive-going pulse being generated for the forty-eighth
channel as a sync pulse.

As in the DME, the ASC uses quad-redundant circuits except for
the analog gates. The packaging technique is much the same as
the DME except that the printed circuit cards are smaller.

9.2.4   Analog Sub-Subcommutator

The analog sub-subcommutator (ASSC) is a commutator capable of
time multiplexing 24 analog signals onto a common output line
and its purpose is to increase telemetry channel capacity of
the sail ASC. The output of the ASSC is fed into channel
40 of the sail ASC. The ASSC is advanced one channel per ASC
subframe.

The ASSC consists of a buffer amplifier, five flip-flops, a diode matrix, and 24 analog switches (Fig. 9-9). The subframe rate pulse from the sail ASC is fed into the buffer amplifier, which provides isolation between the internal flip-flops and the external data handling system. The output of the buffer amplifier drives the first flip-flop in the countdown circuit. The five flip-flops are connected in a conventional ripple-through counter arrangement with feedback from the fifth to fourth stage that causes the counter to recycle at the count of 24. The outputs from the flip-flops sequentially turn-on the analog switches via the diode matrix. The analog switch that is turned-on couples the input signal to the common output line with a minimum of signal loss. The input of each switch has an overvoltage protection circuit to prevent an overvoltage condition on any input channel from interferring with the operation of the remaining channels. The ASSC is synchronized by applying 0 to 5 volt signals to the inputs of channels 23 and 24.

The analog switches use field effect transistors (FET's) and have the advantages of: (1) simple control circuits, (2) no offset voltage, and (3) excellent long term stability since small changes in the FET parameters are not critical to the switch operation.

9.2.5   PCM Junction Box

The purpose of the PCM J-box is to: (1) provide interconnection and distribution of all data-handling timing signals (gates), (2) provide buffering and conditioning of these signals for proper interface with other data handling equipment and (3) provide buffering and gating of main frame data. A functional diagram of the PCM J-box is shown in Fig. 9-10.
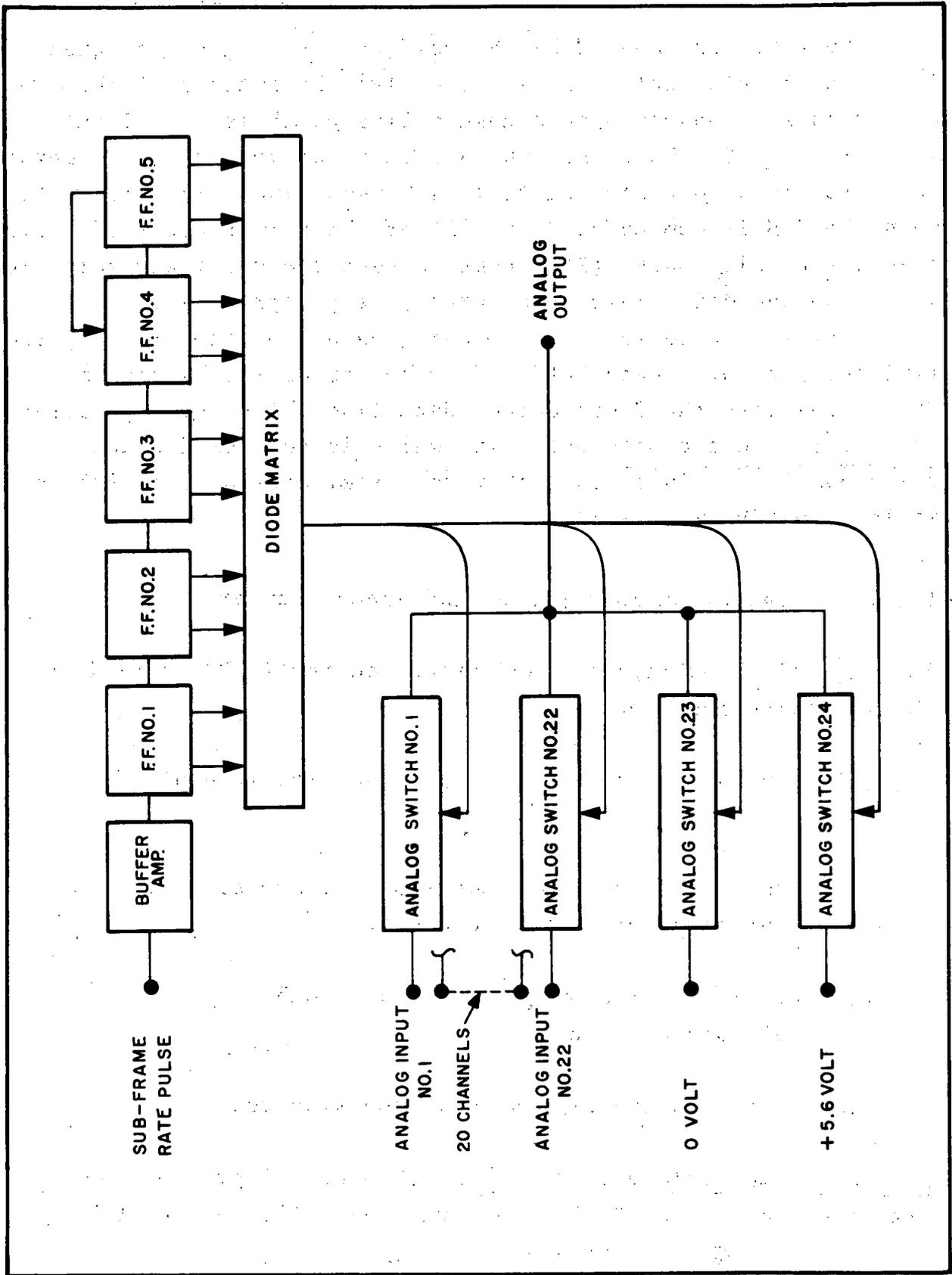
Fig. 9-9 Analog Sub-Subcommutator Block Diagram
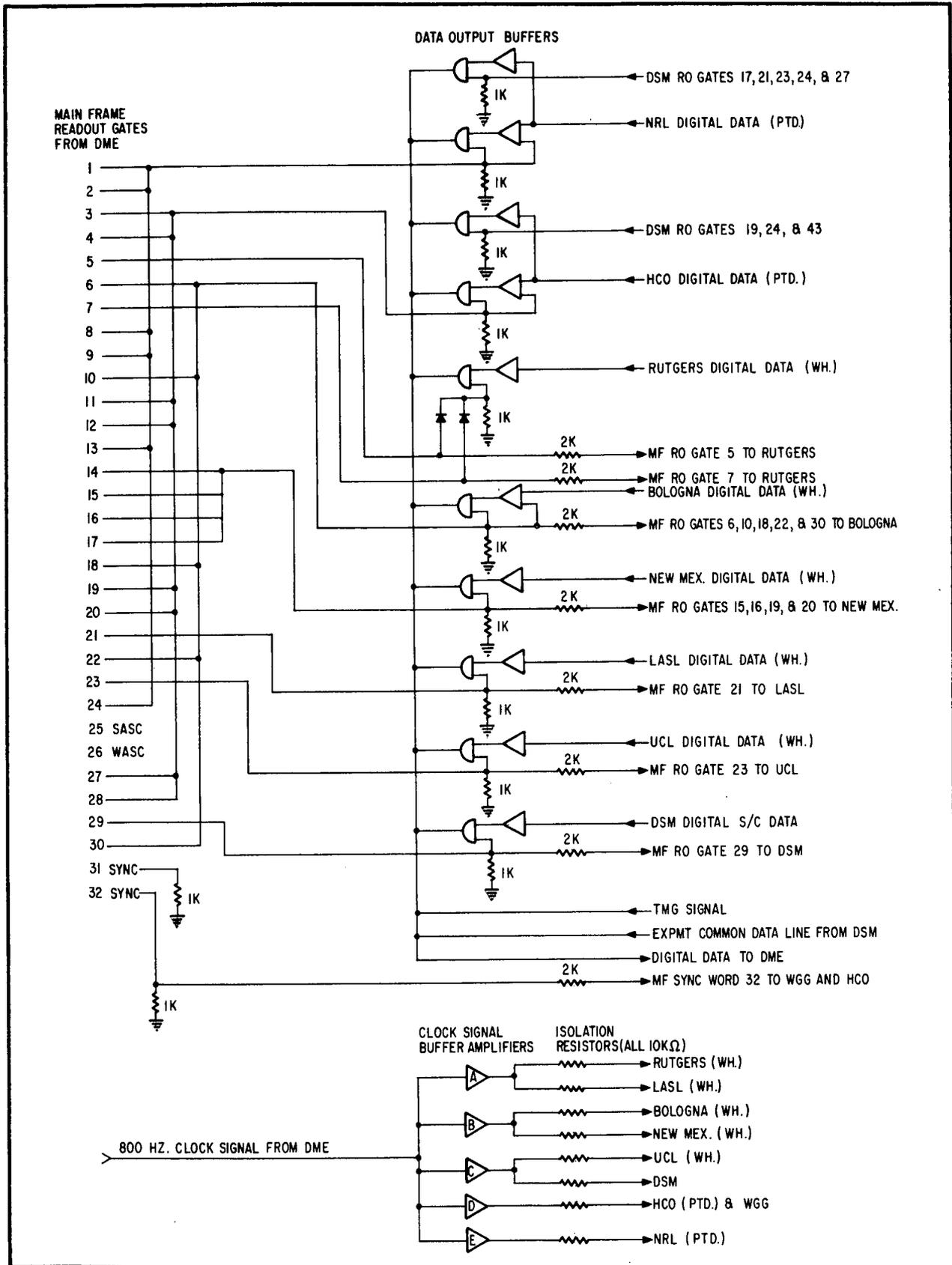
Fig. 9-10.   PCM J-Box Functional Diagram

All of the word gate signals (generated by the DME) are routed
into the J-box where they are interconnected according to the
channel assignments. To prevent excessive loading of the DME
output circuits, each word gate is applied across a 1 k ohm load
resistor and isolation is provided by a 2 k ohm series resistor.

Master Clock Buffer Amplifiers. The 800 Hz master clock is routed
from the DME to the J-box where it is distributed to all
experiments and other spacecraft assemblies requiring this signal.
To prevent loading by one user affecting the signal characteristics,
and to improve overall reliability, the clock signal is routed
through five parallel buffer amplifiers. Each of these buffer
amplifiers is capable of supplying signals to four separate users
in such a way that a short circuit can occur on the external
signal line without affecting the signal characteristics for the
other users. This is accomplished by providing a low output
impedance at the amplifier and inserting series isolation
resistors in each line.

Data Output Buffers. It would be possible to directly inter-
connect the experiment digital data outputs into a common line
to the DME if all the experiments had high impedance outputs
and if there was no chance of the experiment registers being
gated at an improper time. However, due to the possibility
of a circuit failure in one experiment affecting another, all
digital data signals are buffered and isolated in the PCM J-box
before they are multiplexed.

Buffering between the data sources and the data handling equip-
ment is accomplished by connecting each digital data line to a
simple buffer amplifier which also provides a compatible inter-
face (e.g., impedance matching, gain etc.) for each data source.
The output of each of these amplifiers is connected to a diode

AND gate which is enabled by the word gate(s) signal assigned
to the particular data source. Finally, the outputs of the
AND gates are connected to a common line which is the signal
line to the DME. Note that the 5 volt input from the time-marker
generator is also connected to this common line. This means that
when the TMG is commanded "on", all of the main frame words will
read out as all "ones" (255 count) except the wheel and sail ASC
words and the main frame sync words (see paragraph 9.2.8). Also,
note that the spacecraft housekeeping data from the DSM is gated
(multiplexed) by a data buffer (the bottom buffering amp in
Fig. 9-10) onto the J-box output data line while the experiment
data is fed on a common data line which is connected directly to
the J-box line. This means that the experiment data is multiplexed
inside the DSM by gate 29.

9.2.6   Digital Submultiplexer and Frame Counter

The digital submultiplexer and frame counter (DSM/FC) performs
the following functions:

(1)   Generates sequential gate pulses for readout of
digital data

(2)   Stores and updates a count of the number of digital
subframes for a time reference.

(3)   Provides a readout of the frame count into the main
multiplexer, and

(4)   Generates the digital subframe sync code.

Digital Submultiplexer. A functional block diagram of the
DSM is shown in Fig. 9-11. The synchronization of the DSM
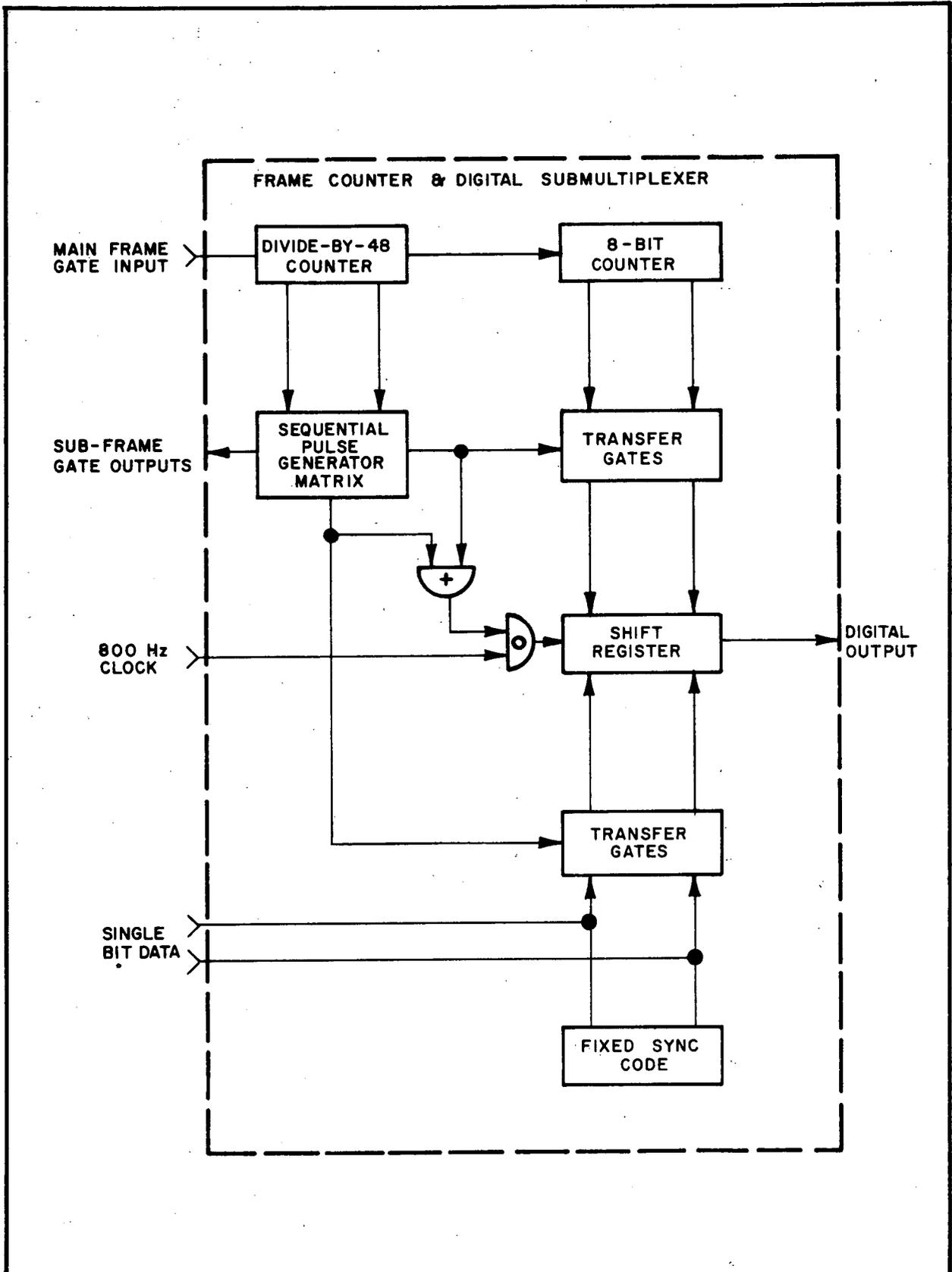
Fig. 9-11   Frame Counter and Digital Submultiplexer
Functional Block Diagram

is provided by a gate signal from the DME which occurs once per main frame. This signal drives a divide-by-48 counter which establishes the subframe length. A diode matrix then generates 48 sequential gate pulses that are displaced in time from each other by one main frame. These gate signals can be used in the same manner as gate signals from the DME to readout data from external shift registers. Some of these gates are connected internally for readout of the internal shift register.

The 800-Hz master clock is used to transfer the data into the DME in serial form. Another set of transfer gates is used to transfer a fixed sync code word (085) into the shift register at the beginning of the subframe. The sync code is then shifted into the main multiplex by the 800 Hz clock. There are several other sets of transfer gates built into the DSM for transferring parallel digital data from external sources such as aspect data from the Spin Orientation and Rate Electronics (SORE).

Frame Counter. A signal from the divide-by-48 counter goes into an eight-stage counter which accumulates a count of the subframes. Transfer gates connected between the eight-stage counter and a shift register are actuated by a selected gate signal to transfer the accumulated frame count into the shift register for readout into the main multiplex. This frame count provides a means of determining time correlation of the playback data.

9.2.7   Tape Recorder/Reproducer

Data that are collected when the satellite is out of range of ground receiving stations are stored with two redundant magnetic tape recorders. The tape recorders then play back the stored data, upon command, when the satellite is within range of ground stations. The placement of ground stations used for OSO establishes the

nominal requirement that, at a maximum, data must be stored for a complete orbit, or about 96 minutes. The nominal time required for playback over a ground station is about 5-1/2 minutes. This allows ample time on either side of the playback to avoid the marginal conditions when the satellite is near either horizon.

The OSO magnetic tape recorder uses a continuous loop of magnetic tape to record the biphase signal from the DME. Data are recorded at a rate of 800 bps and are played back at a rate of 14,400 bps. The tape capacity of the recorder is about 300 feet which results in a bit packing density of about 1,333 bits per inch to store 100 minutes of data. The OSO-6 tape recorders, like the OSO-5 recorders, use single track recording techniques and are belt driven. The playback timer and the drive electronics are driven by an internally contained clock and the tape is erased during playback. The recording tape used in the OSO-6 recorders is different from the lot used in the OSO-5 recorders.

Operation. A functional block diagram of the tape recorder/reproducer is shown in Fig. 9-12. The Manchester-coded data from the DME is amplified and differentiated to produce positive and negative spikes. These spikes are fed to a flip-flop that serves as the record-head driver. The negative pulses are fed to one winding of the record head while the positive pulses are fed to the other winding. The pulses drive the record windings in push-pull so that the tape is driven into saturation in both directions.

In the playback mode, the two windings of the playback head develop a pulse that is rounded off and is not a good reproduction of the recorded signal. The playback signal is therefore amplified and clipped by two amplifiers and a level detector. The playback signal is thus conditioned to look more like the recorded signal. The two squared pulses are then differentiated and are applied to
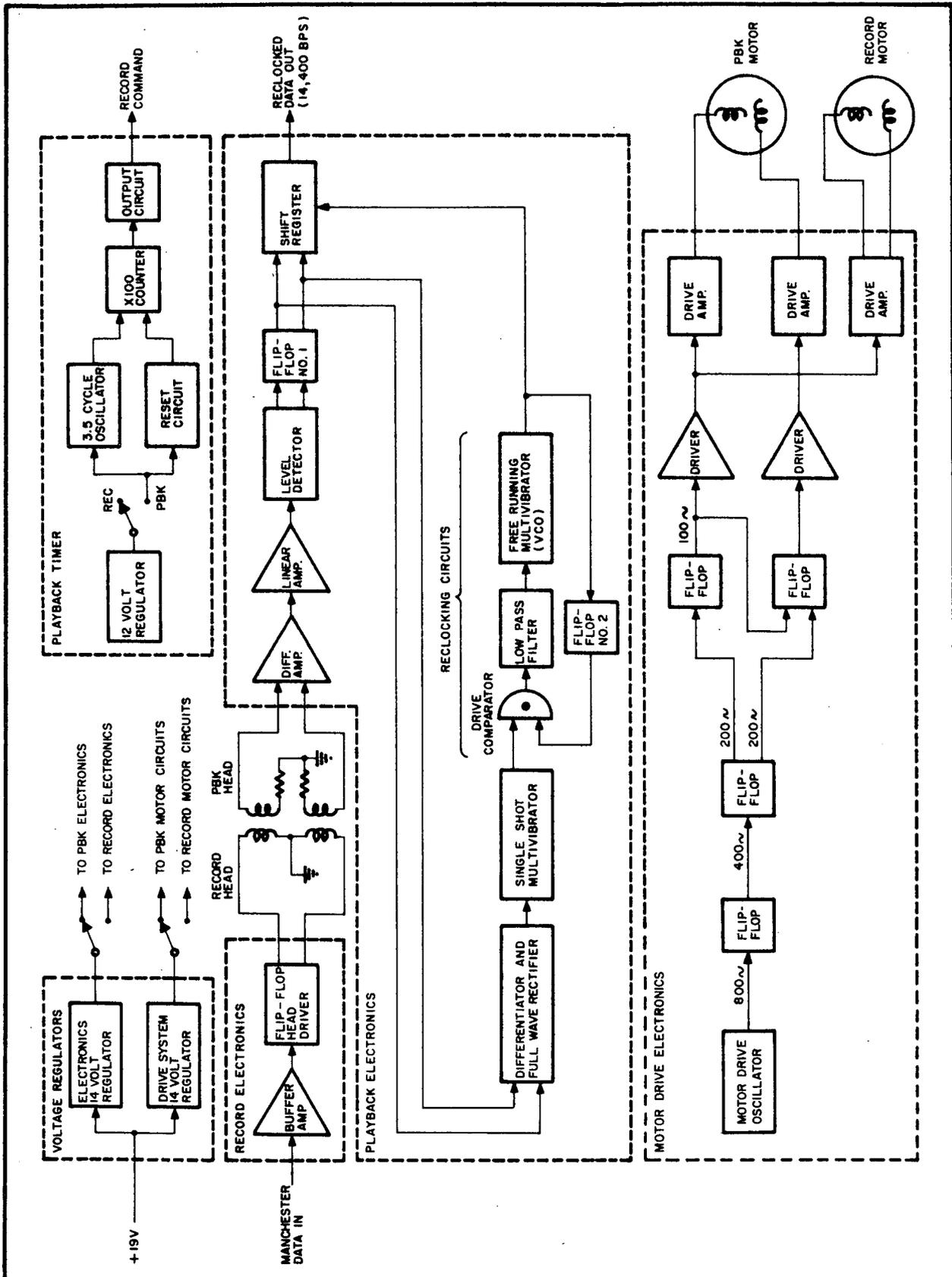
Fig. 9-12 Tape Recorder Block Diagram

the input of the flip-flop No. 1. The output of flip-flop No. 1 is the same as the input signal to the recorder, but is increased in frequency by the 18-to-1 ratio (reproduce/record speed), is modulated by wow and flutter, and has minor distortion caused by the detection electronics. The signal distortion manifests itself in phase and frequency modulation of the signal and is referred to as jitter.

To correct for some of the jitter components, the playback signal is reconditioned by the phase-lock loop circuits. In essence these circuits compare the phase of the flip-flop output with a reference signal obtained from a voltage-controlled oscillator (VCO). The phase difference is then detected and this error signal is filtered and applied to the VCO. The error signal then changes the reference frequency until no error is detected. The filter does not allow high-frequency changes to reach the VCO. Thus, the reference signal is controlled by the average frequency out of flip-flop No. 2. The ac signal from the VCO is used to clock the flip-flop No. 1 output into a one-stage shift register. The output of the shift register has then the same frequency as the flip-flop No. 1 output, with high-frequency jitter components reduced.

Low-frequency changes (wow) in the playback signal are affected substantially by the stability of the motors that drive the tape. Synchronous motors are used and are controlled by the 800-Hz telemetry clock to assure stable operating speed. The 800-Hz clock is divided down to 100 Hz in binaries, and applied to the motor-driver amplifiers. Both motors run at the same speed and drive power is applied to the capstan by a system of pulleys and Mylar belts. Only one motor has power applied to it at any one time. The playback drive mechanism produces a tape speed that is 18 times the record speed.

Tape Recorder Timer. The playback time of the tape recorders is governed by a playback timer contained in each tape recorder. This timer generates a control pulse at the end of a preset time period (337.5 sec) and switches the tape recorders back to the record mode.

When a playback command is received, a latching relay is actuated, applying power to the oscillator. The oscillator consists of a unijunction transistor and an RC timing network which establishes the oscillator period of approximately 3.375 seconds per cycle. The oscillator output is connected to a magnetic counter with a capacity of 100 counts. When power is first applied to the oscillator, the counter receives a reset pulse to assure that it starts at a zero state.

After the counter receives 100 counts, it produces an output pulse which triggers a capacitor discharge in the output circuit. The capacitor discharge is the timer output which switches the recorder back to the record mode of operation. At the same time that the tape recorder is switched, the timer relay disconnects power from the oscillator.

The timer is set for a nominal 337.5 seconds and this varies little over the environmental extremes, thereby assuring that the full tape will be reproduced under all temperature conditions.

Power Switching Change. The OSO-6 tape recorders differ from the OSO-5 recorder in that the OSO-6 recorders have individual on/off commands while the OSO-5 recorders had common power on and individual power-off commands.

Recorder/Transmitter Interface Change. When the spacecraft digital data bit rate was increased from 400 to 800 beginning

with OSO-5, problems were encountered in obtaining the proper
modulation characteristics from the transmitter during play-
back. Analysis of the problem showed that the modulation
characteristics are dependent on the amplitude, frequency,
rise and fall times and symmetry of the input signal. On
OSO-5, the problem was handled by adjusting the output inter-
face network. The range of adjustment, however, was limited
by the output circuit of the tape recorder. For OSO-6, the
output circuit of the tape recorder was modified to allow a
lower impedance interface network to be used and thus improve
the rise and fall times of the transmitter input signal. The
new interface circuit provided the proper impedance levels to
make the best use of the tape recorder output and used diode
arrays to provide isolation between the outputs of the two
tape recorders.

9.2.8   Time-Marker Generator

The Time-Marker Generator (TMG) is a secondary means by which
a time-reference signal may be incorporated into the OSO tele-
metry signal. It changes the data by applying a positive dc
voltage to the digital input line of the DME, thus causing the
digital data to appear as all "ones". This feature is used
as a backup to the frame counter which provides the primary
time reference. The "TMG-On" command function simply applies
a dc signal to the DME input line (this is done in the PCM
J-box, see Fig. 9-10). This signal is subsequently removed by
the "Playback On" command.

9.2.9   Word Gate Generator

The word gate generator (WGG) is a new feature on OSO-6. Its
purpose is to generate up to 16 main frame and 12 subframe

word gates for use by the pointed instruments while using only 6 slip ring circuits and thus conserve the use of slip rings. The WGG also provides a data expansion capability for future OSO missions.

Background of Development. The main frame and subframe digital multiplexing requirements of the OSO-6 pointed experiments required the availability of 18 parallel circuits for instrument gating functions. Since, in the current OSO configuration, gating requirements were met by multiplexing equipment (i.e., the PCM J-box and DME) located in the wheel, the pointed instrument gating requirements dictated the need for 18 slip ring circuits. These requirements, when coupled with other sail slip ring circuit requirements, resulted in the need for a minimum of 40 slip ring circuits. The currently qualified slip ring design available for OSO use was for only 22 circuits. These considerations justified the need for adding the WGG to the data handling system.

Operation. The functional purpose of the WGG is to deliver to the pointed instruments, over parallel lines, a set of properly positioned word gates. These word gates sequentially "trigger" the transmittal of data from the pointed instruments. The WGG is located in the sail and operates in synchronism with the DME and DSM located in the wheel.

The word gate generator consists basically of a *main frame generator* and a *subframe generator* which have the capability of being prewired to supply any 16 of the 32 main-frame gates and any 12 of the 48 subframe gates (Fig. 9-13). The main-frame output gates use quad redundancy and both the main-frame and subframe generators have a backup unit. The backup unit can be selected by the command system. Fig. 9-13 shows how the WGG interfaces with the data handling and command systems.

Fig. 9-13 Word Gate Generator Interface Diagram

A functional block diagram of the *main-frame gate generator* is shown at the top of Fig. 9-14. The clock signal and word gate 32 are obtained from the DME and used to generate and synchronize the 32 sail gate pulses with the spacecraft data handling system. The clock signal is first applied to a buffer amplifier (B_1) for signal isolation and conditioning and then to the synchronous counter. The synchronous counter divides the 800 cps clock signal down to 3.125 cps and thus provides the timing signals that control the output gate drivers. The counter is connected for synchronous operation to eliminate the propagation delay that is inherent in a normal ripple-through counter. The gate drivers provide the 32 sequential drive pulses that control the output gates. Any 16 gate drivers can be selected to drive the 16 output gates. Word gate 32 is fed through an isolation amplifier where it is differentiated. The trailing edge of word gate 32 resets all counter stages to the main-frame gate 1 state which synchronizes the counter with the DME main-frame gate 1.

A functional block diagram of the *subframe gate generator* is shown at the bottom of Fig. 9-14. The first sync pulse (SASC-48) from the sail ASC is fed through a buffer amplifier which isolates the subframe gate generator from other users of this pulse (e.g., the DME) and provides the necessary drive to reset the ripple counter flip-flops to count 48. The resetting of the counter to 48 synchronizes the subframe generator with channel 48 in the DSM. Main-frame word gate 29 is supplied by the main-frame gate generator (discussed above). This gate enables the diode matrix and also advances the counter at the DSM rate. At the end of 48 DSM word periods, the ripple counter resets itself and thus becomes independent of the sail ASC sync pulse.
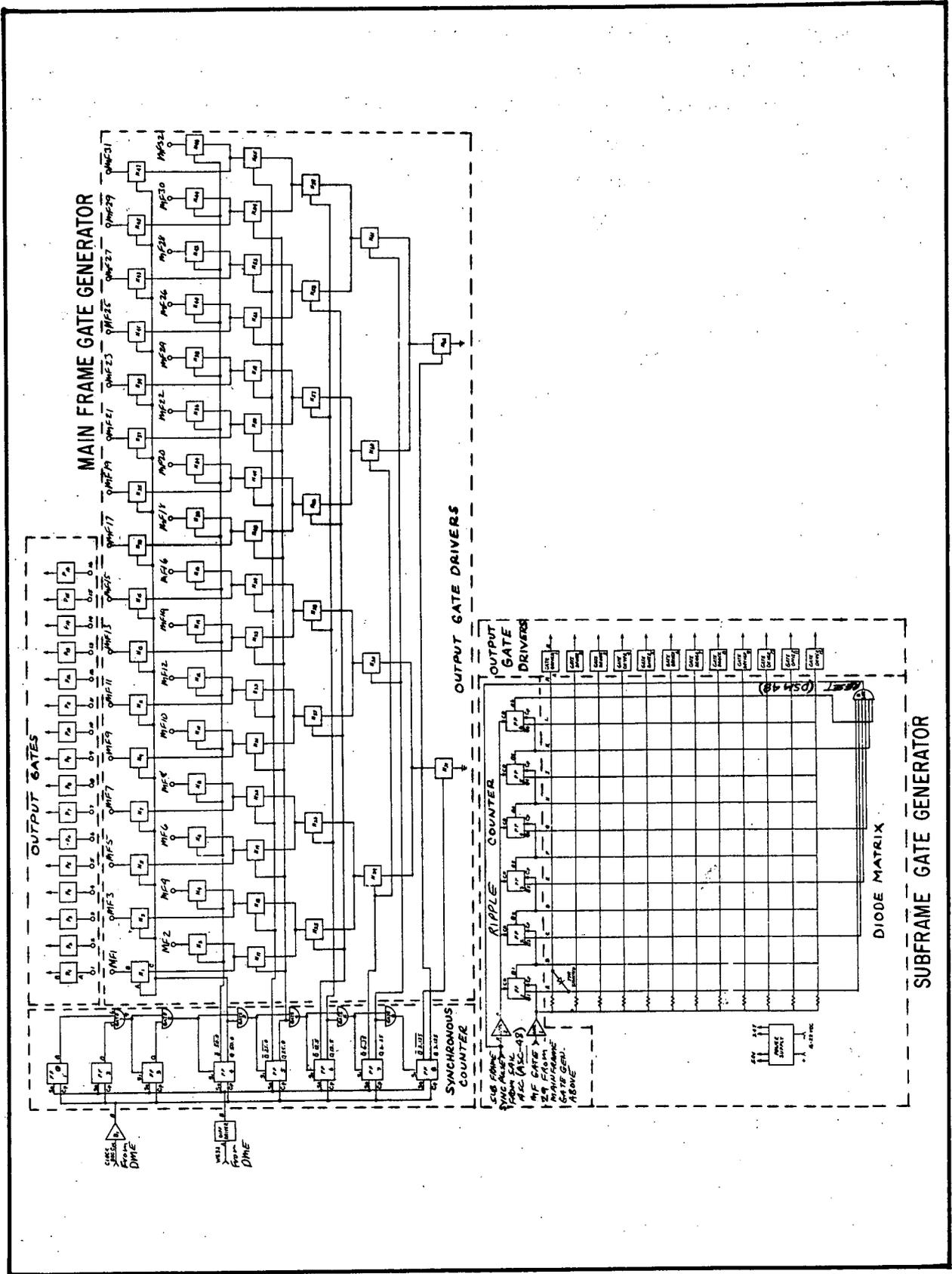
Fig. 9-14 Word Gate Generator Functional Diagram

The WGG is powered by its own internal power supply. This power supply furnishes regulated 5 $\pm$ 0.1 vdc and consists basically of a series regulator, an inverter, and two output rectifiers. The 19 $\pm$ 3 vdc input is first filtered to suppress high frequency spikes that might damage the regulator transistors and to prevent high frequency noise from the inverter being reflected back onto the 19 vdc bus line. The series regulator then regulates the 19 volt input to about 13.7 volts which is fed to the inverter. The inverter chops this voltage into a 3 Khz signal which is then rectified to provide the 5 volt output. Since regulation is not critical and because the major load is fixed, output regulation is not used.

The WGG uses circuitry essentially identical to that portion of the DME which generates data gates for the pointed instruments. The WGG only results in another series element for the overall spacecraft. The present spacecraft probability of success is R_T = 0.824 (from BBRC TR66-64). Thus the spacecraft reliability with redundant WGG's added is:

$$R_T = (0.824)(0.99909) = 0.82325$$

9.2.10   Sail Junction Box

The sail J-box differs from the PCM J-box in the wheel in that it serves only to distribute the various power, command, and telemetry signals used by the various sail electronic components and the pointed instruments. It does not, however, perform any multiplexing chores as does the PCM J-box. The sail J-box also houses the pointed instruments' power control relays, the pitch backup arming relay, and the offset pointing control relays. Finally, it serves as the attach point for the "flex cables" that carry the various signals to and from the pointed instruments.

<u>Sail J-Box Changes</u>. The OSO-5 sail J-box was sheet metal
construction with two solder terminal boards hardwired into
the box, one in the top and one across the rear, inside the
box. The box was hard-wired into the sail harness. It had
Bendix pygmy connectors on either end for mating with boxes
in the sail and 37 pin connectors for mating with the flex
cables.

The arrangement of the two terminal boards made rework extremely
difficult and all J-box rework had to be performed with the box
on the spacecraft.

The OSO-6 sail J-box is fabricated from standard sheet metal
and terminal board stock. It has five solder-terminal type
plug-in boards which insert vertically from the top. Each
plug-in board mates with the central distribution terminal
board in the bottom of the J-box by means of double density
connectors. On top of the plug-in boards are double density
connectors for mating with the flex cables. On each end of
the plug-in boards are connectors for mating the harness on
either side of the sail.

Each plug-in board can be removed and replaced by a spare
while it is being reworked on the bench. The central distri-
bution board in the bottom of the box can be reworked on the
bench by simply unplugging the harness connectors and unbolt-
ing the box from the sail.

In addition, the OSO-6 sail J-box uses Teledyne TO-5 size
relays which are potted and wired directly to the plug-in boards.
They are smaller and have considerably less mass than the
Potter Brumfield relays used on OSO-5.

9.3 RF SUBSYSTEM

9.3.1 General

The RF subsystem consists of two redundant telemetry transmitters, a coaxial relay, an RF filter, a diplexer, a hybrid circulator and an antenna array. A block diagram of this subsystem is presented as part of Fig. 9-1.

The telemetry transmitters provide phase-shift keyed (PSK) modulation using the Manchester-coded input data. Power to the redundant transmitters, both of which operate on the same frequency, is controlled by ground commands. The coaxial relay, which is used to connect the desired transmitter to the diplexer, is also operated by these commands.

The diplexer provides a low-attenuation path between the selected transmitter and antenna at the transmitter carrier frequency while providing high attenuation at the command receiver frequency. Thus, any high-power RF signals originating from the transmitter, that are at the command receiver frequency, do not reach the command receivers and cause possible damage. Conversely, the diplexer provides low attenuation between the antenna and the receiver when commands are sent to the spacecraft.

The hybrid circulator is used as a power divider for the two command receivers while providing high isolation between the two receivers. A common antenna array is used for transmitting telemetry signals and receiving command signals.

9.3.2   Transmitters

OSO-6 used the same type transmitter (BBRC-PSK-600A) that was used in OSO-5. The only difference was the addition of a modulation filter that was used to decrease the bandwidth during playback. This modification is described below. A block diagram of the PSK-600A is shown in Fig. 9-15. The electrical and mechanical characteristics of the PSK-600A transmitter are given in Table 9-1.

Operation. The modulator driver conditions the Manchester-coded input data from the tape recorders or DME's and establishes a fixed voltage level and thus the phase shift of the reference signal for a "zero" or a "one" input on the information line. The output of the modulator driver is fed to the phase modulator.

A 34 MHz crystal oscillator is used to establish a stable reference for the transmitter. The output level for this oscillator is -3 dbm. The crystal oscillator is followed by a phase modulator that changes the phase of the reference signal in response to the input data supplied by the modulator driver. Some loss is experienced in the phase modulator, and the output signal level becomes approximately -10 dbm.

A buffer amplifier follows the phase modulator and provides isolation between the modulator and the subsequent multipliers, and raises the signal level to +10 dbm. This level is required to overcome the 0.7 v base-to-emitter drop of the subsequent multiplier stage and drive this stage in to class C operation. Class C operation is necessary for the multiplier to establish a nonlinear waveform on the output of the multiplier so that the second harmonic can be extracted by a bandpass filter.
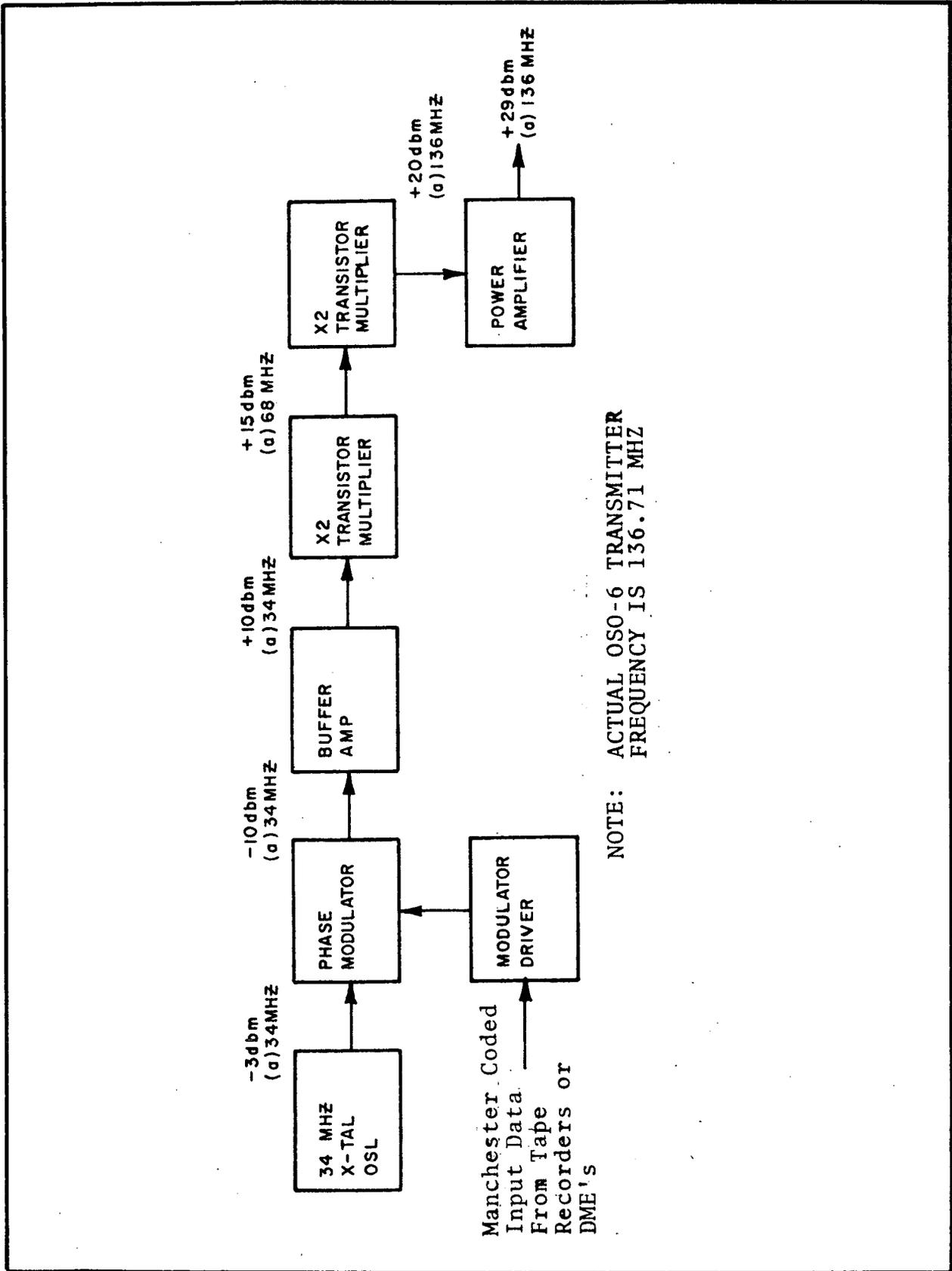
Fig. 9-15 Telemetry Transmitter (PSK-600A) Block Diagram

Table 9-1

SPECIFICATIONS FOR THE PSK-600A TELEMETRY TRANSMITTER

ELECTRICAL CHARACTERISTICS

(1) Frequency: 136 to 137 MHz (fixed tuned at 136.71 MHz)

(2) Frequency Stability $\pm$0.0037 percent

(3) Power Output: 700 mw (minimum, PSK-600A)

(4) DC/RF Efficiency: 25 percent (minimum)

(5) Modulation: Phase shift keyed

(6) Frequency Response: 400 to 15,000 bps

(7) Modulation Sensitivity: 1.75 to 6 v P-P to cause the specified phase deviation

(8) Deviation: Selectable from 50 to 90 deg. peak $\pm$5 deg. (nominally 60 deg. on OSO-6)

(9) Input Impedance: 10,000 ohms

(10) Coherent Spurious Radiation: All harmonics in the range from 30 to 500 MHz are at least 40 db below the carrier

(11) VSWR: The transmitter can operate into a 2:1 all-phase load with no spurious outputs greater than 60 db below the carrier and with a maximum power variation of 4 db. The transmitter can also survive a 5:1 all-phase load without damage.

(12) Power Supply: 16 to 22 v (19 v nominal)

(13) Incidental AM: 5 percent maximum

(14) Incidental FM: $\pm$5 deg. for pulse period at 1/Bit Rate

MECHANICAL CHARACTERISTICS

(1) Size: 3.125 by 1.75 by 3.25 in.

(2) Weight: 1 lb. maximum

The output of the first times-2 multiplier is at a +15 dbm level and at a frequency of 68 MHz. This signal drives a second times-2 multiplier that is identical to the first multiplier, with the exception that the output stage is tuned to the second harmonic or about 136 MHz (136.71 MHz on OSO-6). The signal level at this point in the transmitter is about +20 dbm. The final times-2 multiplier is followed by a power amplifier which raises the 136 MHz signal to about +29 dbm (750 mw) at 19.0 volts supply input, a level suitable for transmission.

Modulation Filter Modification. Previous measurements of the PSK-600A transmitter spectrum indicated that the spectrum was not in conformance with NASA Aerospace Data Systems Standards (X560-63-2). To minimize this nonconformance, the technique of pre-modulation filtering was used. This technique limits the bandwidth of the data signal which in turn limits the bandwidth of the transmitted signal to about 90 MHz during playback.

The pre-modulation filter is an active filter with a cut-off frequency of approximately 35 KH$_z$. Functionally, the filter module consists of a shaping circuit followed by a five pole active filter. The supply voltages for these circuits are derived from zener-diode supplies. An EMI filter was added to minimize the amount of conducted 136 MH$_z$ on the spacecraft power lines.

Physically, the filter module is contained in an aluminum case 4.25 inches by 3.50 inches by 1.00 inch. The case is mounted on the inside of the rim panel in compartment eight and is wired directly to transmitter 1.

9.3.3 Coaxial Relay

The coaxial relay is a single-pole double-throw magnetic latching relay that requires a +19-volt pulse of 30-milli-second duration to operate. This relay closes in the position corresponding to which transmitter is selected. This relay is extremely small with a volume of approximately one cubic inch not including connectors.

9.3.4 Diplexer

As was mentioned earlier, the purpose of the VHF diplexer is to permit simultaneous operation of the telemetry transmitters and command receivers from a common antenna array. The diplexer and RF filter are incorporated into the same assembly.

An M-derived high-pass type filter using lumped constants is used to provide a minimum of 20 db of attenuation of any signal from the transmitter which is at the receiver frequency. At the same time, this filter allows the transmitter frequency to pass with a maximum attenuation of 0.25 db.

Two low-pass filters, used between the antenna and receiver terminals, provide a minimum of 50 db of attenuation at the transmitter frequency while allowing the receiver frequency to pass with less than 0.7 db loss.

9.3.5 Hybrid Circulator

The hybrid circulator permits simultaneous operation of two command receivers on the same frequency. A lumped-constant bridge circuit is used to provide isolation between the two receiver terminals in excess of 40 db. The input signal is divided equally between the two receiver terminals within ±0.2 db with an insertion loss of less than 0.5 db.

9.3.6 Antenna Array

Since the OSO is not stabilized with respect to the earth, nearly-isotropic radiation patterns are required. This nearly-isotropic radiation is accomplished with a three-element array. When the radiation pattern is measured by two orthogonally-polarized receiving antennas, the field sensed by the vector sum of the antennas is greater than -6 db for 95 percent of the aspect angles. Field plots of these two components, Eθ and Eϕ, are shown in Figs. 9-16 and 9-17, respectively. The figures are equal-power-contour plots reflecting the antenna gain with reference to a linear isotropic source. Refer to Figs. 9-18 and 9-19 for the coordinate definition for all radiation patterns.

The same three-element array is used for command reception at approximately 150 MHz; however, the performance characteristics are somewhat different because command transmission is right circularly polarized. Therefore, radiation field measurements must be sensed with a right circularly polarized antenna. For command reception, the measured gain is greater than -18 db with respect to a linear isotropic source for 95 percent of the aspect angles. Figure 9-20 is an equal-power contour plot of the antenna at the receiver frequency of approximately 150 MHz. Again, the plot reflects the gain with reference to a linear isotropic source.

Figure 9-21 is a view showing one antenna element. Each of the arms on OSO is identical which results in a three-element array. Each antenna element consists of an arm support bracket and a stub. The support bracket also serves its primary function of supporting the arm. Each stub is electrically connected to its respective support bracket to form an element having an electrical length of one-half wavelength. Excitation is applied at the point where the support bracket connects the OSO arm.

Fig. 9-16 OSO-6 Transmitter Antenna Pattern (Eθ Polarization)

Fig. 9-17 OSO-6 Transmitter Antenna Pattern (Eφ Polarization)

θ = +90°

Z

P

Eθ

Eφ

φ=270° Y'

X'

φ=180°

θ

φ = 0°

RUTGERS
LIGHT
SHIELD

φ=90°

X

Y

θ=0°

Z'

θ = −90°

VEHICLE TYPE OSO

FED ELEMENTS AT θ = 90°, θ = −30° AND φ =180°
THE BOTTOM OF THE WHEEL (ATTACH FITTING) AT θ = 0°, φ= 90°
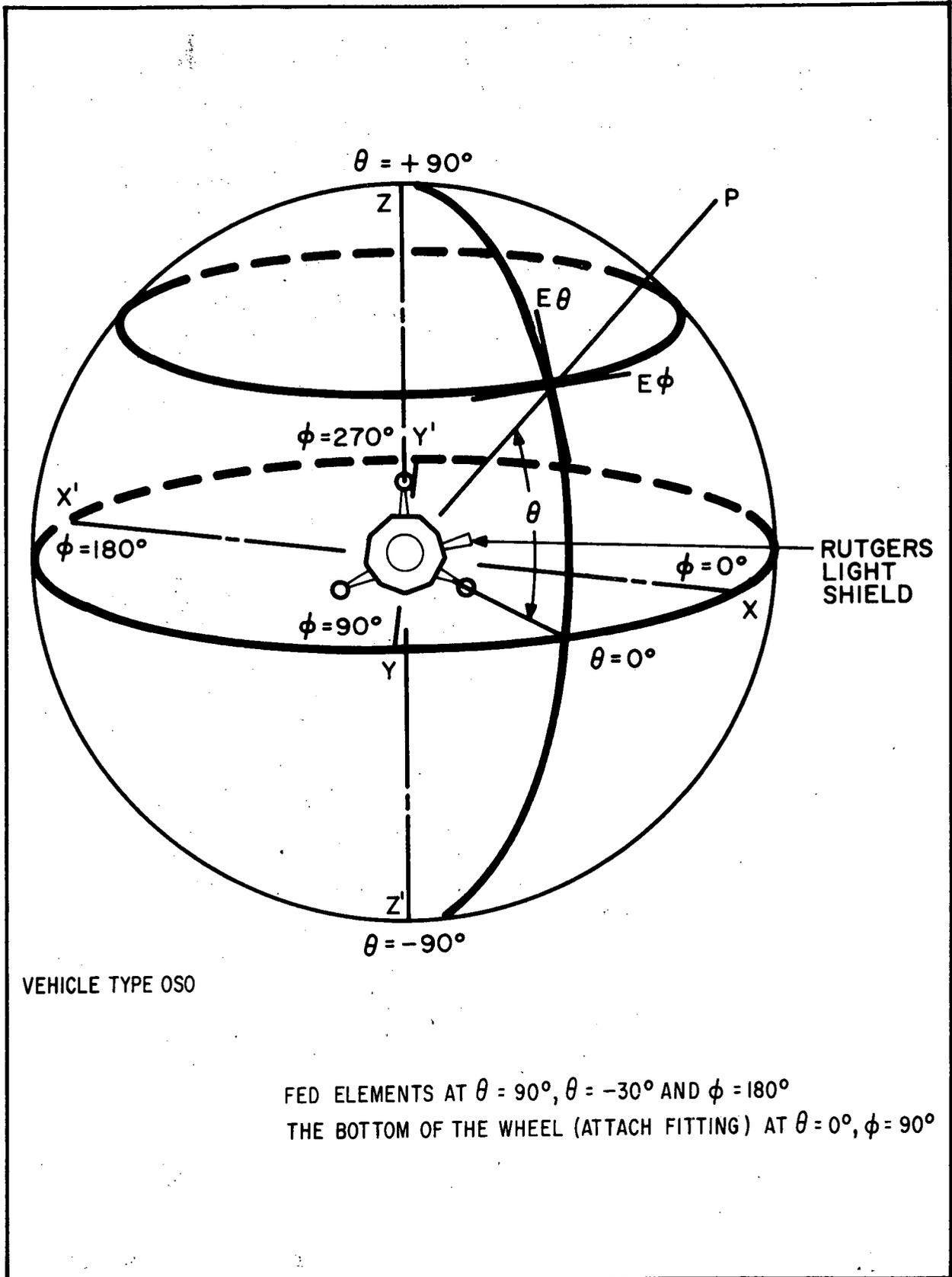
Fig. 9-18   Coordinate System for Radiation Pattern
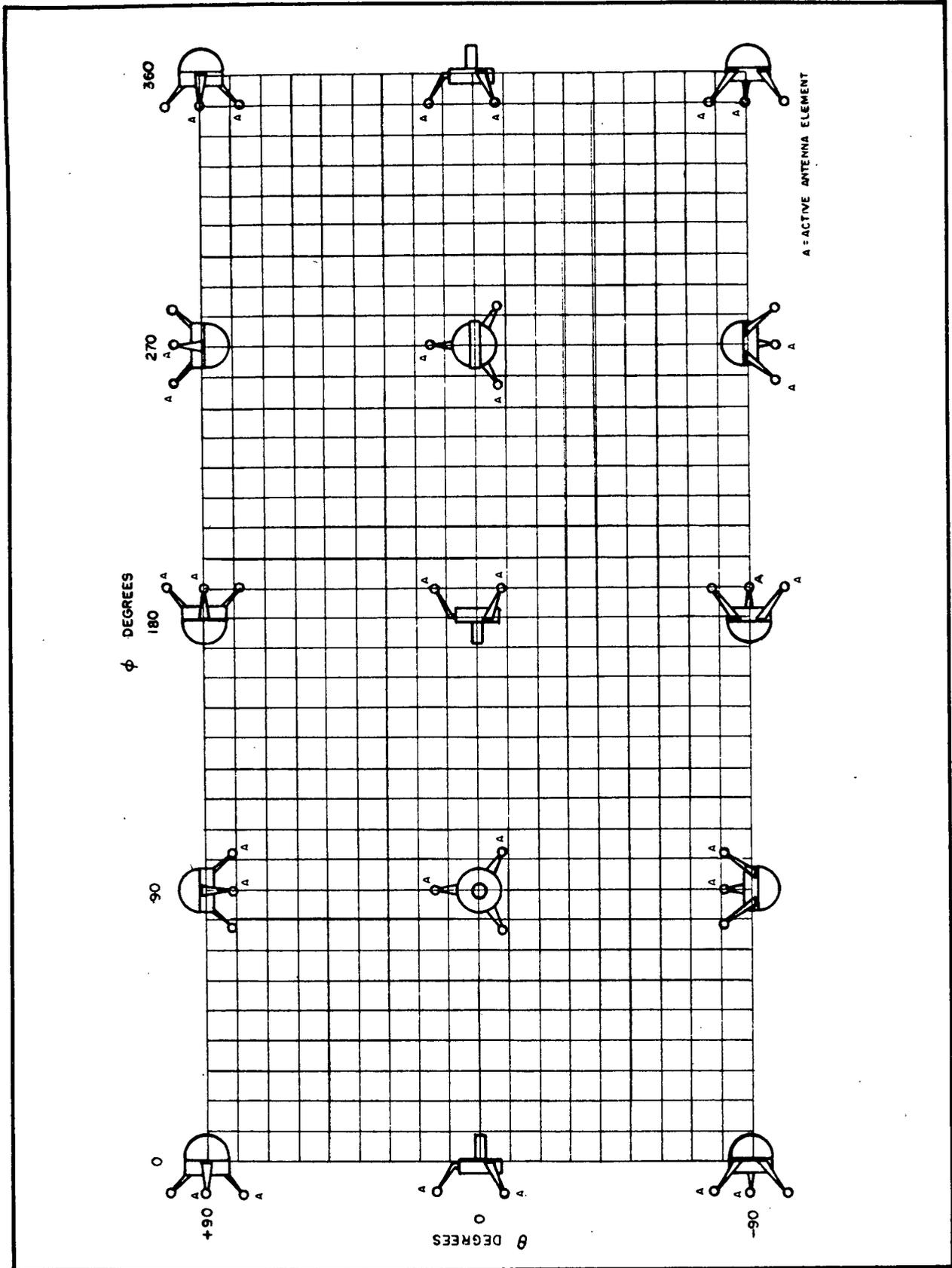
Fig. 9-19   Orientation of OSO Relative to Contour Plots

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DATE 11-27-68

VEHICLE TYPE OSO-G MODEL SCALE I

MODEL FREQUENCY 149.52 MHz

POLARIZATION RT CIRCULAR
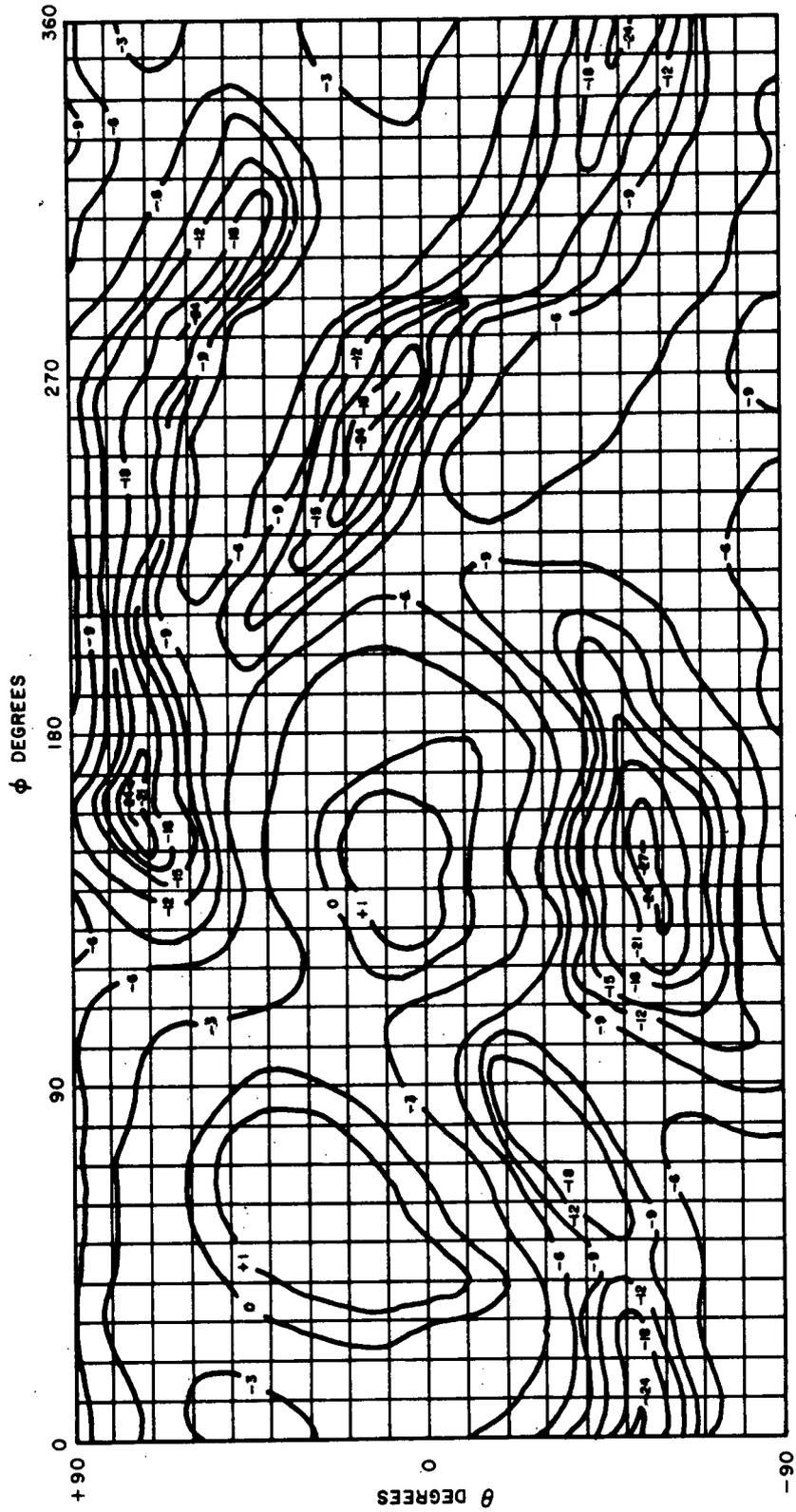
Fig. 9-20 Receiver Antenna Pattern, Right-Circular Polarization

SUPPORT LINK

EXCITATION POINT

SUPPORT LINK
INSULATED FROM SATELLITE
(TYPICAL EACH ARM)

STUB

ARM DETAIL
(SEE ENLARGED VIEW)
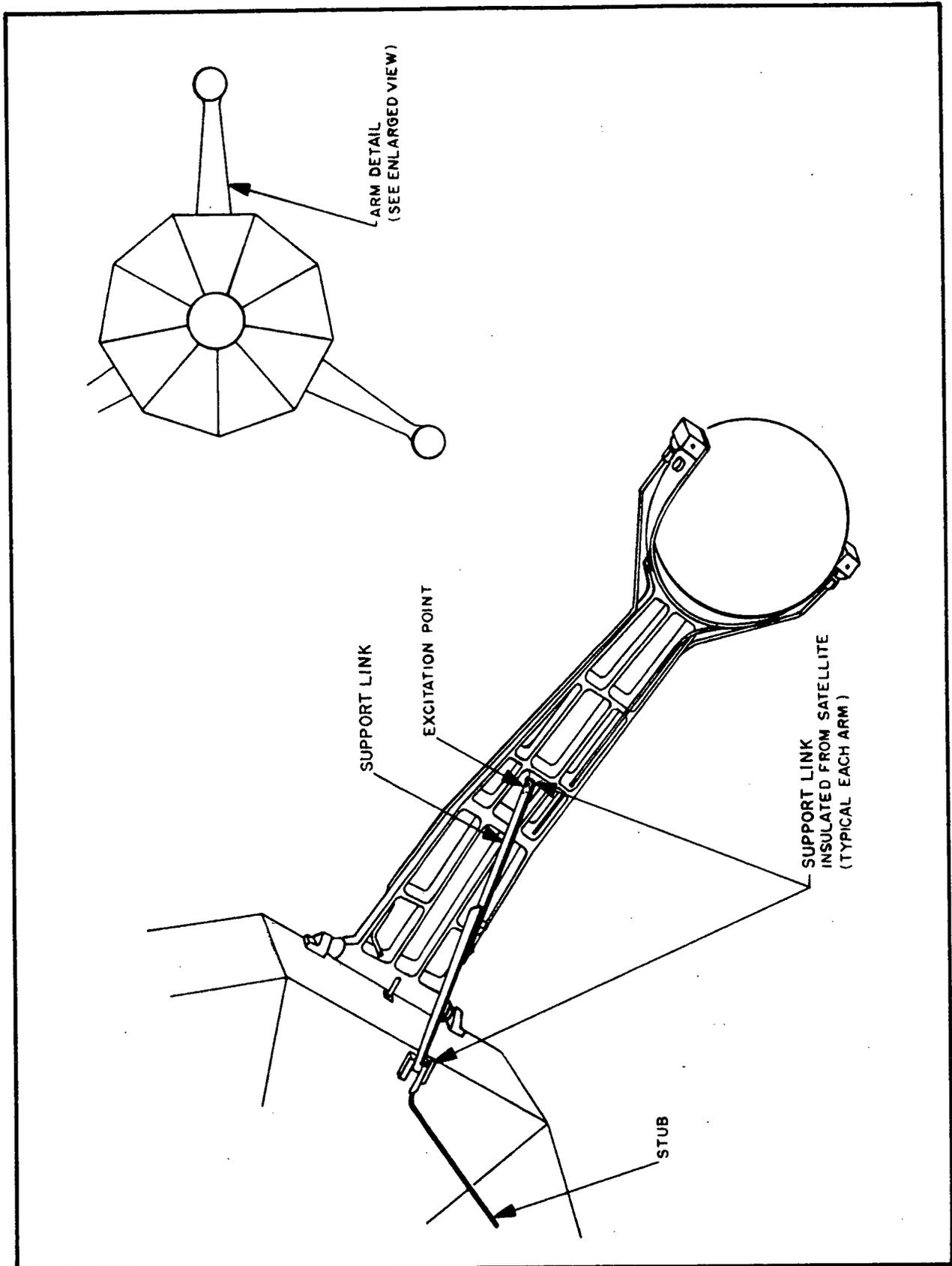
Fig. 9-21 Antenna and Support Bracket Details

The three antenna elements are insulated from the structure by the use of dielectric materials such as fiber glass and nylon.

When the arms are erected, two elements are directly excited and the third element is parasitic. When the arms are stowed, all three elements are driven.

9.4   SPACECRAFT HOUSEKEEPING DATA ACQUISITION

Spacecraft housekeeping data, including performance monitors and status monitors, are read out through the DSM, the two analog subcommutators, and the analog sub-subcommutator.

The input voltages to the analog subcommutators range from zero to +5 v. These monitors include temperatures, voltages, current levels, and others which are not readily converted to digital form in the sensing subassemblies. Conversion is done in the DME's.

The concept of single-bit status monitoring is used in the digital subframe to provide a means by which up to eight status points can be read through a single subframe word. This technique allows more analog channels for experimenter use and provides for simpler machine interpretation of status-monitor data.

Other spacecraft data read through the digital subframe include the aspect (SORE) data and the frame count.

9.5 TELEMETRY SYSTEM PERFORMANCE REQUIREMENTS

The performance of the communications link between the OSO and
the GSFC Satellite Tracking and Data Acquisition Network
(STADAN) is dependent upon both the airborne and the ground
equipment. Table 9-2 shows the gain margin of the communi-
cations link for playback transmission of 14400 bps at a
range of 1150 statute miles. This table also lists the equip-
ment parameters used to establish the gain margin. The gain
margin for real-time transmission at 800 bps is not included
in the table.

The post-detection bandwidth for real-time transmission can be
much narrower than for playback transmission. Therefore, the
output signal-to-noise ratio can be expected to be higher
resulting in a gain margin which exceeds the margin established
for playback transmission.

Computation of Path Loss:

$$\text{Path Loss} = \frac{1}{\left(\dfrac{4\pi\ f_{mc}\ d}{300}\right)^2} = \frac{300^2}{[(4\pi)\,(136.71)\,(1150 \text{ stat. mi.} \times 1609 \text{ m/mi})]^2}$$

$$= \frac{300^2}{(3180 \times 10^6)^2} = \frac{300^2}{(3.18 \times 10^9)^2} = \frac{9 \times 10^4}{9.96 \times 10^{18}} = 0.904 \times 10^{-14}$$

$$= -150 + 9.04 = -140.95 \text{ db}$$

Table 9-2

OSO-6 Telemetry Link Gain Margin at 14400 BPS

Radiated Power (at antenna terminals)	Power Ratios
500 milliwatts	+27 dbm
Receiver Noise Level	
1. Mod. 1 receiver with I.F. bandwidth of 100 kHz and noise figure of 2 including preamp	-116.3 dbm
2. Antenna noise temperature: 1400°K	
Required signal-to-noise ratio (CNR)*	10.6 db
Miscellaneous losses allowed for tracking errors, transmission line losses and other miscellaneous losses	-4 db
Path Loss	
1. Range: 1150** statute miles	
2. Frequency: 136.71 mHz	-141 db
Receiver Antenna Gain	
1. Nine Yagi array	
2. Polarization: horizontal, vertical, right, and left circular	+19.2 db
Transmitter Antenna Gain	
Polarization: linear	-6 db
Margin of Safety (worst predicted case)	
= S/N ratio threshold + gains - losses	
= (116.3 - 10.6) + (27 + 19.2)	
- (141 + 4 + 6) = 0.9 db	

*   The CNR has been increased about 3 db to allow for the 14400 bps bit rate.

** This is worst case since 1150 miles is an unlikely range for playback. 975 mi. is more reasonable and this decreases the path loss about 1.5 db.

Section 10
COMMAND SYSTEM DESCRIPTION

10.1   INTRODUCTION

The OSO-6 command system consists of two redundant VHF-AM
command receivers, two redundant wheel decoders, two sail
decoders, and the necessary wiring and control relays (Fig.
10-1). The command system provides a total complement of 206
command functions (67 for the spacecraft, 108 for the experi-
ments, and 31 spares). Each wheel decoder is driven from one
command receiver while either sail decoder can be connected to
either of the two receivers by ground command. The RF input
is identical for each receiver, since the RF input power is
divided equally by a hybrid circulator. All receivers and
decoders have dc power applied continuously.

The spacecraft uses a STADAN compatible tone-digital command
system. The command RF signal (149.52 MHz) is 80 percent
amplitude modulated by a 7 kHz audio tone. The tone is keyed
by the coded digital-command signal.

The main differences between the OSO-5 and OSO-6 command
systems is the addition of a sail command decoder and the
desensitizing of the command receivers.

10.2   COMMAND FORMAT

The minitrack ground operator transmits commands to the OSO by
either a punched paper tape mode or a manual mode. The paper-
tape mode, which is normally used, allows the transmission of
as many commands as desired in sequence, and the format consists
of: address, execute, 1/2-second delay. As a standard proce-
dure, the command frame is transmitted twice.
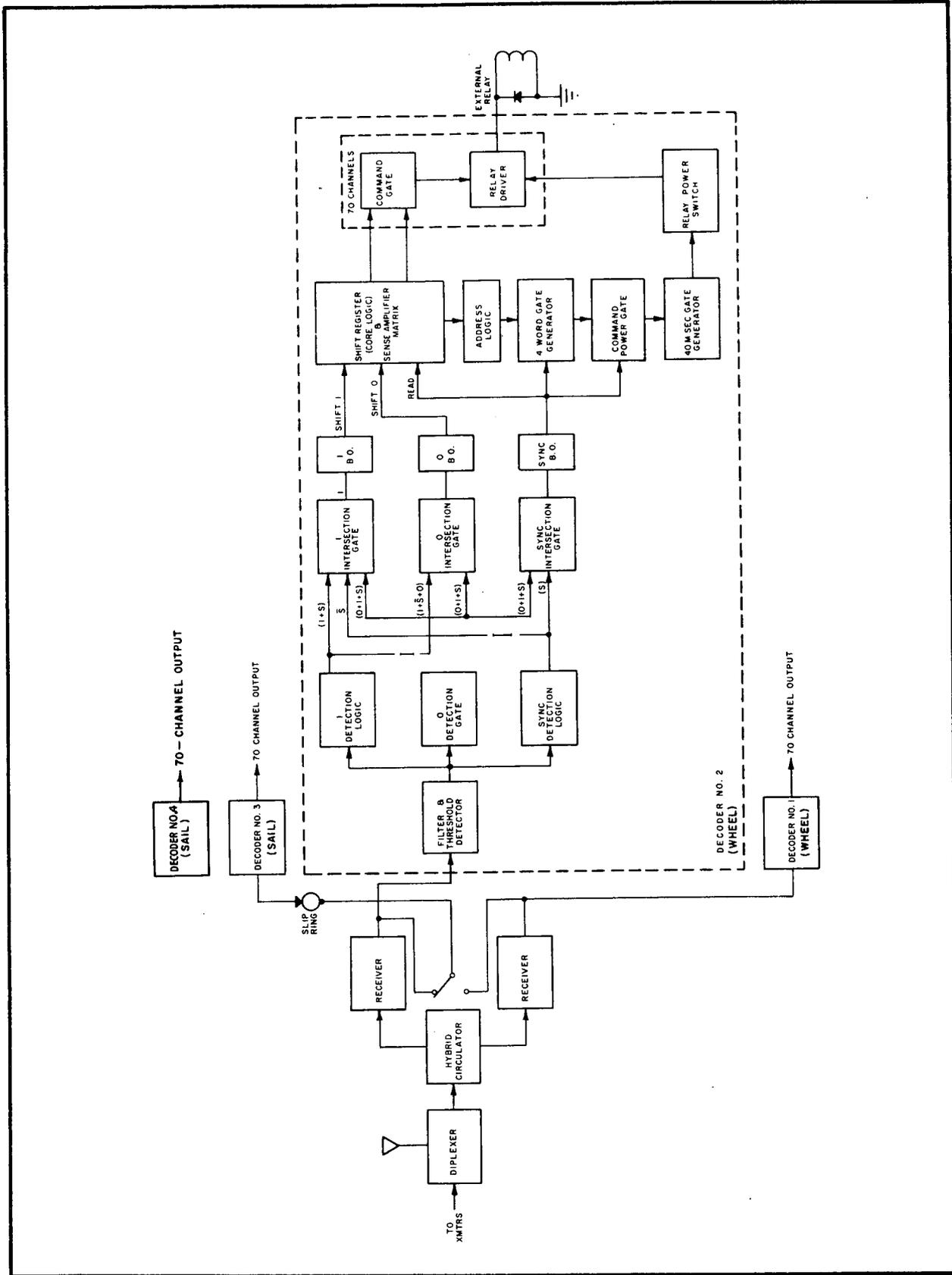
Fig. 10-1   Command System Block Diagram

In emergency situations, the manual mode can be used. This format consists of: address, address, execute, execute, execute.

As the name tone-digital implies, digital information is contained in the pulse duration of the 7 kHz tone (Fig. 10-2). This tone is pulse-duration modulated (PDM) such that three distinct tone bursts occur: 54, 36, and 18 cycles.

A bit period is defined as a time duration equal to 72 cycles of the carrier tone. A sync pulse is defined as a tone burst of 54 cycles in length. Tone bursts of 36 and 18 cycles in length define a "one" and "zero", respectively. A word consists of 10-bit periods in the following order: eight information bits of "ones" and "zeros" followed by a blank bit and a sync pulse. Address words and command words have the same format. They differ only in the combinations of "ones" and "zeros". The arrangement of information bits in an address word is either six "ones" and two "zeros", or six "zeros" and two "ones". The arrangement of information bits in a command word is four "ones" and four "zeros". The period of each 10-bit command or address word is about 0.103 seconds.

Therefore, the output of the minitrack command encoder is a serial train of tone bursts that amplitude-modulate the command transmitter. The OSO command receivers detect this serial train of tone bursts and deliver it to the command decoders. The decoder that is addressed is gated "on", decodes the command word, and outputs a pulse on one of its 70 output lines. The commands used on OSO-6 are listed in Table 10-1.

In Table 10-1, note that there are six different series used in numbering the OSO-6 commands:

COMMAND FRAME

ADDRESS WORD	INSTRUCTION WORD	INSTRUCTION WORD	INSTRUCTION WORD	END OF FRAME
10 BIT INTERVAL	10 BIT INTERVAL	10 BIT INTERVAL	10 BIT INTERVAL	2 BITS

TYPICAL INSTRUCTION WORD WHERE [///] DENOTES PRESENCE OF TONE

SPACE	SYNC	1	0	1	0	0	1	1	0

DEFINITIONS:

1. BIT INTERVAL : TIME DURATION EQUIVALENT TO 72 CYCLES OF THE ASSIGNED TONE

2. WORD INTERVAL : TIME DURATION EQUIVALENT TO 10 BIT INTERVALS

3. FRAME INTERVAL : TIME DURATION EQUIVALENT TO 5 WORD INTERVALS PLUS 2 BITS

4. SPACE: ABSENCE OF TONE FOR 1 BIT INTERVAL

5. SYNC: PRESENCE OF TONE FOR DURATION EQUIVALENT TO 54 CYCLES OF THE ASSIGNED TONE

6. ONE: PRESENCE OF TONE FOR DURATION EQUIVALENT TO 36 CYCLES OF THE ASSIGNED TONE

7. ZERO: PRESENCE OF TONE FOR DURATION EQUIVALENT TO 18 CYCLES OF THE ASSIGNED TONE

8. ADDRESS WORD: WORD STRUCTURE WHICH MUST CONTAIN 6 ONES AND 2 ZEROS OR 6 ZEROS AND 2 ONES

9. INSTRUCTION WORD: WORD STRUCTURE WHICH MUST CONTAIN 4 ONES AND 4 ZEROS

10. END OF FRAME: TWO BITS (SPACE/SYNC) USED TO CLEAR DECODER AND WHICH FOLLOWS ANY COMMAND TRANSMISSION REGARDLESS OF DURATION

Fig. 10-2 OSO Command Format

Table 10-1
OSO-6 COMMAND ALLOCATIONS

WHEEL COMMANDS

CMMD NO.	COMMAND FUNCTIONS	CODE FORMAT
1	PBK TR I / TMG DISAB-OFF/DL V ENAB	0015
2	SPIN UP/ASL DISAB	0107
3	UCL PWR OFF	0111
4	RF OFF (C/W 62) XMTR 2 DISAB	0113
5	DME I ON/DME 2 OFF	0114
6	UVS NOT BYP/TMG ENAB/DL OFF	0207
7	SAIL DAY PWR ENAB/ PITCH COIL POS	0211
8	BOLOGNA PWR ON	0213
9	XMTR I ON/XMTR 2 OFF	0214
10	D-N RELAY BYP/ASL DISAB/SPIN MANUAL	0303
11	BOLOGNA HV ON	0305
12	LASL PWR OFF	0306
13	LASL BIAS LEVEL OO	0309
114	PITCH COIL HALF PWR	0310
214	LSBU DISAB/R SQ DISARM/DL CMMD ENAB	0310
15	PITCH COIL OFF	0312
16	XMTR 2 ENAB	0407
17	TR 2 PWR ON	0411
18	BOLOGNA HV OFF	0413
19	UNM PWR ON	0414
20	BOLOGNA CALIB I ON	0503
21	D-N RELAY NOT BYP/ASL ENAB	0505
22	LASL HV I ON	0506
23	UCL HV OFF	0509
124	UCL CALIB ON - SQ DET	0510
224	UCL HV I ON	0510
125	UCL HV 2 ON	0512
225	UCL HV 3 ON	0512
26	LASL PWR ON	0603
27	BOLOGNA CALIB 2 ON	0605
128	PITCH COIL FULL PWR	0606
228	LSBU ENAB/R SQ ARM/INT PULSE ON	0606
29	UVS BYP (C/W 38)	0609
30	ORBIT PWR RELAY BYP	0610
31	BOLOGNA CALIB OFF	0612
32	SPIN DOWN/ASL ENAB	0701
33	UNM CALIB ON	0702
34	UNM CALIB OFF	0704
35	RUTGERS PWR OFF	0708
36	RUTGERS PWR ON	0807
37	BOLOGNA ANTICOINCIDENCE ON	0811
38	UVS BYP CMMD ENAB	0813
39	SPIN AUTO/ASL ENAB	0814
40	PITCH COIL NEG	0903
41	TR I PWR ON	0905
42	RUTGERS FORWARD MODE	0906
43	RUTGERS REVERSE MODE	0909
44	RUTGERS HV ON	0910
45	RUTGERS HV OFF	0912
46	RUTGERS MANUAL SEQUENCE	1003
47	RUTGERS AUTO SEQUENCE	1005
48	RUTGERS SINGLE STEP	1006
49	BOLOGNA ANTICOINCIDENCE OFF	1009
50	LASL HV 2 ON	1010
51	PITCH COIL ON	1012
52	UCL PWR ON	1101
53	SAIL DAY PWR DISAB	1102
54	TR 2 PWR OFF	1104
55	PBK OFF/TMG ON	1108
56	LASL HV OFF	1203
57	TR I PWR OFF	1205
58	UCL SQ ARM/RCVR I - DCDR 4, 2-3	1206
59	UCL CALIB OFF/SQ DISARM/INT PULSE ON	1209
60	LSBU EXCT/LST OFF/R SQ DET-OR-DL ON	1210
161	LASL CALIB ON	1212
261	LASL CALIB OFF	1212
62	XMTR 2 ON (C/W 16) XMTR I OFF	1301
63	BOLOGNA PWR OFF	1302
164	LASL BIAS LEVEL OI	1304
264	LASL BIAS LEVEL IO	1304
65	UVS BYP CMMD DISAB/RCVR I - DCDR 3, 2-4	1308
66	DME 2 ON / DME I OFF	1401
167	PBK ON REDT/TMG DISAB-OFF/DL V ENAB	1402
267	ORB PWR RELAY NOT BYP	1402
68	UNM PWR OFF	1404
69	SPIN MANUAL/ASL DISAB	1408
70	PBK TR 2 /TMG DISAB-OFF/DL V ENAB	1500

NOTE: CMMDS I THRU 70 ARE REDT ON WHEEL DECODERS NO. I & 2
CMMDS 101 THRU 170 ARE EXECUTED THRU WHEEL DECODER NO. I ONLY
CMMDS 201 THRU 270 ARE EXECUTED THRU WHEEL DECODER NO.2 ONLY

SAIL COMMANDS - DECODER NO. 3

CMMD NO.	COMMAND FUNCTIONS	CODE FORMAT
301	NRL PM HV ON	0015
302	NRL PM & PROP COUNTER HV OFF	0107
303	NRL PROP COUNTER HV ON	0111
304	NRL BRAGG HV ON	0113
305	NRL BRAGG HV OFF	0114
306	NRL BURST HV ON	0207
307	NRL BURST & RSTR HV OFF	0211
308	NRL RSTR HV ON	0213
309	NRL PM & PROP COUNTER HV I	0214
310	NRL PM HV 2	0303
311	NRL PROP COUNTER HV 2	0305
312	NRL BRAGG, BURST, & RSTR HV I	0306
313	NRL BRAGG HV 2	0309
314	NRL BURST & RSTR HV 2	0310
315	NRL LV I	0312
316	NRL LV 2	0407
317	NRL BRAGG & BURST NORMAL-MTR FAST	0411
318	NRL BRAGG & BURST CROSSED	0413
319	NRL MTR DRIVE SPEED AUTO	0414
320	NRL MTR DRIVE SPEED SLOW	0503
321	NRL MTR DRIVE I ON	0505
322	NRL MTR DRIVE 2 ON	0506
323	NRL MTR DRIVE COILS TO I	0509
324	NRL MTR DRIVE COILS TO 2	0510
325	HCO POSITION SELECT 12	0512
326	HCO POSITION SELECT 10	0603
327	HCO POSITION SELECT 14	0605
528	NRL PWR ON	0606
529	NRL PWR OFF	0609
530	HCO LV & HV PWR OFF	0610
331	HCO WAVELENGTH SCAN	0612
532	PITCH DOWN/WG GEN I ON/WG GEN 2 OFF	0701
533	PITCH UP/WG GEN 2 ON/WG GEN I OFF	0702
534	PITCH AUTO	0704
535	HCO LV ON	0708
336	HCO OP CKT DISAB	0807
337	HCO OP CKT ENAB/RST	0811
338	HCO WAVELENGTH RESET	0813
339	HCO RESET OVERRIDE	0814
340	HCO OP CKT. DISAB	0903
341	HCO OP CKT ENAB/RST	0905
342	HCO WAVELENGTH RESET	0906
343	HCO RESET OVERRIDE	0909
344	HCO OPTICAL REFERENCE	0910
345	HCO MECHANICAL REFERENCE	0912
346	HCO OPTICAL REFERENCE	1003
347	HCO MECHANICAL REFERENCE	1005
348	HCO MTR START	1006
349	HCO POSITION SELECT I	1009
350	SPARE	1010
351	HCO POSITION SELECT 3	1012
352	HCO MTR START	1101
353	HCO POSITION SELECT I	1102
354	HCO POSITION SELECT 2	1104
355	HCO POSITION SELECT 3	1108
356	HCO POSITION SELECT 5	1203
357	HCO POSITION SELECT 7	1205
358	HCO POSITION SELECT 9	1206
359	HCO POSITION SELECT 11	1209
360	HCO POSITION SELECT 4	1210
361	HCO POSITION SELECT 8	1212
362	HCO POSITION SELECT 5	1301
363	HCO POSITION SELECT 7	1302
364	HCO POSITION SELECT 9	1304
365	HCO POSITION SELECT 13	1308
366	HCO POSITION SELECT 6	1401
567	HCO HV PWR ON	1402
568	PITCH MANUAL	1404
569	OFST MODE OFF/POINT MODE ON	1408
570	OFST MODE ON/LG RSTR ON/OFST GEN RST	1500

NOTE:
CMMDS 301 THRU 370 ARE EXECUTED THRU SAIL DECODER NO.3 ONLY
CMMDS 501 THRU 570 ARE REDT ON SAIL DECODERS NO. 3 & 4

SAIL COMMANDS - DECODER NO. 4

CMMD NO.	COMMAND FUNCTIONS	CODE FORMAT
401	EL OFST BIT I (LSB)	0015
402	SPARE	0107
403	ROLL COIL I ON	0111
404	AZ OFST BIT 6	0113
405	ROLL COIL I OFF	0114
406	ROLL COIL I POS	0207
407	AZ OFST BIT I (LSB)	0211
408	HCO WAVELENGTH SCAN	0213
409	SPARE	0214
410	SPARE	0303
411	SPARE	0305
412	ROLL COIL I NEG	0306
413	OFST POINT ON (C/W 570)/RSTR OFF	0309
414	OFST GEN RST	0310
415	SMALL RSTR ON/OFST POINT ON	0312
416	HCO HV OFF	0407
417	HCO HV DAY SELECT	0411
418	HCO HV LCH SELECT	0413
419	AZ OFST BIT 2	0414
420	ROLL COIL 2 ON	0503
421	EL OFST BIT 2	0505
422	ROLL COIL 2 OFF	0506
423	ROLL COIL 2 POS	0509
424	ROLL COIL 2 NEG	0510
425	EL OFST BIT 5	0512
426	SPARE	0603
427	AZ OFST BIT 4	0605
528	NRL PWR ON	0606
529	NRL PWR OFF	0609
530	HCO LV & HV PWR OFF	0610
431	SPARE	0612
532	PITCH DOWN/WG GEN I ON/WG GEN 2 OFF	0701
533	PITCH UP/WG GEN 2 ON/WG GEN I OFF	0702
534	PITCH AUTO	0704
535	HCO LV ON	0708
436	SPARE	0807
437	SPARE	0811
438	SPARE	0813
439	SPARE	0814
440	SPARE	0903
441	AZ OFST BIT 3	0905
442	SPARE	0906
443	SPARE	0909
444	SPARE	0910
445	SPARE	0912
446	SPARE	1003
447	SPARE	1005
448	EL OFST BIT 3	1006
449	SPARE	1009
450	SPARE	1010
451	SPARE	1012
452	SPARE	1101
453	SPARE	1102
454	SPARE	1104
455	EL OFST BIT 6	1108
456	EL OFST BIT 4	1203
457	SPARE	1205
458	SPARE	1206
459	SPARE	1209
460	AZ OFST BIT 5	1210
461	SPARE	1212
462	SPARE	1301
463	SPARE	1302
464	SPARE	1304
465	EL VERNIER OFST BIT (UP)	1308
466	AZ VERNIER OFST BIT (RIGHT)	1401
567	HCO HV PWR ON	1402
568	PITCH MANUAL	1404
569	OFST MODE OFF/POINT MODE ON	1408
570	OFST MODE ON/LG RSTR ON/OFST GEN RST	1500

NOTE:
CMMDS 401 THRU 470 ARE EXECUTED THRU SAIL DECODER NO.4 ONLY
CMMDS 501 THRU 570 ARE REDT ON SAIL DECODERS NO. 3 & 4

LEGEND:

AGC	= AUTOMATIC GAIN CONTROL	DL	= DUMMY LOAD	
APL	= AUTOMATIC PITCH LIMITER	DME	= DIGITAL MULTIPLEXER ENCODER	
ASL	= AUTOMATIC SPIN LIMITER	D-N	= DAY - NIGHT	
ASSC	= ANALOG SUB - SUBCOMMUTATOR			
AUTO	= AUTOMATIC	EL	= ELEVATION	
AZ	= AZIMUTH	ENAB	= ENABLE	
		EXCT	= EXECUTE	
BOLOGNA	= UNIVERSITY OF BOLOGNA			
BYP	= BYPASS (ED)	GEN	= GENERATOR	
CALIB	= CALIBRATE	HCO	= HARVARD COLLEGE OBSERVATORY	
CKT	= CIRCUIT	HV	= HIGH VOLTAGE	
CMMD	= COMMAND			
C/W	= CONDITIONAL WITH	ID	= IDENTIFICATION	
		INT	= INTERNAL	
DCDR	= DECODER			
DET	= DETONATE	LA	= LYMAN ALPHA	
DISAB	= DISABLE	LASL	= LOS ALAMOS SCIENTIFIC LABORATORY	

LCH	= LAUNCH	MSB	= MOST SIGNIFICANT BIT	
LG	= LARGE	MTR	= MOTOR	
LSB	= LEAST SIGNIFICANT BIT			
LSBU	= LAUNCH SEQUENCE BACKUP	NEG	= NEGATIVE	
LST	= LAUNCH SEQUENCE TIMER	NRL	= NAVAL RESEARCH LABORATORY	
LV	= LOW VOLTAGE	NUT	= NUTATION	
LWR	= LOWER			
		OFST	= OFFSET	
		OP	= OVERLOAD PROTECTION	
		ORB	= ORBIT	
		PBK	= PLAYBACK	
		POS	= POSITIVE	
		PM	= PHOTOMULTIPLIER	
		PROP	= PROPORTIONAL	
		PWM	= PULSE WIDTH MODULATION	
		PWR	= POWER	

R(RUTGERS)	= RUTGERS STATE UNIVERSITY			
RCVR	= RECEIVER			
REDT	= REDUNDANT			
REL	= RELEASE			
RF	= RADIO FREQUENCY			
RST	= RESET			
RSTR	= RASTER			
RT	= RIGHT			
SASC	= SAIL ANALOG SUBCOMMUTATOR			
SORE	= SPIN ORIENTATION & RATE ELECTRONICS			
SQ	= SQUIB			
S SW	= SEPARATION SWITCH			
TEMP	= TEMPERATURE			
TMG	= TIME MARKER GENERATOR			
TR	= TAPE RECORDER			
UCL	= UNIVERSITY COLLEGE LONDON			
UNM	= UNIVERSITY OF NEW MEXICO			
UVS	= UNDERVOLTAGE SWITCH			
V	= VOLT			
WASC	= WHEEL ANALOG SUBCOMMUTATOR			
WG	= WORD GATE			
XMTR	= TRANSMITTER			

(1) The 10 series (1 through 70) commands can be executed via either of the two wheel command decoders.

(2) The 100 series (100 through 170) through decoder No. 1 (wheel) only.

(3) The 200 series (201 through 270) through decoder No. 2 (wheel) only.

(4) The 300 series (301 through 370) through decoder No. 3 (sail) only.

(5) The 400 series (401 through 470) through decoder No. 4 (sail) only.

(6) The 500 series (501 through 570) through either decoder Nos. 3 or 4.

10.3   COMMAND RECEIVERS

The OSO command receivers use single conversion superheterodyning, and are fix-tuned to the NASA specified frequency of 149.52 MHz. The receivers are solid state, and some of their circuits are connected in series across the power supply to reduce power consumption.

The OSO command receivers are manufactured by AVCO Corporation, Cincinnati, Ohio. A block diagram of the command receiver is shown in Fig. 10-3. A list of electrical characteristics is given in Table 10-2.

The two redundant command receivers are fed from a common antenna system through a diplexer and a hybrid circulator. The diplexer
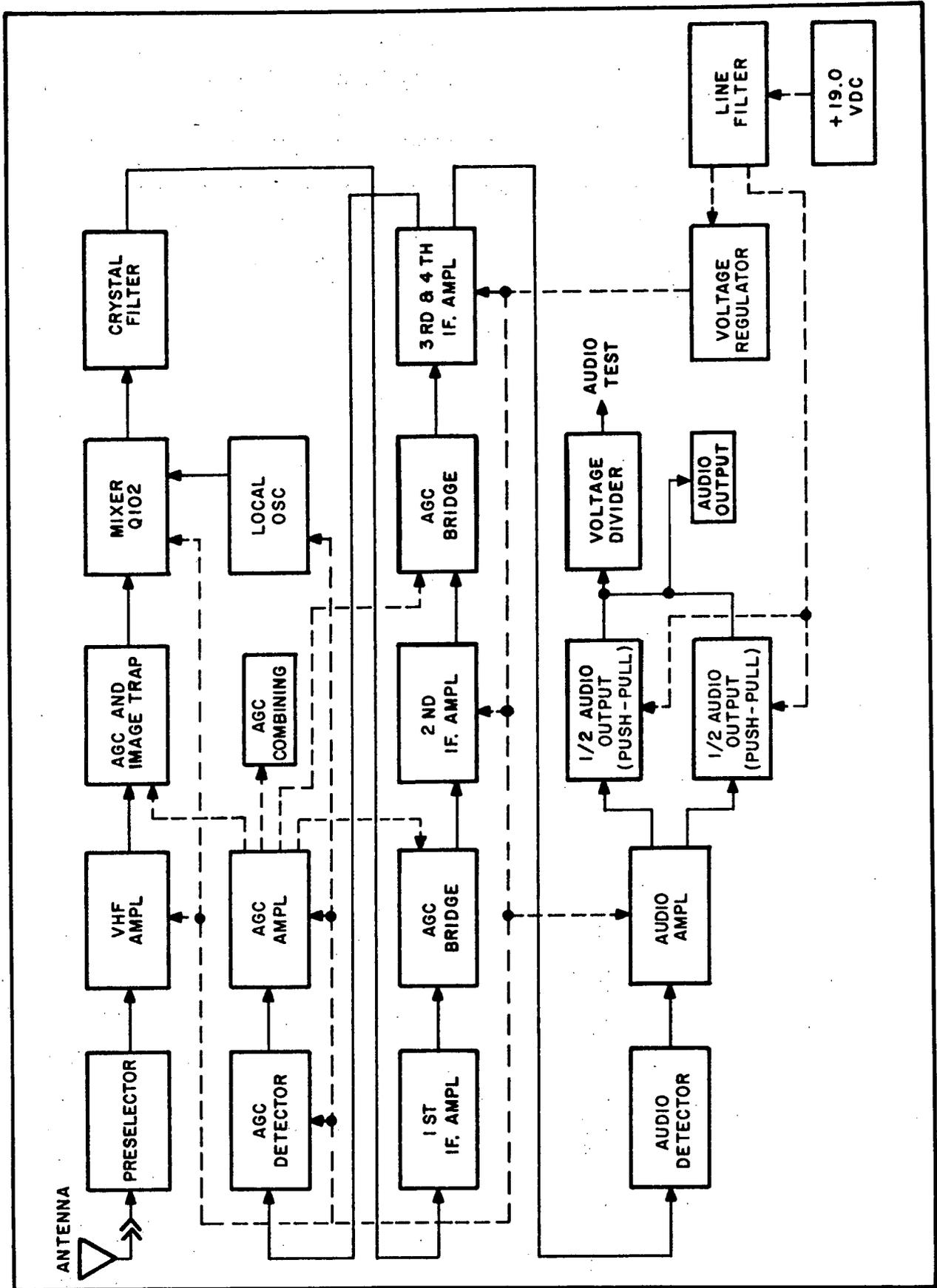
Fig. 10-3 VHF Command Receiver Block Diagram

Table 10-2

RECEIVER ELECTRICAL CHARACTERISTICS

Characteristic	Specification
Type	Superheterodyne, single conversion fixed tuned
Modulation received	Amplitude (am.)
Frequency	149.52 MHz
Minimum Sensitivity	1.5 µv
Audio response within 1.5 db	5000 to 9000 Hz
Dynamic Range	1.5 µv to 1 mv
Bandwidth:	
At 6 db points	35 to 40 kHz
At 60 db points	100 kHz maximum
Receiver local oscillator stability	±2 kHz
Undesired response rejection:	
Image frequency	80 db
Other spurious signals.	60 db
Power Requirements:	
Input voltage	16 to 22 vdc
Standby mode	250 mw maximum
Interrogate mode	330 mw maximum

isolates the OSO telemetry transmitters from the command receivers and permits a common antenna system to be used for both transmitting and receiving. The hybrid circulator provides 40 db isolation between the command receivers to permit independent operation in case one of the receivers fails.

Each receiver has a single conversion 10.2 MHz IF stage with a bandwidth of 36 kHz. The audio frequency response is ±1 db from 1.5 to 7.5 kHz. Each receiver has an independent Automatic Gain Control (AGC) circuit that uses varicap diodes in the interstage coupling circuits. This AGC loop holds the audio output power between 50 and 90 milliwatts into a 500-ohm load for input voltage levels of 2 microvolts to 2 millivolts (75 percent modulated).

VHF selectivity is obtained by using a triple-tuned preselector followed by a RF amplifier stage and a single-tuned coupling transformer. An image trap is designed into the coupling transformer. Image frequencies are attenuated 50 db by the preselector and 30 db by the image trap, resulting in an over-all image rejection of 80 db.

A high degree of rejection to spurious signals is obtained by use of a crystal-controlled local oscillator and a crystal filter ahead of the IF amplifier stage.

- *Receiver Sensitivity Change*

    In an effort to decrease the sensitivity of the command system to any given RF environment, the sensitivity of the receivers was reduced. The measured command threshold using these receivers was on the order of -103 dbm. This is an approximate 10 db increase in system threshold when compared to the -110 to -113 dbm measured on previous command receivers.

10.4   COMMAND DECODERS

A block diagram of the decoder is shown in Fig. 10-4. The incoming serial train of tone bursts from the command receivers are fed to the command decoders. The PWM tone bursts are then passed through a bandpass filter and on to a demodulator where

Fig. 10-4 Command Decoder Functional Diagram

C.7

the envelope of the tone bursts is detected and shaped. The width of the pulses from the demodulator correspond to the width of the tone pulses, i.e., a "one" or a "zero." The three distinct pulses are then separated, and the "ones" are shifted into the "one's" register and "zeros" are shifted into the "zero's" register. A sync pulse is then used to clear the registers and to apply power to the output relay drivers. However, before power is applied to the relay drivers, the proper address word must have been sensed during the preceding 550 milliseconds.

As the registers are being cleared by the sync pulse, the command word is decoded by the sense amplifiers. The sense amplifiers then output a low-level pulse to the appropriate relay driver. The relay driver amplifies this pulse and supplies a 15-volt, 40-millisecond pulse to the corresponding relay coil.

An address word is treated in the same manner as a command word. However, instead of the decoder outputting a control pulse to a relay, a 550-millisecond gate pulse is generated within the decoder. During the gate period, the decoder will accept and decode the command words. Each decoder is prewired to generate the 550-millisecond gate pulse for only one unique address code.

The interfacing circuits between the decoder outputs and space-craft subsystems are shown in Fig. 10-5.

10.5   SYSTEM PERFORMANCE REQUIREMENTS

The performance requirements of the command system are to provide reliable command control of OSO from earth-located stations. The OSO command system design is based on a maximum

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range of 1750 kilometers and within the constraints of the existing ground equipment. The ground station equipment consists of a 200 watt AM transmitter, a right-circularly polarized transmitting antenna having a gain of +12.5 db, and a command encoder. An RF transmission line loss of three db is assumed for the ground equipment.

The OSO command transmission gain margin calculation is presented in Table 10-3. The characteristics of the OSO command link, which determine the performance of the command system, are given in the table.

Table 10-3

COMMAND LINK GAIN MARGINS

Transmitting power (200 watts) (dbm)	+53
Transmitting antenna gain (db)	+12.5
Cable loss (ground) (db)	-3
Path loss (1750 kilometers) (db)	-138.3
Receiving antenna gain (db)	-18
Miscellaneous losses (OSO) (db)	-4
Command receiver sensitivity (dbm)	-103
Margin of gain (db)	+5.2

$$\begin{aligned} \text{Margin of Gain} &= \text{Gains} - \text{Losses} \\ &= (53 + 12.5 + 103) - (3 + 138.3 + 18 + 4) \\ &= 166.5 - 163.3 = +5.2 \text{ db} \end{aligned}$$

The system gain margin is somewhat misleading since the worst case receiving antenna gain was used. As shown in the receiving antenna pattern of Section 9, it can be seen that the antenna gain of -18 db or less occurs only for brief periods of time as the OSO spins. Normal operating procedure at the ground station is to

transmit each command at least twice. This procedure assures
that commanding of the OSO will be executed reliably.

Section 11
STRUCTURES, THERMAL CONTROL, PNEUMATICS, AND HYDRAULICS

11.1   SPACECRAFT STRUCTURES

The structural design of the OSO-6 observatory is basically
the same as that of earlier OSO's. For completeness, however,
a brief description of the structures is repeated here.

The OSO spacecraft consists of two main structural sections:
the wheel section and the sail section. The main structural
features of these sections are shown in Fig. 1-14. In flight,
the wheel section rotates about the spin axis; ideally, the
azimuth shaft. During the daylight portion of the orbit, the
azimuth servo stops sail rotation, pointing the solar-cell
array and the pointed instruments at the sun (in azimuth).
The elevation servo aligns the instruments with the sun line
in elevation.

11.1.1   The Wheel Structure

The wheel structure provides the interface (attach fitting)
for mounting the spacecraft to the two-stage Delta launch
vehicle. The attach fitting is located at the base of the
central hub assembly. The wheel, so named because of its
similarity to a wheel, is a nine-sided polygon varying in depth
from about 10 inches at its rim to about 12 inches at the hub.
Three places, 120 degrees apart, are provided at the rim for
attachment of a lifting assembly used in ground handling.

The spokes of the wheel are radial ribs which separate the
wheel into nine compartments. Structural pie-shaped panels,
or decks, form the link between the ribs and provide the primary
equipment support in the compartments. The compartments house

equipment and scientific payload instruments, and are closed by
the rim panels, and top and bottom non-structural aluminum honey-
comb thermal shields. Three arms are attached to the rim of the
wheel. These are folded along the spin axis for launch and
extended radially during the launch sequence. They are arranged
symmetrically about the wheel. Each arm supports a spherical pres-
sure vessel containing nitrogen gas for spin-rate control.

The inside of the hub provides the bearing supports for the
azimuth shaft and the sail structure. It also provides space
for a spherical pressure vessel containing the pitch-control gas
supply. The compartment side walls and decks have a hydroformed
rib pattern to control flexing.

All primary structural elements except the pressure vessels are
aluminum alloy. Sheet-metal elements are 6061-T6 and 2024-T4
aluminum. All castings are A356-T6 aluminum alloy. The azimuth
shaft is 7075-T6 aluminum alloy.

11.1.2   The Sail Structure

The sail is a fan-shaped structure that supports the solar-cell
array, the pointed instruments, and other equipment. The sail
is mounted to an azimuth shaft which provides the mounting inter-
face between the sail and wheel structures. The sail structure
is not directly attached to the booster or to any pre-flight
lifting equipment.

The pitch-gas vessel is mounted to the azimuth shaft, and rotates
with the sail. Two azimuth ball bearings (type CEVM 440C stain-
less steel) provide the supports between sail and wheel. A
torque motor near the thrust bearing provides control of the
relative rotations between the sail and wheel. Thus the sail
may be fixed to the solar direction while the wheel rotates.

Two castings support the pointed instruments in the sail structure. The instruments mount to the elevation casting, which is trunnion-mounted to the azimuth casting. The azimuth casting is mounted to the azimuth shaft. The trunnion axis is called the spacecraft elevation axis. A torque motor mounted to the azimuth casting provides controlled motion of the elevation casting about the elevation axis of ± 5 degrees.

The remainder of the sail structure is an open framework which mounts to the azimuth casting. Electrical connection between the sail and wheel is provided by slip rings at the base of the azimuth shaft assembly (see Fig. 1-16).

11.1.3   Structural Analysis

Prior to the launch of the OSO-6, a detailed analysis of the spacecraft structures was performed. The purpose of this analysis was to verify that the structure could withstand the loads imposed by ground handling and flight.

Method: The method used for determining internal structural loads for the sail and wheel structures is called the "Stiffness Method" of structural analysis. With this method, the structure is idealized into realistic structural elements; such as beams, bars, and panels. A sufficient number of degree-of-freedom vectors are then established on the total structure so that a realistic stiffness, or spring matrix, may be determined. External loadings are then applied along these degree-of-freedom vectors.

The flight loadings used in the analysis resulted from a study of a two-mass and a three-mass model of the OSO observatory, separately coupled analytically to a flexible Delta N two-stage booster. For X-axis loading, the observatory was idealized into three masses, namely the sail mass, the pointed instrument

assembly mass, and the wheel mass. For X-axis loads, the pointed instruments are free to pitch relative to the basic sail, thus accounting for treating the pointed instrument assembly as a separate mass. For Y-axis loading, the sail and pointed instruments are essentially rigidly attached and, therefore, were lumped into one mass.

First, the primary structure of the OSO-6 was idealized into many beam, bar, and plate elements to represent a realistic model of the observatory. The mass of the observatory was then distributed at discrete points on the idealized model so that the actual mass and center-of-gravity for the total observatory were unchanged. Load factors resulting from flight conditions and ground handling conditions were then applied to the idealized masses of the observatory so that the product of mass times acceleration resulted in applied loadings on the structure.

Finally, the internal loads for the most critical primary structure elements for both the flight loading conditions and the ground handling conditions were compared to internal loads for the same critical primary structure in the OSO-5 observatory. From this comparison, it was found that the loads on the most critical structure in the OSO-6 observatory were equivalent to or less than the loads for the same elements of primary structure that were most critical on the OSO-5 observatory. Since the internal loads on the critical structural elements of OSO-6 were found equivalent to or less than those of OSO-5, the structural analysis of OSO-6 was then made by a simple comparison of the loads that acted upon the primary structure of the two obser-vatories.

Design Loading Conditions: The design loading conditions for the OSO-6 observatory consisted of five flight conditions and three

ground handling conditions. The five flight conditions were selected from analytically derived load factors and modal forces. From the coupled configuration, load factors were determined at the observatory mass points for the first four natural vibration modes. Steady-state load factors of the observatory-booster combination were also supplied for the following phases of flight.

- Liftoff
- Transonic
- POGO
- MECO

Selection of the analysis design conditions from the predicted flight conditions resulted in choosing design conditions that satisfied the following constraints:

(1) Flight conditions exhibiting maximum sail load factors at sail center-of-gravity.

(2) Flight conditions that produced maximum moments on the azimuth shaft.

(3) Flight conditions that resulted in maximum combination of thrust and lateral load at wheel center-of-gravity.

(4) Flight conditions that produced maximum loading on the observatory during acceptance vibration testing.

(5) Flight conditions that caused maximum loads at booster/observatory interface.

Items 1 and 3 involved no more than a search for maximums in a table of flight loads.

Items 2 and 4 involved a calculation of the moments induced into the azimuth shaft due to the sail load factors and a calculation of the moments at the booster/observatory interface.

For Item 5, the equivalent axial loads at the booster interface were calculated for all flight conditions and the maximums were then chosen as analysis design conditions.

Ground handling of the observatory in this analysis was considered as the phase in which the observatory is suspended at three points for shipping purposes. The observatory is expected to withstand load factors of 3.0g in any direction in combination with 1.0g in a mutually perpendicular direction during this phase.

Stress Analysis: A complete detail stress analysis of the OSO-6 observatory primary structure was not performed. Instead, a comparison of the loads for each critical structural element within the OSO-6 observatory was made to the loads in the identical structural element within the OSO-5 observatory. It was found that the internal critical structural loads for the OSO-6 observatory are less than or equivalent to the same internal critical load within the OSO-5 observatory. Therefore, it was concluded that the stresses produced within the OSO-6 critical structural elements were equivalent to or less than those within the OSO-5 observatory.

Summary of Structural Analysis: The detailed structural analysis of the OSO-6 observatory was performed using the OSO-5 structural idealization with OSO-6 mass properties and applying predicted external loads; however, a detailed stress analysis of the OSO-6 structure was not performed because the internal loads upon the

primary structural elements were found to be less than or equivalent to the loads for the same structural elements in the OSO-5 observatory. This being the case the results of the analysis indicated that the OSO-6 observatory was capable of withstanding all flight and ground handling loads imposed upon it.

It was found, in all cases, that the maximum loads acting upon the critical primary structural elements within the OSO-6 observatory were less than the maximum loads acting upon the same primary structural elements within the OSO-5 observatory. Thus, is was confirmed that the OSO-6 observatory possessed sufficient structural integrity to withstand the flight and ground handling environment imposed upon it.

11.1.4   Acceptance Vibration Test Notch Calculations

The acceptance vibration test levels of the OSO-6 observatory were limited in the lateral X-axis and the lateral Y-axis so that the stresses in the most critical primary structural elements during these tests would not exceed the stress levels these structural elements would experience during the actual launch phase.

11.2   THERMAL CONTROL

11.2.1   Why Thermal Control is Needed

For OSO to perform properly, its temperature must be kept within specified limits in those sections of the spacecraft housing temperature-sensitive equipment. Two examples of such equipment are: (1) the solar cells, whose output decreases with rising temperature, and (2) the main storage batteries, which perform most efficiently between -10 and 35°C. Generally, most OSO equipment is designed for the -10 to 35°C temperature range.

Without such temperature restrictions, OSO could be designed as a perfect heat absorber and emitter ("blackbody"). It would then reach an equilibrium temperature (probably very hot) where the amount of solar energy absorbed would equal the amount of infrared energy emitted. Because of the restrictions, however, OSO must have thermal properties somewhere between a perfect "blackbody" and a perfect reflector (which would gain no heat at all). This is done by selecting the proper absorptivities and emittances of the component surfaces where:

absorption (α) = the ratio of the energy absorbed by an object to that absorbed by a "blackbody".

emittance (ε) = the ratio of the energy emitted by an object to that emitted by a "blackbody".

11.2.2 Surface Finishes

Special paints and surface finishes and carefully designed heat-conduction paths within the structure are used to provide OSO's passive thermal control.

The surface finishes are selected on the basis of their thermal characteristics. Those having a low α/ε ratio are used where it is desired to reduce temperature, and those with a high α/ε ratio used to increase temperature. The distribution of the most-used finishes is shown in Fig. 11-1, along with the α/ε ratio of each.

Except for the analog sub-subcommutator, the surfaces of OSO-6's sail components are essentially the same as OSO-5's. The surface finishes on the wheel are identical to the OSO-3 finishes.

(ALL)
① J-BOX

(ALL)
① ASC

④ ASSC ①

(ALL)
⑥ DECODERS

① ③
SERVO & POWER
AMP ASSEMBLIES

⑤

① ③

ALL
⑤ WGG EXCEPT ①

SANDBLASTED TITANIUM
($\alpha/\epsilon = 0.77/0.40$)

③

② TOP COVER SLIDES
⑤ BOTTOM COVER SLIDES

⑤

④

④

③

④

④

②

⑤

⑤

②

① 63W WHITE PAINT
($\alpha/\epsilon = 0.3/0.83$)
② 80U ALUMINUM PAINT
($\alpha/\epsilon = 0.28/0.35$)
③ POLISHED ALUMINUM
($\alpha/\epsilon = 0.25/0.05$)
④ CHROMICOAT ALUMINUM
($\alpha/\epsilon = 0.48/0.10$)
⑤ BARE ALUMINUM
($\alpha/\epsilon = 0.35/0.10$)
⑥ GOLD PLATE
($\alpha/\epsilon = 0.3/0.03$)
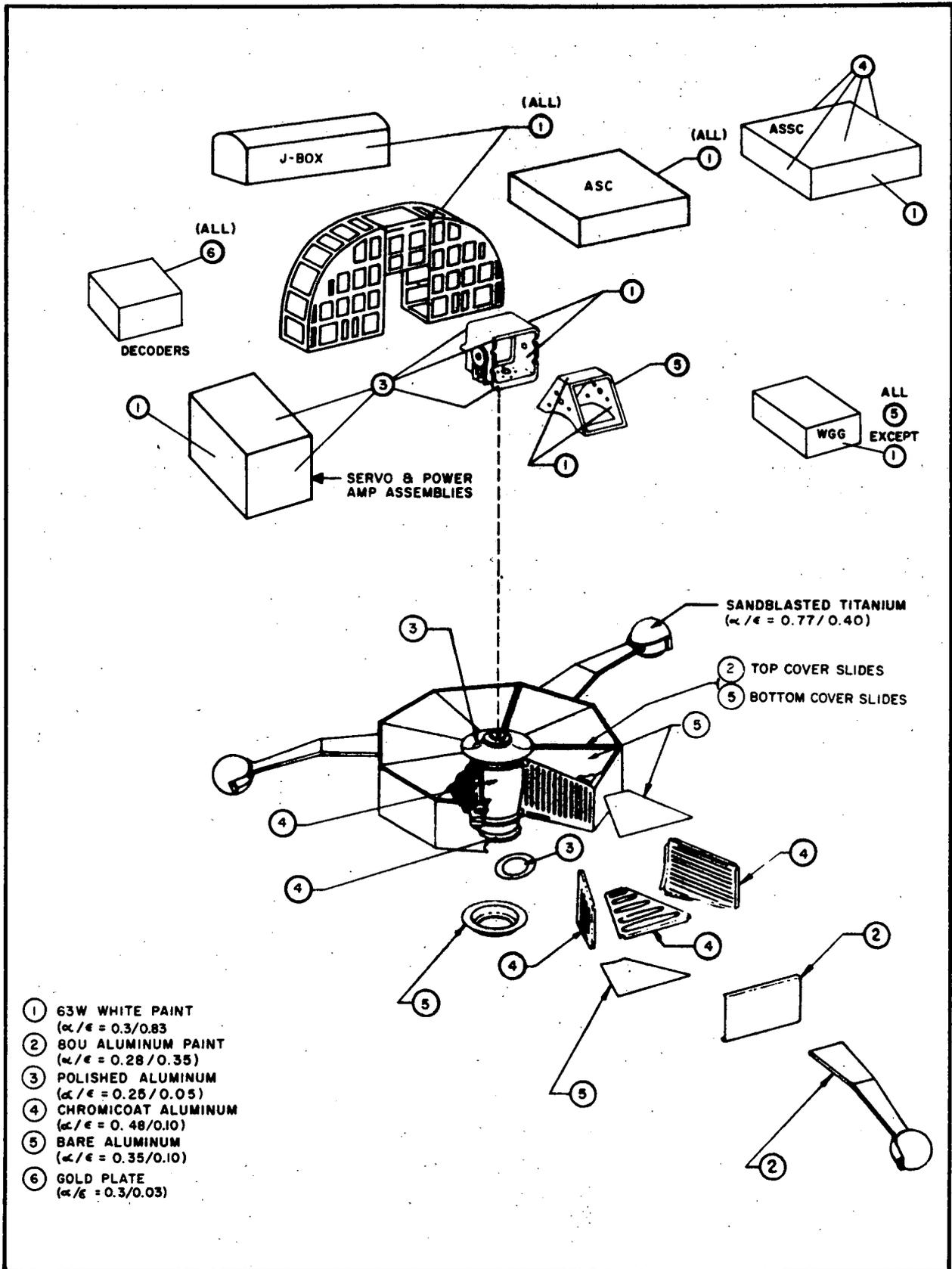
Fig. 11-1   OSO-6 Surface Finishes

Two specially-formulated paints were used for passive thermal control: 63W satellite white and 80U satellite aluminum. The 63W paint is pigmented with titanium dioxide and is used to decrease temperature, since it is an excellent infrared emitter and a poor solar absorber. It is used in the sail, where there is a tendency for components to overheat. The 80U paint is pigmented with a "leafing" aluminum flake and is used in the wheel section to increase the equilibrium temperature. In addition, gold plating, polished aluminum, and bare aluminum are used to provide other α/ϵ ratios where required.

11.2.3   Thermal Analysis

In support of the OSO program, a mathematical system was developed for thermal analysis of the OSO spacecraft. This system was developed for the OSO-3 and is used for predicting the operating temperatures of various components and surfaces and make changes when necessary.

<u>Analysis Method</u>: Because a closed form solution for a complex structure treated as a continuous system is impractical--if not impossible--a mathematical model of OSO, composed of thermally-coupled discrete elements, was constructed and used for the thermal analysis. The elements (or nodes) were assumed to have uniform temperature discontinuously different from adjacent elements.

The original mathematical model 138 nodes, each of which represented a component, part of a component, or group of components. The nodes are designated on the basis of the apparent validity of an assumption of uniform temperature. The heat balance for each node involves the heat capacity of the node, the thermal-radiation interchange between the node and all others, the conduction between the node and all others, the external heat fluxes impinging upon that node, the electrical dissipation within the node, and the heat loss by radiation.

OSO-6 Thermal Analysis: The OSO-6 spacecraft differs from
earlier OSO's due to the installation of a word-gate generator
and a second decoder on the sail egg-crate structure. We were
therefore requested to perform a thermal analysis of the OSO-6
sail to predict the operating temperatures of these components.
This analysis is documented in BBRC Document F68-11, OSO-G
Sail Thermal Analysis Final Report, dated 27 December 1968.

Although a thermal analysis previously had been performed on the
OSO spacecraft, including the egg-crate structure, we decided
that satisfactory results would not be produced by attempting to
add the two electronic boxes to the OSO mathematical thermal
model. The main reasons for this decision were:

(1) The additional components required that the
geometrical model of the egg-crate portion of the
OSO sail be taken through the entire computing
procedure to obtain radiation exchange factors.

(2) The egg-crate structure of the OSO-5 sail was
crudely modeled because computing programs were
not available to solve the heat balance equation
for the complex structure. Since then the thermal
analysis computer programs have been improved
in accuracy and in capacity to handle larger
numbers of components simultaneously.

(3) The junction between the OSO sail substrate and
the egg-crate structure was found to be
discontinuous. Thus, only a small part of the
possible thermal conduction between the substrate
and the egg-crate actually took place. Therefore,
to predict realistic temperatures, the substrate-
to-egg-crate conductances had to be modified to
more closely represent the actual conditions.

From the above considerations, we decided that the most practical procedure would be to construct a new mathematical thermal model of the OSO sail egg-crate structure and the components mounted on the aft portion of the sail. This new model would then be used with the other previously modeled sections of the spacecraft to produce a new set of predicted temperatures.

The mathematical thermal model used in OSO-6 analysis is the same as that used for the thermal analysis of earlier OSO's, with the following exceptions:

(1) The thermal model of the egg-crate structure of the sail is more detailed and thus more closely represents the actual structure.

(2) Consideration of the word-gate generator and a decoder which are mounted on the egg-crate structure.

(3) White paint and chromacoat surface finish emissivities and absorptivities were assigned new values in accordance with recent data obtained from tests.

(4) The junction between the sail substrate and the egg-crate structure was assumed to be in tight physical contact only in the regions of the attachment screws. This area of contact was estimated to be 10 percent of the possible contact area.

(5) The aft sections of both pointing instruments were assigned gold plated surface finishes in place of one gold and one bare beryllium instrument as used in previous analyses. This was done only to improve

the reflecting view factors and the reflecting
external fluxes on the aft section of the sail.

(6)   The surface finish of the sub-subcommutator was
changed from totally white to white on the side
next to the sail substrate and light chromacoat
on the other five sides. This was done to raise
the components' operating temperature.

(7)   The nutation damper base was remodeled by replacing
the six-sided box representation with a three-legged
pedestal shape. This was done to improve the
radiation interchange calculations.

(8)   The total number of nodes used in the OSO-6
mathematical thermal model is 206 as compared
with 138 nodes used in previous models.

Summary and Conclusions: A mathematical thermal model consist-
ing of 206 nodes now exists for the OSO spacecraft. The complete
system of nodes has not been investigated using the surface
finishes of a single spacecraft. The present analysis is for
OSO-3 surface finishes on the wheel and pointing instruments
and OSO-6 surface finishes on the sail.

Transient temperatures were computed for the word-gate generator
and decoder, which are new additions to the OSO sail. Improved
computing techniques and thermal modeling of the sail structure
and components have resulted in predicted temperatures that
compare more closely with the measured temperature data from
OSO-2 and OSO-3 than did the previous thermal analysis. This
is largely because the egg-crate structure has been more
closely modeled and because the computation methods have been
greatly improved. The improved modeling includes the use of

many more nodes to represent the complex egg-crate structure
and the application of the more accurate values of paint
emissivity and absorptivity recently measured by the
materials and processes laboratory.

11.2.4   Temperature Monitors

The temperature of the spacecraft is monitored in several
places by sensors cemented to components, experiments, and
the structure. The sensors use the thermal properties of
a transistor to develop a 0 to 5 volt signal input to the
analog subcommutator. The voltage is representative of
the local temperature.

The sensors in the sail are powered by a single 15-volt regula-
tor located in the servo amplifier assembly. Those in the
wheel are powered by a single 15-volt regulated supply located
in the spin-control assembly.

11.3   HYDRAULIC SYSTEMS

The OSO uses only one hydraulic system: the arm damper system.
The arm damper subsystem controls the rate at which the arms
deploy during the launch phase. The system consists of three
hydraulic cylinders with piston rods connected to each arm
(Fig. 11-2). Denatured alcohol is used as the hydraulic fluid
and piston travel rate is controlled by forcing this fluid
through three 0.020 inch orifices, one in each cylinder.

The three hydraulic cylinders are interconnected by plumbing
to ensure that the arms deploy in unison.
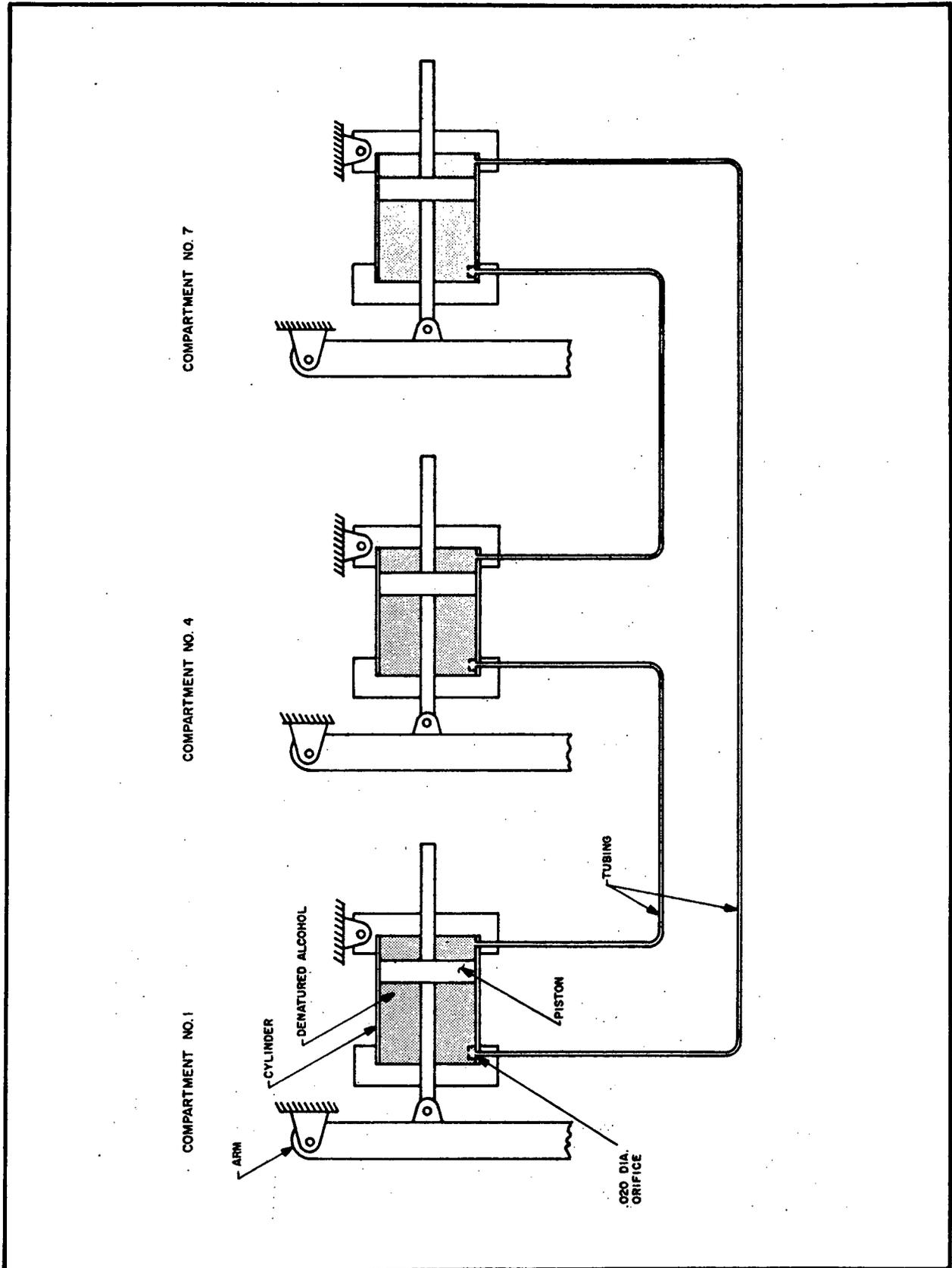
Fig. 11-2 Arm Damper System

11.4   PNEUMATIC SYSTEMS

OSO uses two pneumatic systems: one for pitch control and one for spin-rate control (see Figs. 1-19 and 11-3). Each system produces the required torques by exhausting nitrogen gas through small nozzles (thrusters).

11.4.1   Spin Pneumatics

The spin pneumatic system provides the mechanism through which the spin-control circuits, (described in Section 7) apply accelerating or retarding torques to the spacecraft wheel.

The gas is stored in three titanium bottles at the ends of the spacecraft arms. The initial storage pressure is about 3000 psi. The gas flows from the storage bottles to a manifold block, join-ing the three bottles into a single gas supply and equalizing the pressures in the bottles. At this point, a pressure trans-ducer senses the pressure and produces a voltage proportional to the pressure. The transducer output is sampled once per wheel ASC subframe and is telemetered to receiving ground stations.

From the manifold block, the gas flows through a filter to a pressure regulator, which reduces from the storage pressure to about 100 psi (static). Fluid friction reduces this pressure to provide a gas-flowing pressure of about 60 psig at the spin nozzles. From the regulator, the gas flows to the spinup and the spindown solenoid valves, which are actuated by the spin control circuits. The spinup valve feeds three nozzles, and the spindown valve feeds three nozzles at the ends of the three arms.

Each nozzle produces a thrust of about 0.10 lb for a total of 0.3 lb for either the three spinup or the three spindown nozzles. This thrust acts through a lever arm of 3.97 ft to produce about 1.19 ft-lb of torque. This torque changes the
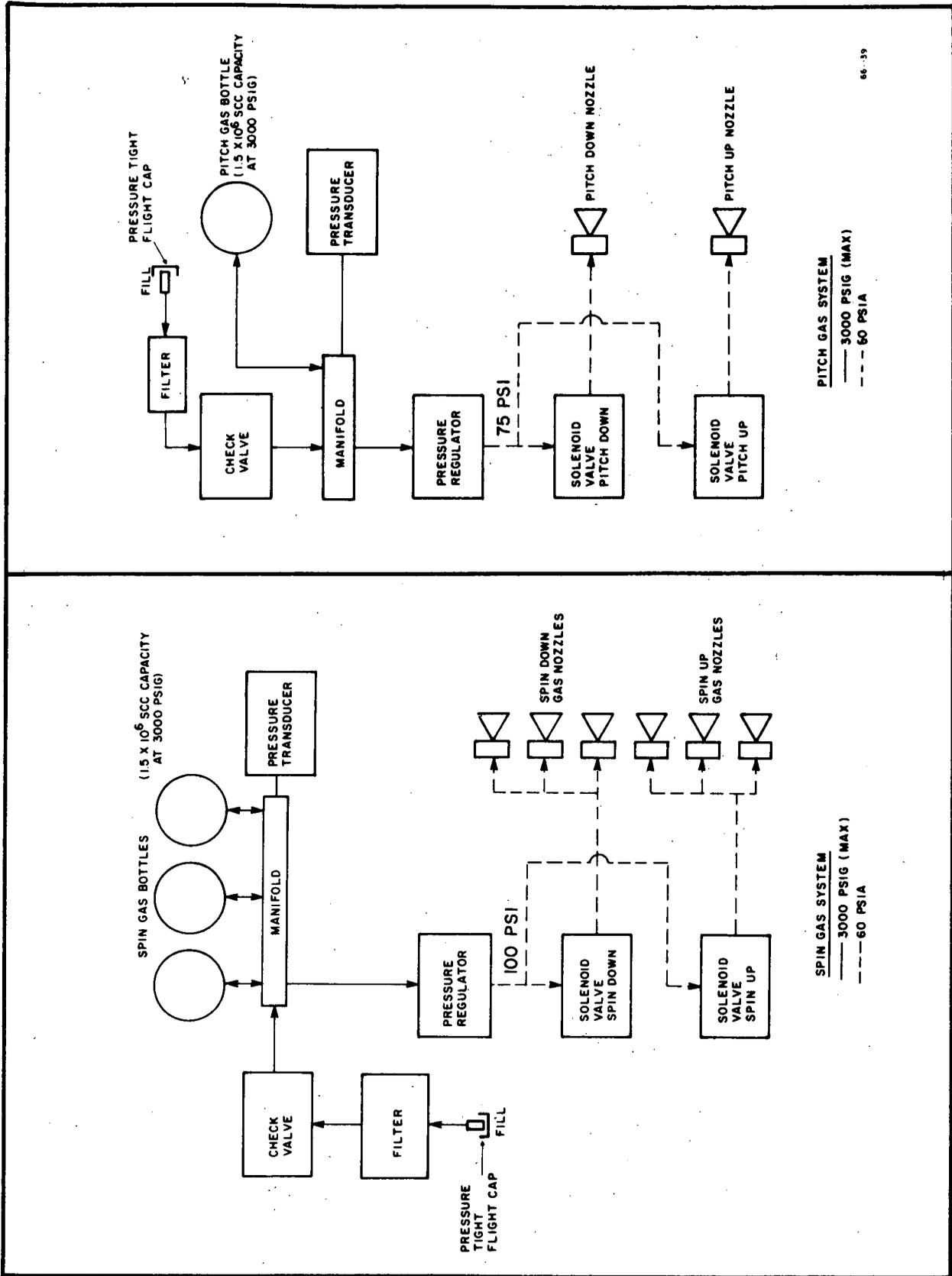
Fig. 11-3 Gas System Block Diagram

wheel spin rate at the rate of about 0.0076 rps every second. The nozzle thrust efficiency in vacuum is about 94 percent. Effective specific impulse is about 66 lb/lb/sec., and the gas consumption rate is about 2,150 std cc/sec. At 3,000 psi storage pressure, the gas stored is equivalent to about 1.5×10^6 standard cc which provides a spin correction capability of about 5.5 rps.

11.4.2   Pitch Pneumatics

The pitch pneumatic system, operated by the pitch control circuits (described in Section 7) provides pitch attitude control of the entire observatory by applying correction torques of either sense about the roll axis. The thrusters that apply these torques are located on either side of the sail and are activated only when the sail is stationary (pointed). The observatory precesses about the pitch axis, thereby providing pitch control, because the torques act upon the stationary sail and because of the gyroscopic properties of the spinning wheel (these properties are discussed in Section 6).

Nitrogen gas, at an initial storage pressure of about 3,000 psi, is stored in a single titanium bottle housed within the hub assembly. A pressure transducer provides information to receiving ground stations, through the spacecraft telemetry system, about the pitch-gas storage pressure. A pressure regulator provides about 75 psig in its static condition, and about 60 psig with gas flowing.

The gas is controlled by the pitch-up and pitch-down solenoid valves. These are actuated by the pitch-control circuits and supply gas to the appropriate nozzles. The pitch-up nozzle is on the left side of the sail (as viewed from the rear) and is aimed upward. The pitch-down nozzle is on the right side, likewise aimed upward.

Each nozzle produces about 0.1 lb. of thrust which acts on a lever arm of 1.93 ft. to produce a torque of 0.193 ft-lb. This torque produces a correction rate of 0.0645 deg.-rps/sec. in pitch angle. The nozzle thrust efficiency in vacuum is about 94 percent. Effective specific impulse is about 66 lb/lb/sec., and the gas consumption rate is about 716 std cc/sec. At 3,000 psi, the gas stored is equivalent to about 1.5×10^6 standard cc which provides about 169° rps of pitch correction.

Appendix A

ABBREVIATIONS USED IN THIS REPORT

(This list excludes many abbreviations which
may be found in any standard dictionary or
which are extremely common in technical usage.
Also excluded are some of those which are
defined in the text whenever they are used.)

Table A-1

ABBREVIATIONS USED IN THIS REPORT

A	ampere	cont	continuous
Å	angstrom	CPIF	cost plus incentive fee
AC/ac	alternating current	cps	cycles per second (Hertz)
A/D	analog-to-digital		
AGC	automatic gain control	c/w	conditional with
alloc	allocated		
amp	ampere, amplifier	D	DSF or DSM (short form)
ANG	Air National Guard	db	decibels
APL	automatic-pitch limiter	dbm	decibels (ref 1 mw)
		DC/dc	direct current
ASC	analog subcommutator	dcdr	decoder
ASL	automatic-spin limiter	deg	degrees
ASSC	analog sub-sub-commutator	disab	disab, disabled, disabling
ass'y	assembly	DL	dummy load
auto	automatic	DME	digital multiplex-encoder
az/azim	azimuth		
		D-N, D/N	day-night
		doc	document
batt	battery	DS	design specification (BBRC doc)
BBRC	Ball Brothers Research Corporation		
		DSF	digital subframe
Bev	billion electron volts	DSM	digital submultiplexer
bps	bits per second	dwg	drawing
BU	backup		
byp	bypass		
		EDT	Eastern Daylight Savings Time
C	Celsius (Centigrade) temperature scale	el/elev	elevation
calib	calibration	EMI	electro-magnetic interference
cc	cubic centimeter		
CG	center of gravity	enab	enable, enabled, enabling
ckt	circuit	EST	Eastern Standard Time
cm	centimeter	ETR	Eastern Test Range
cmd/cmmd	command	ev	electron volt
comp	comprehensive test	exct	execute
compt	compartment	expmt	experiment

Table A-1 (continuation)

ext	external	k	prefix kilo (one thousand)
FC	frame count/counter	k	kilocycles per second
FET	field-effect transistor	KE	kinetic energy
FF	flip flop	KSC	Kennedy Space Center
fig.	figure		
FM	frequency modulation	LA	Lyman-alpha
freq	frequency	LASL	Los Alamos Scientific Laboratory
ft	foot	lch	launch
		LSB	least-significant bit
g	acceleration due to gravity	LSBU	launch-sequence backup
gen	generator	LST	launch-sequence timer
GFE	government-furnished equipment	LV	low voltage
gnd	ground	M	prefix mega (one million)
GSFC	Goddard Space Flight Center	m	meter, prefix milli one one-thousandth)
HCO	Harvard College Observatory	ma	milliamperes
		mag	magnetometer
hr	hour	man	manual
HV	high voltage	max	maximum
hz	hertz (cps)	MDAC	McDonnell-Douglas Astronautics Company
I	electrical current in amperes	MDC	Mission Director's Center
I and T	Integration and Test	meas	measured
ID	sync identification	MF	main frame
IF	intermediate frequency	min	minutes, minimum
in.	inch	mod	modulation
int	internal	MOI	moment of inertia
IR	infra-red	mon	monitor
		MS	model specification (BBRC doc)
J-box	junction box	MSB	most-significant bit

A-3

Table A-1 (continuation)

MSFC	Marshall Space Flight Center	psi/psia	pounds per square inch (absolute pressure)
		psig	pounds per square inch (gage pressure)
NASA	National Aeronautics and Space Administration	ptd	pointed
		PWM	pulse-width modulator
neg	negative	pwr	power
NM/N.Mex	New Mexico		
nm/nmi	nautical mile	QP	qualification plan (BBRC doc)
No.	number		
NRL	Naval Research Laboratory		
		rcvr	receiver
nut.	nutation/nutation damper	red't	redundant
		ref	reference
N2	nitrogen gas	rel	release
OATS	Observatory Automatic Test System	RF	radio frequency
		RFI	RF interference
ofst	offset	rms	root mean square
OP	overload protection	RO	readout
orb	orbit	rpm	revolutions per minute
OSO	Orbiting Solar Observatory	rps	revolutions per second
		rst	reset
ov	overvoltage	rstr	raster
PAC	packaged attitude control	S/SASC	Sail Analog Subcom
		SC	sail command
para	paragraph	S/C	spacecraft
pbk	playback	sec	seconds
PCM	pulse-code modulation	sel	select
PERT	Program Evaluation and Review Technique	sep	separation
		seq	sequence
PM	photomultiplier	S/N	serial number
pos	positive, position	SOL	start of line
P-P	peak-to-peak	SOR	start of raster
press.	pressure	SORE	spin orientation and rate electronics
prop	proportional, proposal		

Table A-1 (continuation)

sq	squib	V/volt	volts, voltage
SS	ASSC (short form)	vhf	very high frequency
STADAN	Satellite Tracking and Data Acquisition Network	VSWR	voltage-standing-wave ratio
std	standard	w	watts (power)
subcom	subcommutator	W/WASC	Wheel Analog Subcom
sw	switch	WC	wheel command
sync	synchronization signal	WG	word gate
sys	system	WGG	word gate generator
S3	Decoder 3 (sail)	wh	wheel
S4	Decoder 4 (sail)	W1	Decoder 1 (wheel)
		W2	Decoder 2 (wheel)
temp	temperature		
tlm	telemetry	xmtr	transmitter
TMG	time-mark generator		
TN	technical note (BBRC doc)		
TP	test point or test procedure (BBRC doc)		
TR	tape recorder or technical report (BBRC doc)		
TS	test specification (BBRC doc)		
T-V	thermal vacuum		
TWX	Teletypewriter Exchange Service		
U/Univ	University		
UCL	University College of London		
UNM	University of N. Mex		
UT	universal time		
UV	ultraviolet, under-voltage		
UVS	undervoltage switch		

Appendix B

BIBLIOGRAPHY

Table B-1

BIBLIOGRAPHY

Document	Title
	SPACECRAFT TEST AND HANDLING DOCUMENTATION
TN65-258-A1	OSO Mass Balance Procedures (29 Apr 69)
TN65-286	Operating Procedure for 10-Foot Diameter NASA-4 Thermal-Vacuum Chamber (9 Sep 65)
TN66-31-B1	Procedures for OSO Weight, Center of Gravity, and Moments-of-Inertia Measurements (12 Nov 68)
TN66-49-A2	OSO Demagnetization and Magnetic Balance Procedure (4 Dec 68)
TN67-104-A	OSO Handling Procedures (18 May 68)
TN68-04-A1	OSO-G Operations Plan (13 May 69)
TN68-05-A3	OSO-G Supplementary Test Procedures (10 Jul 69)
TN68-06	OSO-G Upper Structure Test Procedure (15 May 68)
TN68-14-B	OSO-G Comprehensive Test Procedure (8 Jul 69)
TN68-17	OSO-G Spacecraft Thermal-Vacuum Test Procedure (2 Dec 68)
TN68-18-B	OSO-G Test Data Summary (15 Jul 69)
TN68-20-1	OSO-G Observatory Vibration Test Procedure (19 May 69)
TN68-21-B	OSO-G Observatory Thermal-Vacuum Test Procedure (14 Jul 69)
TN68-23-1	OSO-G ETR Handling and Special Checkout Procedures (10 Jun 69)
TN68-24-3	OSO-G ETR Operations Plan (11 Jul 69)
TN68-27	OSO-G Launch Tower/Payload Countdown Procedures (10 Jun 69)
TN68-53	OSO Battery Conditioning Procedure (17 May 68)
TN68-57-A	OSO-G Spacecraft Acceptance Plan (19 Nov 68)
TN68-167-A	OSO-G Failure and Problem Summary (22 Jul 69)
TN69-63	OSO-G Observatory Acceptance Plan (9 May 69)
TR69-43	OSO Shock Test Qualification Report (12 Jun 69)
16704	OSO Pneumatic System Leak Rate Procedure (13 Sep 65)
20366-B	OSO Pneumatic Pressurization Procedure (2 May 69)
21854-C	OSO-G Observatory Painting Procedure (2 May 67)
26493-D	Model Specification -- OSO-G Spacecraft (19 Mar 69)
26494-G	Test Specification -- OSO-G Spacecraft (14 Mar 69)

Table B-1 (continuation)

Document	Title
	ANALYSIS DOCUMENTATION
F66-06	OSO Dynamics Analysis (15 Nov 66)
F67-12	A Mathematical Thermal Model of the OSO Spacecraft (17 Nov 67)
F68-01	Evaluation of Present OSO Antenna Patterns (15 Mar 68)
F68-11	OSO-G Sail Thermal Analysis Final Report (27 Dec 68)
TN65-304	OSO Roll Attitude Study (18 Oct 65)
TN66-01	OSO Dynamic Analysis (3 Feb 66)
TN66-37	The Effect of Various Orbital Conditions on Observatory Shadow Time (1 Mar 66)
TN66-114	OSO Magnetic Control Study -- Theoretical Investigation (28 Apr 66)
TN68-158	Torque Effect Graphs Showing Pitch, Roll, and Spin Rate Drift and How They Can Be Used for OSO Attitude Control (12 Dec 68)
TN68-161	OSO-G Reliability Analysis (21 Jan 69)
TN68-162	Reliability Analysis of OSO-G Offset Point and Scan System (19 Dec 68)
TN68-166	An Analysis of the OSO-G Roll Coil Efficiency (20 Dec 68)
TN69-79	OSO-G Structural Analysis (27 Jun 69)
TN69-81	OSO-G Flight Readiness Review (2 Jul 69)
TN69-111	OSO-G Roll Coil Error Analysis (27 Oct 69)
TR67-74-E	OSO-G Mass Properties (Report) (3 Jan 69)
TR68-55	Optimum Tape Recorder/Transmitter Interface Evaluation (16 Sep 68)
	COMPONENT/SYSTEM TEST DOCUMENTATION (These documents are listed in addition to the specifications, procedures, and reports shown in Table 3-1)
TN68-55-A	OSO-G Component Qualification Status List (24 Nov 69)
TN68-159-A	OSO-G Certification Log Manual (15 Oct 69)
TN69-49	Calibration Procedure for Offset Point Module Test Fixture (26 Mar 69)
TR69-01	Technical Report -- OSO-G Antenna (24 Jan 69)
TR69-03	OSO Command and RF Systems Susceptibility Report (3 Feb 69)

Table B-1 (continuation)

Document	Title
	MISCELLANEOUS DOCUMENTATION
F63-3	OSO-1 Final Report (30 Oct 64)
F65-9	OSO-C Final Report
F66-03	OSO-2 Final Report (15 Sep 66)
F67-14	OSO-3 Final Report (12 Dec 69)
F68-07	OSO-4 Final Report (14 Aug 70)
F70-02	OSO-5 Final Report (6 Apr 71)
TM65-01	OSO Aspect System Manual (15 Jul 65)
TN65-145-D	Manual of Safety Practice--OSO Integration and Test (18 Mar 66)
	BBRC Specifications for the Integration and Test Laminar Flow Clean Rooms (Revision A, 23 Jan 68)
TN65-175	Ephemeris Equations for Predicting Locations of OSO Earth Coordinates (Apr 66)
TN65-205	Magnetotropometer (8 Jul 66)
TN65-310	OSO Technical Documentation Procedures (1 Sep 65)
TN66-48-A	Observatory Demagnetizer (20 Dec 66)
TN66-169	Aspect Data Reduction Program for an OSO Wheel Experiment (16 Dec 66)
TN67-10	Magnetics for OSO Designers (15 Mar 67)
TN68-16-B1	OSO-G Telemetry Data Conversion Tables (16 Jul 69)
TN68-22	OSO-G Post-Launch Data Reduction Procedure
TN68-25-A	OSO-G In-Orbit Operation Procedures (16 Jul 69)
TN68-26	OSO-G Post-Launch Data Requirements (10 Jul 69)
TN68-61	Preferred Parts Handbook (12 Jul 68)
TN68-80-A	OSO-G Characteristics Data and Launch Requirements (21 Jul 69)
TN69-07	Contamination Tolerance for OSO-G and H Experiments (11 Feb 69)
TN69-47	Preferred Parts List for Space Systems (1 Apr 69)
TN69-56	Preparing OSO Test Procedures for Computerization (23 Apr 69)
TN69-66	Soldering Certification and Recertification Procedure (6 Jun 69)
TN69-91-A	OSO-G Contingency Plan (6 Aug 69)

Table B-1 (continuation)

Document	Title
TR69-08	Category and Contamination Ranking Lists for OSO Materials (1 Mar 69)
TR69-64	OSO New Technology Report (28 Oct 69)
23732B	Installation Control Drawing, OSO-G/DELTA Interface (11 Jul 69)

NON-BBRC PUBLICATIONS

DAC SM-48897	Improved DELTA Spacecraft Design Restraints (Jan 66)
NASA SP57	Orbiting Solar Observatory 1 (1965)
NASA TR-985	DELTA-72 Orbiting Solar Observatory-G (OSO-G) Operations Summary (5 Aug 69)
NASA TR-987	DELTA-72 Orbiting Solar Observatory-G (OSO-G) Flash Flight Report
NASA X-513-69-274	NASA-GSFC Operations Plan 10-69 Orbiting Solar Observatory OSO-G (Jul 69)
NPC-200-2	Quality Program Provision for Space System Contractors
NPC-250-1	NASA Reliability Publication

EXPERIMENT DOCUMENTATION (Experiments BBRC documents are listed on Table 3-3)

Appendix C

OSO-6 SYSTEMS AND COMPONENTS
DESCRIPTION

Table C-1

FUNCTIONAL DESCRIPTION OF OSO-6 SPACECRAFT SYSTEMS

System	Functional Description
Pitch control system	Provides pitch control of spacecraft by two means: (1) exhausting nitrogen gas through pitch up and pitch down nozzles and (2) torquing the spacecraft with a pitch coil; pneumatic system has two modes of operation: (1) Automatic pitch control (2) Manual pitch control Automatic mode maintains spin axis normal to solar vector within ±4 deg by action of automatic pitch control electronics. Manual mode provides backup in event of automatic mode failure; manual control is accomplished by ground station commands. The pitch coil is controlled solely by ground station commands.
Roll control system	Provides roll attitude control of the spacecraft by means of two magnetic roll coils mounted on the sail. Control is by ground command only.
Spin control system	Provides spin rate control of wheel by exhausting nitrogen gas through spin and despin nozzles; has two modes of operation: (1) Automatic spin control (2) Manual spin control Automatic mode maintains spin rate between 0.45 and 0.67 rps by action of automatic spin control electronics; automatic spin control circuits are disabled and the manual mode automatically selected if spin rate goes outside the range of 0.39 to 0.70 rps. Manual mode provides backup in event of automatic mode failure; manual spin-up and spin-down control is accomplished by ground station commands.
Pointing control system	Provides initial solar acquisition and accurate pointing of solar oriented scientific experiments by driving sail opposite to wheel spin direction (azimuth pointing) and rotating the experiments in elevation (elevation pointing); has four modes of operation: (1) Normal pointing (experiments point within ±1 arc minute of solar disk center) (2) Large raster scan (experiments scan the solar disk with a centered 64 line raster pattern of 46 by 46 arc minutes) (3) Offset pointing (experiments can be pointed at any of 16384 points within the large raster pattern) (4) Small raster (experiments can be rastered about any of the 16384 offset points with a 16-line raster 7 1/2 arc minutes high and 7 arc minutes wide. Pointing is accomplished automatically every orbit-morning by the coarse and fine pointing control electronics, raster scanning and offset pointing is enacted by ground station command; pointing accuracy is better than ±1 arc minute.
Nutation damping system	Damps nutational movement (coning) of the spacecraft spin axis; consists of a passive nutation damper which absorbs nutational energy and dissipates it in the form of heat.
Launch sequence system	Controls the occurrence of observatory events during launch phase and changes observatory from launch configuration to orbit configuration; consists of four separation switches (that sense OSO separation from booster) which start motor-driven timers that directly or indirectly perform the following functions: (1) Activates squib pin pullers to release arms, nutation damper, and pointed instruments. (2) Applies orbit-power to observatory and enables the ASL and APL at proper time.
Telemetry system	Provides means by which spacecraft and experiment data are sampled and transmitted to ground stations; consists of PCM data handling subsystem and RF subsystem. Performs the following basic operations: (1) Data sampling (2) Data encoding (3) Time multiplexing (4) Data storage (magnetic tape) (5) Transmitting of data
Aspect measuring system	Provides aspect data from which the roll attitude, wheel angular position, and wheel spin rate can be computed; consists of a flux-gate magnetometer and the Spin Orientation and Rate Electronics (SORE box).
Power system	Provides electrical power for spacecraft and experiment consumption by converting sunlight into electrical energy; consists basically of a solar cell array, storage batteries, and a power distribution system.
Command System	Provides means by which the operation mode of the spacecraft and its experiments can be changed by commands sent from ground stations located on earth; consists of spacecraft command receivers, command decoders, and control relays.

Table C-2

FUNCTIONAL DESCRIPTION OF OSO-6 MAJOR SPACECRAFT COMPONENTS

Component	Functional Description
WHEEL STRUCTURE:	Consists basically of nine wedge-shaped compartments arranged around a central hub; houses wheel electronics, wheel experiments, and spin pneumatics; supports sail structure and three extendible arms; provides stable platform for scientific instruments because of gyroscopic properties.
Wheel Mechanical Components (1) Wheel hub assembly (1)[a]	Serves as central connecting hub for wheel equipment compartments; houses the following components:
a. Azimuth shaft (1)	Connects to sail azimuth casting and supports sail structure.
b. Azimuth torque motor (1)	Provides azimuth torque for sail despinning and pointing control; powered by pointing control system.
c. Slip ring assy. (1)	Consists of 22 slip ring contacts that transmit electrical signals and power between wheel and sail.
d. Azimuth bearings (2)	Supports azimuth shaft and provides low friction rotational drag between wheel and sail.
e. Pitch gas bottle (1)	Stores high pressure (3000 psi) nitrogen gas for Pitch Control System.
(2) Equipment compartments (9)	Houses wheel electronics and unoriented experiments; consists of nine wedge-shaped compartments; provides thermal control of electronic equipment by means of thermal shields, passive radiation control, and thermal conduction schemes.
(3) Extendible arms (3)	Extends during launch phase to increase the axial moment-of-inertia of the wheel; includes the following components:
a. Spin gas bottles (3)	Stores high pressure (3000 psi) nitrogen gas for spin control system.
b. Support brackets (3)	Forms part of three RF antennas; stabilizes arms.
(4) Arm damper hydraulics	Reduces arm velocity during deployment to prevent damage to arms and components; consists of three hydraulic cylinders, flow restricting orifices, and interconnecting plumbing.
(5) Spin pneumatics	Provides distribution and control of spin correction gas; actuated by spin control circuits. Consists of the following components:
a. Spin gas bottles (3)	Stores high pressure (3000 psi) nitrogen gas for spin control system.
b. Pressure regulator (1)	Reduces spin gas pressure from 3000 psi initial storage pressure to approximately 100 psi; this pressure is further reduced to approximately 60 psi at the spin nozzles because of fluid friction in the plumbing.
c. Manifold block (1)	Distributes spin gas to system plumbing; maintains pressure balance in spin gas bottles.
d. Spin-up solenoid valve (1)	Controls gas flow to spin-up nozzles; actuated by spin control circuits.
e. Pressure transducer (1)	Provides monitoring of spin gas pressure by telemetry.
f. Spin-down solenoid valve (1)	Controls gas flow to despin nozzles; actuated by spin control circuits.
g. Spin-up nozzles (3)	Provide spin-up thrust.
h. Despin nozzles (3)	Provide spin-down thrust.
i. Check valve (1)	Serves as filling port.

(a) Number in parentheses indicates total number of components.

Table C-2 (continuation)

Component	Functional Description
Wheel Electrical Components	
(1) Spin control box (1)	Contains spin control electronics, undervoltage switch circuits, day turn-on circuits, and two 15 v DC regulators; provides spin rate control (auto and man.), day turn-on, and battery undervoltage protection.
(2) Command receivers (2) (redundant)	Detects and amplifies PWM coded RF command signals from ground stations at 149.52 Mhz, the two receivers have common inputs but the outputs drive separate wheel decoders; output selection for sail decoder is accomplished by ground station command; receivers operate continuously.
(3) Command decoders (2) (redundant)	Accept PWM command tones from command receivers and provide 77 wheel command functions; inputs are provided by separate command receivers; 63 command outputs are redundant; decoder selection is accomplished by unique decoder addresses from ground stations; decoders operate continuously; command functions are executed by the activation of appropriate control relays and circuits.
(4) Launch sequence timers (2) (redundant)	Controls the occurrence of spacecraft operational events during launch phase; consists of redundant, motor-driven, cam operated, rotary switches; activation is by separation switches during separation from booster. Provides the following launch sequence functions:

Legend:

T_L = Liftoff

T_T = Timer start

Nominal Time (sec)		Function
T_L + 721	T_T + 0	OSO separation; OSO timers started; arms released
T_L + 1051	T_T + 330	Turns on orbit power; selects auto-spin mode. (These functions initiate azimuth acquisition, wheel despin, and pitch error correction.)
T_L + 1321	T_T + 600	Uncages nutation damper
T_L + 1521	T_T + 800	Releases elevation frame; enables ASL and APL circuits for spin rate and pitch attitude runaway protection.
T_L + 1971	T_T + 1250	Repeats all of the above functions to insure operation.
T_L + 2051	T_T + 1330	Power removed from timers.

Component	Functional Description
(5) Separation switches (4) (redundant)	Consists of four individual switches in series-parallel arrangement mounted on the bottom of the OSO attach fitting; detects separation of spacecraft from booster and activates launch sequence timers.
(6) Hybrid circulator (1)	Isolates the inputs of the two redundant command receivers to allow independent operation in case of receiver malfunction.
(7) Antennas (3)	Radiate RF telemetry signals and receive RF command signals; consist of three support brackets and three antenna stubs; two dipoles are end-driven and one is parasitic.
(8) Diplexer (1)	Provides transmitting and receiving with a common antenna array.

Table C-2 (continuation)

Component	Functional Description
(9) Transmitters (2) (redundant)	Generates a RF signal modulated by Manchester coded data; uses PM (phase - modulation) techniques in which a digital "1" or "0" is denoted by a 57 deg phase shift of the RF carrier; operates at 136.71 Mhz with an output power of 600 mw; transmitter selection is accomplished by ground commands.
(10) Coax switch (1)	Connects appropriate transmitter to antenna array; operates simultaneously with transmitter select commands.
(11) Spin eye assembly (1)	Produces a "sun" pulse each time the wheel sweeps the sun; triggers the spin control circuits to maintain auto-spin control; provides day/night control of observatory; provides ground station monitoring of spin rate by telemetry.
(12) Tape recorders (2) (redundant)	Continuously records telemetry data on magnetic tape during each orbit (approx. 103 min. capacity); plays back recorded data at 18 times record rate (approx. 5.6 min.) upon ground command; automatically reverts to record mode after playback completion; selection is accomplished by ground commands.
(13) Analog subcommutator (1) (WASC)	Commutates wheel and experiment housekeeping data into a 48 word subcom frame; gating and timing is provided by the digital multiplexer-encoders (DME's).
(14) Digital submultiplexer and frame counter (1) (DSM & FC)	Multiplexes digital spacecraft aspect data, frame count, single-bit status monitors, and experiment data into 48 word frame; gating and timing is provided by the DME's.
(15) Main battery assembly (1)	Stores electrical energy for observatory night consumption and peak demands; consists of 42 ni-cad cells arranged in series-parallel; capable of producing more than 12 amp-hr before going into undervoltage state (16.2 volts).
(16) Digital multiplexers and encoders (2) (redundant) (DME No. 1 and No. 2)	Generates telemetry clock and data readout gates that control the commutating and multiplexing of telemetry data; provides analog to digital encoding of analog data; multiplexes all telemetry data (RZ or NRZ) into Manchester Code.
(17) PCM junction box (1)	Provides interconnections within the PCM data handling subsystem; contains isolation circuits for digital data outputs from various experiments and for the telemetry clock; contains TMG gate circuits which, when energized convert all main frame data to "ones" except WASC and SASC words, and the main frame sync words, (used as backup to frame counter).
(18) Pressure cartridges (squibs)	Provides mechanical uncaging of various wheel components during launch phase; are detonated by the wheel squib batteries in conjunction with the launch sequence timers. Consists of 6 armlock squibs (2 for each arm) which release arms at T_T + 0 sec.
(19) Squib battery packs (2) (redundant)	Provides electrical power for detonating wheel pressure cartridges; each pack consists of 7 ni-cad cells connected in series; capable of supplying 20 amps into 1/4 ohm load for 2 seconds.

Table C-2 (continuation)

Component	Functional Description
(20) Magnetometer and Electronics (1)	Measures the direction of the earth's magnetic field by detecting when the magnetic component is a positive going zero; the magnetometer is mounted on arm No. 4 (corresponds to compartment 4) with its sensitive axis parallel to the wheel plane; power is obtained from the magnetometer electronics and its output is coupled to the SORE box.
(21) SORE Box (1)	Provides a binary count of the clock pulses that occur between a "rising zero crossing" (from magnetometer electronics) and the first sun pulse after the zero field signal (from this count the angle between the roll axis and field vector can be computed); also provides a binary count of the clock pulses that occur between DSM word No. 3 and the sun pulse or DSM word No. 3 and the magnetometer pulse (from this count the wheel spin rate can be computed).
(22) Pitch Coil (1)	Augments the pneumatic pitch control system by interacting with the earth's magnetic field which produces torques that causes OSO to rotate in the pitch plane; consists of a large inductor placed parallel to the wheel plane around the wheel hub; can be commanded on, off, full pwr, 1/2 pwr, positive or negative to satisfy operation requirements.
SAIL STRUCTURE:	Consists basically of a semicircular framework attached to the wheel by the azimuth shaft; the front surface is covered by solar cells behind which are housed the sail electronics, pitch pneumatics, and nutation damper; provides accurate azimuth and elevation pointing of the solar-oriented instruments by action of pointing control system.
Sail mechanical components	
(1) Azimuth casting (1)	Forms the base of the sail structure at which point the sail is connected to the wheel azimuth shaft; supports the elevation casting and "pointed" instruments.
a. Elevation torque motor (1)	Provides elevation torque for pointing of oriented instruments; actuated by pointing control system.
b. Elevation bearings (2)	Supports elevation stub shafts and assures low-friction drag between azimuth casting and elevation casting.
(2) Elevation Casting (1)	Supports pointed instruments; rotates in elevation within the azimuth casting by action of the elevation torque motor and pointing control system.
a. Elevation shaft (2)	Supports elevation casting by means of elevation bearings; consists of two stub shafts.
(3) Nutation damper (1)	Damps nutational motion of observatory by converting kinetic nutation energy into heat energy; consists of a pendulum that moves in a bath of silicone oil.
(4) Pitch pneumatics	Provides distribution and control of pitch precession gas; is actuated by pitch control circuits; consists of the following components:
a. Pitch gas bottle (1)	Stores high-pressure (3000 psi) nitrogen gas for pitch control system. NOTE: This component is mounted around the azimuth shaft inside the wheel hub assembly.

Table C-2 (continuation)

Component	Functional Description
b. Pressure regulator (1)	Reduces pitch gas pressure from 3000 psi initial storage pressure to approximately 72 psi; this pressure is further reduced to approximately 60 psi because of fluid friction in the plumbing.
c. Pressure transducer (1)	Provides monitoring of pitch gas pressure by telemetry.
d. Filter (1)	Prevents foreign material from entering pneumatics during filling operations.
e. Pitch up nozzle (1)	Provides pitch up thrust.
f. Pitch down nozzle (1)	Provides pitch down thrust.
g. Check valve (1)	Serves as filling port.
Sail Electrical Components	
(1) Command decoders (2)	Accept pulse width modulated (PWM) command tones from the command receivers and provides 98 sail command functions; the command functions are executed by the activation of appropriate control relays and circuits.
(2) Servo amplifier box (1)	Amplifies and conditions the azimuth and elevation control signals from the azimuth and elevation control eye assembly before application to the power amplifier; amplifies target eye output and switches from coarse azimuth to fine azimuth control when coarse alignment (±3 deg) has been achieved; provides target eye, az. and elev. preamp., az. and elev. error signal, and servo box 15 v DC regulator monitoring by telemetry.
(3) Power amplifier box (1)	Amplifies and conditions the azimuth and elevation control signals from the servo amplifier before application to the az. and elev. power switches and torque motors; generates az. and elev. raster scan and offset point signals; generates auto. and manual pitch correction signals from pitch control error signals and ground-station commands, respectively; provides telemetry monitoring of the az. and elev. raster signals, offset point generator status, az. and elev. torque motor currents.
(4) Sail Junction Box (1)	Provides distribution of electrical signals in the sail; houses on-off relays for the pointed instruments and various other control relays.
(5) Analog subcommutator (1) (SASC)	Commutates sail and oriented experiment housekeeping data into a 48 word subcom frame; gating and timing is provided by the digital multiplexer-encoders located in the wheel.
(6) Analog sub-subcommutator (1) (ASSC)	Subcommutates spacecraft analog housekeeping data into a 24-word subframe before inputing to sail ASC; gating and timing is provided by sail ASC.
(7) Word Gate Generators (2) (Redundant)	Regenerate the 32-word main frame for use as read-out gates for the NRL and HCO pointed instruments; selectable by ground command.
(8) Flex cables (2)	Provides electrical connection between right and left pointed instruments and the sail junction box.
(9) Azimuth coarse eye assemblies (4)	Provides azimuth drive signals for initially despinning and pointing the sail; inputs to the pointing control system.
(10) Azimuth and elevation fine control eye assembly (1)	Provides fine azimuth and elevation error and drive signals for fine pointing control of sail and pointed instruments; inputs to the pointing control system.

Table C-2 (continuation)

Component	Functional Description
(11) Pitch control eye assembly (1)	Provides pitch error signals for maintaining the sail within ±4 deg of normal to the solar vector; inputs to the pitch control system.
(12) Pitch readout eye and amplifier assembly (1)	Provides monitoring of the pitch angle by telemetry.
(13) Pressure cartridges (Squibs)	Provide mechanical uncaging of various sail components during launch phase; are detonated by the sail squib batteries in conjunction with the launch sequence timers. Consists of the following components:
a. Nutation damper squibs (2) (redundant)	Uncage nutation damper bob at T_T + 600 sec
b. Elevation lock squibs (2) (redundant)	Unlatch elevation frame at T_T + 800 sec
(14) Squib battery packs (2) (redundant)	Provide electrical power for detonating sail pressure cartridges; each pack consists of 3 ni-cad cells arranged in series; capable of supplying 6 amps into 1/2 ohm load for two seconds.
(15) Solar cell array (1)	Converts solar energy into electrical energy for observatory consumption; produces approximately 40 watts at about 2 amps and 20 volts; consists of 960 silicon solar cells with an overall area of about 4.2 ft^2.
(16) Target eye assembly (part of fine control eye assembly, item #10)	Detects solar intensity for the purpose of determining when the pointed instruments are within approx. ±3 deg of stationary azimuth pointing; inputs to the pointing control system and causes switching from coarse to fine control when solar intensity is approx. 10 percent of maximum.
(17) Readout Eye Assembly (1)	Provides monitoring of the azimuth and elevation pointing error by telemetry.
(18) Roll coils (2)	Provides roll attitude control of spacecraft by interacting with the geomagnetic field; consists of two inductors mounted at the top of the sail, one on either side of the spin axis; either coil can be commanded on or off, positive or negative to satisfy operation requirements.

Appendix D

OSO-G

Model Specification

(Retyped from BBRC Doc. 26493
Revision D dated 19 Mar 69)

1.0   SCOPE

This specification describes the physical and functional
characteristics of the Orbiting Solar Observatory spacecraft.
The spacecraft shall be designed to carry scientific experi-
ments in a 300 nautical mile earth orbit.

Definitions of terms used in this specification are given in
paragraph 5.1.

2.0   APPLICABLE DOCUMENTS

The following documents, of exact issue shown, or the latest
revision where issue is not specified form a part of this
specification as described herein:

Document	Title
BBRC 19541-505	Spacecraft Assembly
23732	Installation Control Dwg., OSO-G/Delta Interface
BBRC 29800	OSO-G Experiment/Spacecraft Interface Specification
BBRC 26494	Test Specifications for the OSO-G Spacecraft
BBRC 21883	Environmental Test Specification for Subassemblies of the F, G and H Orbiting Solar Observatory
BBRC 19540-505	Observatory Assembly Drawing
Douglas SM-48897	Improved Delta Spacecraft Design Restraints Dated January 1966

3.0   REQUIREMENTS

3.1   Physical Requirements

3.1.1   Size and Configuration

The maximum overall dimensions of the spacecraft are shown in Ball Brothers Research Corporation (BBRC) Drawing No. 23732.

The spacecraft assembly shall be as defined in BBRC Drawing 19541-505.

3.1.2   Weight

The maximum weight of the spacecraft shall be 375 pounds.

3.1.3   Interface Requirements

3.1.3.1   Spacecraft and Launch Vehicle Interface

3.1.3.1.1 The spacecraft mechanical interface with the Improved Delta launch vehicle shall be compatible with Douglas Aircraft Company Report SM-48897, except that spacecraft rotational alignment shall be in accordance with BBRC Drawing 19540-505.

3.1.3.1.2 The following shall ocurr as part of the launch sequence:

(a) The spacecraft shall be spun about the rocket thrust axis in a clockwise direction as viewed from the launch vehicle.

(b) The spin rate, after spin-up, shall be between 70 and 134 rpm. The spin rate will be determined based on the wheel moment of inertia only.

(c) At the instant after release from the booster stage, the average spacecraft angular momentum vector shall be pointed perpendicular to the solar direction within ±8 degrees.

This error includes vehicle attitude errors at injection, tip off errors, and launch window errors. Nutation at this time shall be less than 1.5 degrees half angle.

3.1.3.2   Experiment Interface

The experiment interface with the spacecraft shall be defined in BBRC 29800, OSO-G Experiment/Spacecraft Interface Specification.

3.1.3.3   Experiment Payload Capability

The spacecraft shall be capable of carrying two solar oriented scientific experiments each weighing a maximum of 45 pounds. Maximum single experiment envelope shall be approximately 4 inches by 8 inches and 38 inches long. The spacecraft shall be capable of carrying 150 pounds of nonsolar oriented scientific experiments in five separate pie-shaped compartments. The maximum weight of any one experiment shall be 45 pounds and the maximum experiment envelope shall be approximately 13 inches wide at the front by 4 inches at the back by 13 inches deep and 8-1/2 inches high.

3.1.4   Thermal Characteristics

3.1.4.1   General

Thermal control of the spacecraft shall be accomplished passively by using spacecraft surfaces with known thermodynamic properties.

3.1.4.2   Rotating section compartment temperature excursion over an orbit shall be equal to or less than 5° C. The mean temperature shall be 15 ±5° C.

3.2   Functional Characteristics

3.2.1   Launch Conditions

3.2.1.1   The following equipment will have power applied and be operable during launch.

       (a) Command Decoders
       (b) Command Receivers
       (c) Charge Rate Monitor Circuit
       (d) Undervoltage Monitor Circuit
       (e) Tape Recorders
       (f) Transmitter
       (g) PCM Junction Box
       (h) Digital Multiplexer and Encoder
       (i) Analog Subcommutators
       (j) Digital Submultiplexer and Frame Counter
       (k) Analog Sub-Subcommutator

3.2.1.2   The spacecraft shall be capable of remaining in the launch condition for six hours before the batteries must be recharged.

3.2.1.3   The spacecraft shall perform in accordance with the orbital performance characteristics of paragraph 3.2.2 within 1400 seconds following spin-up.

3.2.2   Orbital Performance

The orbital performance characteristics are defined in terms of spacecraft operation in a 300 ±50 nautical mile, circular earth orbit inclined 33 ±3 degrees to the equator.

3.2.2.1   Power Subsystem Characteristics

3.2.2.1.1 General

The power source for the spacecraft shall be an array of solar
cells. Energy storage shall be provided by batteries to supply
peak loads and power during the spacecraft night. During space-
craft day, the solar array shall furnish power for spacecraft
operation and for charging the batteries. The solar array
(mounted on the oriented section) shall be maintained perpen-
ducular to the solar direction within 2 degrees around the spin
axis and 5 degrees around the pitch axis.

During the first six months of orbit, the solar array shall
provide an energy of 1.87 ampere hours per orbit with the space-
craft continuous bus voltage at 19 ±3 volts. The spacecraft
shall withstand a current surge of 8 amps for 5 milliseconds
with the spacecraft continuous bus voltage at 19 volts.

3.2.2.1.2 The maximum D.C. power source resistance measured at
the spin box launch bus for a 5 ±3 second duration load of 2.0
±0.5 amps, shall be less than 0.3 ohms. The maximum resistance
between the launch and the other busses are tabulated below when
fuses are jumpered as required for flight.

Wheel exp. day bus	0.13 ohms
Wheel exp. nite bus	0.18 ohms
Sail launch bus	0.30 ohms
Sail day bus	0.20 ohms

Resistance between the above listed busses and associated experi-
ment interface connectors shall not exceed 0.6 ohms for a 5 ±3
second load of .2 ±.02 amps. If there is an RF choke in series
with the power line to the instrument, the resistance shall not
exceed 2.4 ohms.

The power busses exhibit increased impedance characteristics to high-frequency signals and pulses. The exact nature of these impedances is not known well enough at this time to be included as a part of this specification.

3.2.2.1.3 Power shall be automatically removed from all experiments and spacecraft electronics, except those listed below, if the voltage at the under voltage sensor drops below 16.2 ±0.2 VDC. Power shall be reapplied when the batteries have been recharged to 19.0 ±0.2 VDC.

        a.   Rotating Section Command Decoders
        b.   Command Receivers
        c.   Array Current Monitor Circuit
        d.   Undervoltage Switch

3.2.2.1.4 The day-night turn on control circuit shall switch on the power to the pointing control electronics, and the pointed experiments when the spacecraft enters the sunlight.

3.2.2.2   Spacecraft Attitude Control

3.2.2.2.1 General

The attitude control and stabilization system of the spacecraft shall be designed to orient the spacecraft spin axis and oriented section to the sun. This system shall utilize the gyroscopic properties of the rotating section. Solar error sensors shall be used to determine alignment errors with respect to the solar direction. The spin rate of the rotating section and the spin axis orientation with respect to the solar direction shall be maintained within specified limits by attitude correcting gas jet controls. The gas jet controls shall be actuated by sun sensing error detectors. The gas jets shall be supplied with

dry nitrogen gas from storage bottles pressurized to 3000 psig at launch.

3.2.2.2.2   The spin axis of the spacecraft shall be maintained perpendicular to the solar direction within 4.0 degrees.

3.2.2.2.3   The rotating section spin rate shall be maintained between 0.44 rps and 0.68 rps by the automatic system.   The spin rate change per correction shall be 0.028 ±0.01 rps.   The "switch to manual" set points shall be 0.39 ±0.012 rps and 0.70 ±0.015 rps.

3.2.2.2.4   Roll control of the spacecraft during orbit day shall be achieved by generating a magnetic dipole parallel to the space-craft X-axis.   The dipole will be produced by current flowing through coils mounted in the sail.   The amount and direction of current may be altered by command to produce dipole moments of 0, ±2500, or ±5000 dyne-cm. gauss^{-1}.   These values will vary in direct proportion to the spacecraft supply voltage.   The tolerance for these values with a supply voltage of 19.0 volts shall be ±10%.

The system will require less than one watt to produce a moment of 5000 dyne-cm. gauss^{-1}.

The spacecraft roll position measurement shall be accurate to ±3 degrees with the roll-coil assembly turned off, and ±6 degrees with the roll-coil assembly turned on.   The solar referenced roll rate shall be less than 2 degrees per terrestrial day with the roll-coil assembly turned off.

3.2.2.3   Fixed Pointing Accuracy

3.2.2.3.1 General

The control and readout sensor assemblies shall be mounted to the end of the pointed experiments which face the sun. The surfaces on which the solar sensors are mounted shall be the reference plane from which the pointing accuracies are determined. Zero pointing error shall be when the reference surface is perpendicular to the solar direction. The experiment control and readout sensor assembly interface shall be defined in BBRC 29800, OSO-G Experiment/Spacecraft Interface Specification.

3.2.2.3.2 The spacecraft pointing controls shall orient the pointed experiments to the sun at the beginning of each spacecraft day (the oriented section shall spin with the rotating section during the spacecraft night). The pointed experiments shall be oriented, as specified below, within 90 seconds after the spacecraft enters full sunlight.

3.2.2.3.3 The pointed experiments shall be controlled about the elevation axis. The pointed experiment elevation position shall be maintained within 1 arc minute of the solar direction during the spacecraft day.

3.2.2.3.4 The pointed experiments shall be controlled in azimuth about the spacecraft spin axis. The azimuth position shall be maintained within 1 arc minute of the solar direction during the spacecraft day.

3.2.2.3.5 The pointed instrument package will have an angular motion about the average pointing direction, henceforth called jitter.

The jitter will be less than 5 arc seconds peak (clipped sinu-soid) in elevation and less than 3 arc seconds rms random motion in a band between 0.5 and 5 cps in azimuth. It is doubtful that this low a jitter can be demonstrated directly on the ground because atmospheric shimmer is normally of the order of 5 to 10 arc seconds peak-to-peak.

This jitter will be present and unchanged whether in the point or raster mode.

3.2.2.4   Offset Mode

3.2.2.4.1   General

In addition to the basic sun centered pointing as described above, the spacecraft is capable of causing the pointed instru-ment package to:

1. Generate a large raster centered on and covering the entire solar disc.

2. Generate a small raster that will cover approxi-mately 1/30 to 1/40 of the large raster area and have the capability of being centered on any one of the offset points mentioned below.

3. Provide offset point to any position in a 128 x 128 (16,384) point grid pattern centered on and covering the entire solar disc.

3.2.2.4.2   Characteristics

The characteristics of this mode are given in Table D-1 and the following paragraphs.

Table D-1

OFFSET MODE CHARACTERISTICS

Parameter	Large Raster	Small Raster	Offset Grid
Size	46 x 46 ±2 arc min.	7 Az x 7.5 El ±0.5 arc min.	46 x 46 ±2 arc min.
Resolution	64 lines	16 lines	128 x 128 points
Frame Period	491.5 sec. (24 MF/line)	30.72 sec. (6 MF/line)	------------

(a) Size Variations

The size stated is set up prior to launch for the estimated mean solar intensity, i.e., the solar intensity in orbit at an equinox.

Additional size variations can be expected from the following three sources. In orbit, the absolute size must be computed periodically using preflight calibration data and target intensity because of the following variations:

(1) A ±1.5 percent size uncertainty at launch results from a like uncertainty in the mean solar intensity.

(2) Size will vary ±3.3 percent due to the annual solar intensity variation.

(3) Size will increase on the order of 6 percent per year due to sensor degradation.

(b)   Scan Velocity

The line scan velocity is determined by the pattern size and frame time as stated in Table I.   The nominal velocities are 360 arc sec/sec and 218 arc sec/sec for the large and small raster respectively.

(c)   Pattern Synchronization

The start of each line occurs at the instant of the rise of every 24th MF word gate 32 for the large raster and every 6th MF word gate 32 for the small raster.

(d)   Servo Lags

The electronic generators instantaneously reverse in azimuth and jump to the next line in elevation.   However, the pointed instrument package momentarily lags behind the generators. The angle required for the servo to "catch up" is less than one arc minute from the start of line, i.e., about 0.6 main frames (MF).   During "catch up", deviations from the ideal may amount to as much as 0.5 arc minutes.

When changing mode, the servo will "catch up" to the electronics within 16 MF after the mode change command if the servo has been in the fine point mode immediately prior to the command.

After each raster reset occurrence, the servo "catches up" to the electronics in 16 and 6 MF for the large and small rasters respectively.

"Catch up" following acquisition at dawn will
be completed within 30 MF following the "on
target" indication.

(e)   Retrace and Reset

Retrace will always be from lower left to upper
left, i.e., the raster will start in the upper
left hand corner when facing the sun. When the
offset generators are reset, the upper left
corner of the pattern will be achieved.

(f)   Raster Reconstruction Signals

Recreation of the raster on the ground is
accomplished by use of start of line (SOL) and
start of raster (SOR) signals plus a status
monitor to indicate whether the large or small
raster has been selected. Pulses indicating
these events are available in the spacecraft.
The various experimenters will be responsible
for telemetering those signals to the ground.
The characteristics of these pulses are listed
below:

<u>SOL and SOR Pulses</u>

Width	20 ±4 msec	
Rise (SOR)	<10µs	
Fall	<100µs	
Height	10 ±2V	
Source Z	ON 1.2 K	20%
(excluding	OFF SOL 52 KΩ ±10%	
5-Ω fan-on	SOR 10 KΩ ±10%	
resistors)		
Rise (SOL)	<50µs	

Raster Status

Small Raster	<0.5 V
Large Raster	9 ±1 V
Source Z	<400 K

(g)   Generator Startup

When either raster generator is commanded on, there is a certain settling time required before the generator is within specification. This settling time is 160 main frames for large raster and 55 main frames for the small raster. The detailed position equivalent output of the generators is given below.

The large raster starts at the center of the first line within 2 seconds after the initiating command.

The small raster starts at the center of the first line within 2 seconds after the initiating command. However, this is an out-of-synchronization condition and motion should be considered to have started in the upper left hand corner. The actual output goes from center to 1/2 line past the right hand edge. The motion starts back at the sixth word 32. Each succeeding line ends nearer the right hand edge and at the seventh line, the system is within specification.

3.2.2.4.3 Performance Specifications

Table D-2 and the following paragraphs list the specifications for the performance of the operations stated in 3.2.2.4.1. These specifications are for in orbit operation.

Table D-2

OFFSET MODE PERFORMANCE

Parameter	Large Raster	Small Raster	Offset Grid
Repeatability[1]	+1.5% +0.1 minute	+1.5% +0.1 arc minute	+1.5% +0.1 arc minute
Relative Size Uncertainty	+0.7%	+2.0%	Ref.
Absolute Size Uncertainty	+2%	+3%	+2%
Pattern Error[2]	+0.4 arc min.	+0.5 arc min.	+0.4 arc min.
Pattern Center Offset	+0.1 arc min.	+0.1 arc min.	+0.2 arc min.

1. Repeatability is medium term (1 week) and assumes an eyeblock
    temperature variation of less than ±5°C and an electronic
    temperature anywhere between +10 and +35°C. It will not be
    possible to demonstrate this as a system directly with
    presently planned test equipment.

2. The primary source of error is the sensor response. Generator
    errors are about 0.1 arc minute. The error specification
    does not apply on the first line nor outside of a 40 x 40
    arc minute square for the large raster nor outside of 5 x 5
    arc minute square for the small raster. The small raster
    error specification does not apply for parts of the small
    raster extending beyond a 40 x 40 arc minute square centered
    on the solar disc.

3.2.2.4.4 Definitions

The following is a list of definitions used in describing the
offset mode characteristics and performance.

(a) Pattern Size - is the size of the ideal pattern
     when adjusted for minimum errors with an actual
     pattern. Error adjustment shall be made in the
     region specified under pattern error.

(b) Ideal Raster Pattern - is one in which line
     spacing in elevation is uniform and in which the
     line sweep rate in azimuth is constant.

(c) Ideal Offset Point Grid - is one in which all
     points are uniformly spaced.

(d) Resolution - is the number of elevation lines
     in a total raster frame or the number of grid
     points in an offset point grid.

(e) Offset Point Grid - is the grid comprised of
     all the points that the control sensor optical
     axis can point to when in the offset point mode
     of operation.

(f) Frame Period - is the time required to complete
     one frame of a raster pattern.

(g) Repeatability - is accuracy to which the space-
     craft can return to a given previous position.
     In the case of raster, the given position is
     defined in spacecraft time from the start of
     the raster frame, and in the case of offset
     point the given position is defined by the off-
     set lines commanded. The accuracy is specified
     as a percent of the distance of the given posi-
     tion from the center of the solar disc plus an
     uncertainty angle.

(h) Relative Size Uncertainty - is the uncertainty in the ratio of the size between patterns. The size ratio is important in selecting the proper point in the offset point grid. The offset point grid is used as the size reference.

(i) Absolute Size Uncertainty - is the size uncertainty in absolute angle. To obtain the absolute size estimate, preflight calibration and actual target intensity must be taken into account.

(j) Pattern Error - is the deviation in the average position of each point in the actual pattern from that of an ideal pattern. The size of the ideal pattern is adjusted to minimize errors.

(k) Pattern Center Offset - is the offset of the center of the best fit ideal pattern from the reference.

   (1) Large raster offset is referenced to the offset point grid center.

   (2) Small raster offset is referenced to the offset point about which the raster is generated.

   (3) The offset point grid is referenced to the normal "on point" position.

The center of the pattern is defined as the geometric center of an ideal pattern that is a best fit for the actual pattern within the 34 x 34 arc minute region in the pattern center.

3.2.2.5   Data Handling and Data Transmission Characteristics

3.2.2.5.1 General

Experiment and spacecraft data shall be continuously time-multiplexed. The data rate shall be 100 words per second. This information shall be stored and continuously transmitted. Upon command from a ground station, the stored information shall be transmitted instead of the "real time" information. The continuous transmission shall be called "real time" transmission, and the stored information transmission shall be called "playback" transmission. No information shall be recorded during "playback" transmission. The experiment and spacecraft information shall be pulse-coded. The code format shall be Manchester (also called biphase or split-phase).

3.2.2.5.2 A data word shall consist of eight binary bits.

3.2.2.5.3 A data main frame shall consist of 32 data words.

3.2.2.5.4 One data word of the main frame shall be for data from a 48-channel digital submultiplexer. One word of the digital submultiplexer shall indicate the main frame count to permit correlation between spacecraft real time and Greenwich mean time. The last word of each submultiplexer frame shall be a subframe synchronization word.

3.2.2.5.5 One data word of the main frame shall be for data from a 48-channel analog subcommutator in the rotating section.

3.2.2.5.6 One data word of the main frame shall be for data from a 48-channel analog subcommutator in the oriented section. The last word period of each subcommutator frame shall consist of a subframe synchronization word. The three subcommutators shall be synchronized to provide data channel correlation.

3.2.2.5.6.1 One data word of the oriented section analog sub-commutator shall be for data from a 24 channel analog sub-sub-commutator in the oriented section. The last two word periods of each sub-subcommutator frame shall be a sub-subframe synchronization word.

3.2.2.5.7 The last two word periods in each main frame shall consist of a frame synchronization word.

3.2.2.5.8 The remaining 27 word period of the main frame shall be assigned to the experimenters.

3.2.2.5.9 The transmitted carrier shall be within ±5 kc of a fixed frequency between 136 mc and 137 mc. The exact frequency shall be assigned by NASA for each spacecraft.

3.2.2.5.10 The carrier shall be angle-modulated by a pulse code signal. The peak deviation shall be 57 ±5 degrees for both playback and real-time transmission.

3.2.2.5.11 The transmitted bit rate during playback transmission shall be 14400 ±288 bits per second.

3.2.2.5.12 The transmitted bit rate during real-time transmission shall be 800 ±0.4 bits per second.

3.2.2.5.13 The spacecraft shall be capable of storing pulse-coded data at 800 bits per second for 98 minutes minimum, 101 minutes maximum. This data shall be transmitted within 338 $\pm$5 seconds after execution of a command from the ground station.

3.2.2.5.14 The transmitter output power referenced to the antenna connectors shall be +27 dbm nominal, and +25 dbm minimum.

3.2.2.5.15 Antenna polarization at the transmitter frequency shall be linear when considering the vector sum of two orthogonal components (see paragraph 5.2).

3.2.2.5.16 The transmitting antenna gain shall be greater than -6db over a minimum surface area of 3.8π steradians with reference to a linear isotropic source.

3.2.2.5.17 The following housekeeping data shall be available as part of the calibrated telemetered data from the spacecraft.

1. Temperatures shall be monitored at 18 spacecraft locations.

2. Spacecraft voltage shall be monitored at the launch bus.

3. A magnetometer pulse shall be provided once each wheel revolution.

4. A sun pulse shall be provided once each wheel revolution during orbit day.

5. Pointing and raster signals shall be provided to indicate pointing direction relative to the solar vector axis.

3.2.2.6   Command Characteristics

3.2.2.6.1   General

The command subsystem shall enable the ground controllers to command the spacecraft and experiments into different modes of operation. Each decoder shall be capable of executing a set of 70 discrete instruction commands. There are two decoders in the rotating section and two decoders in the oriented section each of which will execute any one of its 70 discrete instruction commands only upon receipt of its unique address command. Thus there are a maximum of 280 (4 x 70) address/instruction command functions. To avoid loss of critical command functions due to the loss of a single decoder, the outputs of two decoders are combined to operate a common load. Spacecraft functions of a non-critical nature are operated by single decoder outputs. Thus there are a total of 74 redundant commands. The command subsystem shall receive and decode a serial train of tone bursts which are amplitude-modulated on the command carrier frequency. The information shall be contained in three distinct tone burst widths. Separate and distinct address codes shall be used for actuating the desired decoder.

3.2.2.6.2   The command receiver shall be fixed tuned to 149.52 ±2 kHz.

3.2.2.6.3   The command carrier shall be amplitude modulated by a 7 kc duration-modulated tone. The carrier shall be modulated a minimum to 75 percent and a maximum of 100 percent.

3.2.2.6.4 A command word shall consist of 10 pulse duration modulated bits. The command word format shall be as follows:

    a.   Bit Period: A time duration equivalent to 72 cycles of the assigned audio frequency.

    b.   The first bit period in each word shall be a blank. (No modulation for a period of 72 cycles.)

    c.   The second bit shall be a "sync" bit having a time duration of 54 cycles of the assigned audio frequency.

    d.   The remaining 8 bit periods shall contain "ones" and "zeros".

    e.   A "one" shall have a time duration of 36 cycles of the assigned audio frequency.

    f.   A "zero" shall have a time duration of 18 cycles of the assigned audio frequency.

    g.   The 8 information bits of a command word shall be combinations of 4 "ones" and 4 "zeros".

3.2.2.6.5 An address word shall have the same format as the command word with the exception of the number of "ones" and "zeros" in the message. An address word shall contain a combination of 6 "ones" and 2 "zeros" or 6 "zeros" and 2 "ones". The exact address codes shall be assigned by NASA for each spacecraft.

3.2.2.6.6 A blank bit followed by a sync bit shall appear at the end of any command transmission.

3.2.2.6.7 These channels shall be assigned either to BBRC or the experiments as negotiated by BBRC, the experimenters, and the NASA Project Office.

3.2.2.6.8 The minimum threshold of the command subsystem shall be -95 dbm at the spacecraft antenna terminal when the carrier is 75 percent modulated.

3.2.2.6.9 Antenna polarization at the receiver frequency shall be right circular (see paragraph 5.2).

3.2.2.6.10 The antenna gain at the command frequency shall be greater than -18 db over a minimum surface area of 3.8π steradians with reference to a linear isotropic source.

3.3   Life and Reliability

The design objective shall be for a minimum operating life of six months in a 300 ±50 nautical mile circular earth orbit inclined 33 ±3 degrees to the equator.

3.4   Environmental Conditions

The spacecraft shall perform as required during and after exposure to the following environmental conditions. The conditions stated shall be the extreme environmental conditions expected for the OSO.

3.4.1   Temperature - Pressure

The following temperatures with an ambient pressure of 10^{-5} mmHg shall be the design requirements for components located in the indicated area.

-10°C to 35°C	Internal Rotating Section
-10°C to 40°C	External Rotating Section
-10°C to 50°C	Back of Oriented Section
-30°C to 75°C	Front of Oriented Section
-10°C to 40°C	Azimuth Casting

4.0   QUALITY ASSURANCE

4.1   The Quality Assurance Department shall verify compliance with the following requirements.

4.1.1   The spacecraft shall be tested and accepted in
accordance with BBRC 26494, Test Specification for the OSO-G
Spacecraft.

4.1.2   Spacecraft assemblies and subassemblies environmental
tests shall be in accordance with BBRC 21883, Environmental Test
Specification for Subassemblies of the Orbiting Solar Observatory.

4.1.3   Two categories of tests which shall be performed on
the spacecraft and spacecraft equipment are given below:

    a.   Qualification Test
    b.   Acceptance Test

5.0   NOTES

5.1   Definitions

    a.   Solar Direction: The line-of-sight between the
          spacecraft and the center of the sun.

    b.   Roll Axis (Fig. D-1): An axis through the center
          of mass of the spacecraft which is perpendicular
          to the spin axis and is in the plane formed by
          the solar direction and the spin axis.

    c.   Spin Axis (Fig. D-1): The axis about which the
          spacecraft is rotating.

    d.   Pitch Axis (Fig. D-1): An axis through the center
          of mass of the spacecraft which is perpendicular
          to the spin axis and to the roll axis.

SPIN-ROLL PLANE-A

SPIN AXIS

ELEVATION AXIS — ORIENTED SECTION

POINTED INSTRUMENTS

SOLAR DIRECTION

PITCH ATTITUDE ANGLE
(SHOWN NEGATIVE)

ROLL AXIS

CENTER OF MASS

ROTATING SECTION

SPIN-PITCH PLANE-B

SUN FACING END OF
POINTED INSTRUMENTS

SPIN AXIS

POINTED INSTRUMENTS

ROLL ATTITUDE ANGLE

ECLIPTIC NORTH

ORIENTED SECTION

CENTER OF MASS

PITCH AXIS

ROTATING SECTION

PITCH-ROLL PLANE-C

PITCH AXIS

ROTATING SECTION

ORIENTED SECTION

REFERENCE

ROTATION ANGLE

ROLL AXIS
(SOLAR DIRECTION)

POINTED INSTRUMENTS

Figure D-1

OSO Coordinate System

e.   Elevation Axis (Fig. D-1): The axis on which the
pointed experiments are mounted. The elevation
axis is perpendicular to the solar direction
and the spin axis.

f.   Spacecraft: The spacecraft is an Orbiting Solar
Observatory less the experiments and ballast.

g.   Spacecraft Day: That portion of the orbit in
which the spacecraft is in full sunlight.

h.   Spacecraft Night: That portion of the orbit in
which the spacecraft is in the earth's shadow.

i.   Oriented Section: The portion of the observatory
which is stationary in azimuth with respect to
the sun when the pointing controls have oriented
the pointed experiments to look at the sun.

j.   Rotating Section: The portion of the observa-
tory which is rotating in azimuth with respect
to the sun.

k.   Manchester Code: The Manchester Code is defined
by the figure below:
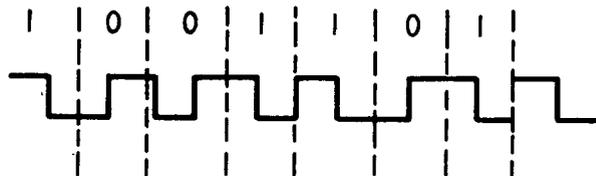
5.2   Antenna Polarization

The Orbiting Solar Observatory is a complex geometric structure which, for antenna considerations, contains two driven elements and one parasitic element with the structure serving as the ground plane. Studies have shown that the polarization of this system is neither linear nor circular, but some combination of these depending on the look angle to the observatory. Since the observatory is constantly spinning and moving through space, the look angle is constantly changing. This implies that the optimum telemetry link is achieved by diversity combining at the ground station; thus the antenna polarization at the transmitter frequency is specified to be linear when diversity combining techniques are used. The observatory antenna does not use diversity combining. To optimize the command link, it is necessary to transmit to the observatory via a right-circularly polarized antenna. Thus the antenna is specified to be right-circularly polarized at the observatory receiver frequency.