2

X-580-73-308
PREPRINT

NASA TM X- 70488

IMP-J ATTITUDE CONTROL PRELAUNCH ANALYSIS AND OPERATIONS PLAN

(NASA-TM-X-70488) IMP-J ATTITUDE CONTROL PRELAUNCH ANALYSIS AND OPERATIONS PLAN (NASA) 78 p HC $6.00 CSCL 17G
N73-33567
Unclas
G3/21 19342

H. L. HOOPER
J. B. McKENDREW
G. D. REPASS

OCTOBER 1973

GSFC

GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

78

IMP-J ATTITUDE CONTROL PRELAUNCH

ANALYSIS AND OPERATIONS PLAN

October 1973

H. L. Hooper
J. B. McKendrew
G. D. Repass

ABSTRACT

This document was prepared by the Attitude Determination and Control Section and contractor (Computer Sciences Corporation) to provide Mr. William R. Limberis, IMP Project Manager, with a description of the attitude control support being supplied by the Mission and Data Operations Directorate for the IMP-J mission. Included in the document are descriptions of the computer programs being used to support attitude determination, prediction, and control for the mission and descriptions of the operating procedures that will be used to accomplish mission objectives.

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont'd)

LIST OF ILLUSTRATIONS

Figure

LIST OF TABLES

Table

SECTION 1 - INTRODUCTION

1.1 PURPOSE AND SUMMARY

This report documents the attitude analysis performed to support the Interplanetary Monitoring Platform-J (IMP-J) mission and gives a brief description of the attitude software systems developed by CSC for the Attitude Determination and Control Section, Mission Support Computing and Analysis Division, in support of the mission.

The IMP-7 Attitude Support System has been modified to meet the attitude requirements for IMP-J. The system consists of the following four programs written to operate on the IBM S/360-95 or S/360-75 computers at Goddard Space Flight Center (GSFC): (1) an IMP Attitude Support System consisting of the Attitude Determination Subsystem, the Multi-Satellite Attitude Prediction Subsystem and the Telemetry Data Handling Subsystem; (2) an Optical Aspect Prediction Program; (3) a Quick-Look Program to be used for analyzing the telemetry data; and (4) a Bias Determination Program. A description of the IMP Attitude Support System (IMPASS) is presented later in this document.

1.2 MISSION OBJECTIVES AND REQUIREMENTS

The primary objective of the IMP-J mission is to study solar and galactic cosmic radiation, solar plasma, solar wind, energetic particles, electromagnetic field variations, and the interplanetary magnetic field. To accomplish this objective, the spacecraft conducts the experiments shown in Table 1-1.

IMP-J is a spin-stabilized satellite with a nominal spin rate of 23 rpm during normal mission lifetime. The spacecraft will be launched from the Eastern Test Range (ETR) into an elliptical transfer orbit capable of lasting at least one year. Before the first apogee of this orbit, spacecraft attitude must be determined within an accuracy of two degrees. To determine the attitude and biases, optical aspect (OA) data will be received approximately 20 hours before the

Table 1-1. IMP-J Experiments

Contributing Organization	Experiment	Principal Investigator	Project Code
FIELDS			
GSFC	Magnetic Fields	Dr. N.F. Ness	GNF
GSFC	DC Electric Fields	Dr. T.L. Aggson	GAF
University of Iowa	AC Electric & Magnetic Fields	Dr. D.A. Gurnett	IOF
ENERGETIC PARTICLES			
GSFC	Cosmic Ray	Dr. F.B. McDonald	GME
University of Chicago	Cosmic Ray	Dr. J.A. Simpson	CHE
National Oceanic and Atmospheric Administration	Energetic Particles	Dr. D.J. Williams	GWP
Applied Physics Laboratory	Charged Particles	Dr. S.M. Krimigis	APP
California Institute of Technology	Electron Isotopes	Dr. E.C. Stone	CAI
University of Maryland	Ion and Electron	Dr. G. Gloeckler	MAE
PLASMA			
University of Iowa	Low Energy	Dr. L. A. Frank	IOE
Los Alamos Scientific Laboratory	Plasma	Dr. S.J. Bame	LAP
Massachusetts Institute of Technology	Plasma	Dr. H.S. Bridge	MAP
ENGINEERING TEST			
GSFC	Data Systems Engineering Test (DST)	Mr. T. C. Goldsmith	DST
GSFC	Solar Panel Test (SPT)	Mr. E. Gaddy	SPT

spacecraft reaches apogee. The spacecraft will then be maneuvered to an intermediate attitude when the attitude will again be determined and the biases refined. A maneuver to the kick motor firing attitude will be made at about 8 hours before firing. The kick motor will then be fired placing the spacecraft in the mission orbit. This final orbit should remain within the bounds of 30 to 40 Earth radii distance for at least three years. A maneuver will be made approximately 6 hours after the kick motor firing to acquire 4.5 hours of OA data for attitude verification. A maneuver will then be made during the next 16 hours to position the spin axis normal to the ecliptic plane. Optical aspect data will be received approximately 26 hours after kick motor firing. If necessary, a trim maneuver will be made 12 days later and the 60-meter antennas will then be deployed.

The satellite will be tracked and commanded from the Space Tracking and Data Network (STDN). Data to be used for attitude determination will be transmitted to GSFC, where it will be processed on an XDS 930 computer in the Multi-Satellite Operations Control Center (MSOCC). After attitude data is stripped from the telemetry stream by this computer, the data will be sent to the IBM S/360-95 and/or -75 via a data link for processing by the IMPASS System.

1.3 SPACECRAFT CONFIGURATION

The IMP-J, as shown in Figure 1-1, is a 16-sided drum measuring 135.6 cm across the flats and 157.8 cm in overall height. The spacecraft consists of an aluminum honeycomb shelf supported by eight struts and a 45.7 cm diameter truss tube on the underside. The experiment modules are mounted on the top-side of this shelf, and a solid propellant kick motor for orbit circularization is centrally located in the upper part of the structure. To satisfy stringent RF and thermal requirements, the experiments are fully enclosed by metallic covers and side panels. Two bands of solar panels above the midsection, and one below, supply all the required electrical power. Four active and four passive turnstile RF antennas extend radially from a spacer between the two upper solar panel

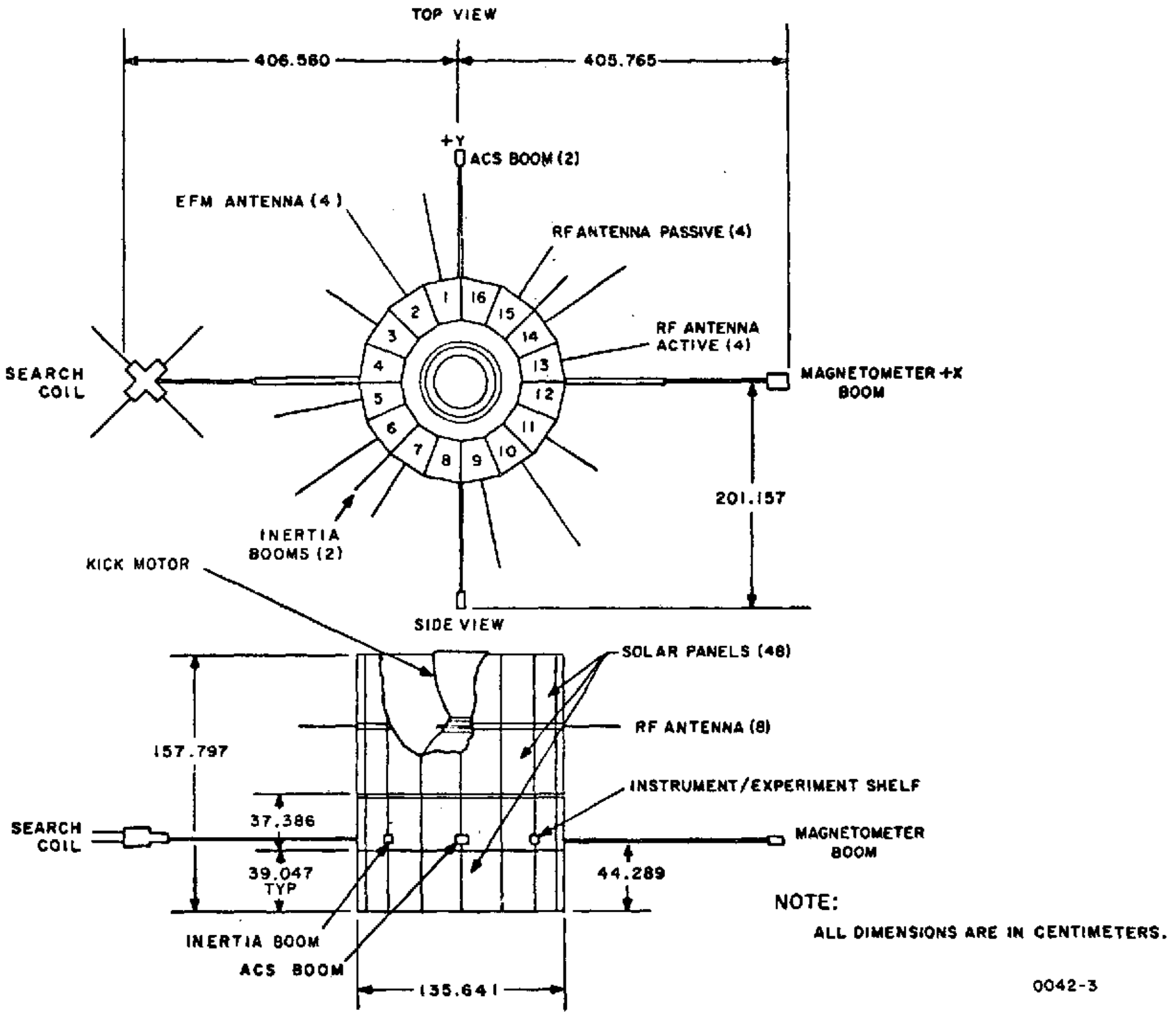


Figure 1-1. IMP-J Spacecraft Configuration

bands. Appended to the exterior of the structure are two ACS booms 1.5 meters long, two experiment booms approximately 3 meters long, and two inertia booms 50.8 cm long. These booms are designed to fold alongside the spacecraft and will be deployed at a preselected time and sequence. In addition, four 61-meter wire antennas, used by the fields experiments, will also be deployed by ground command after the mission orbit is achieved.

1.4 ATTITUDE SENSORS AND ATTITUDE CONTROL SYSTEM

The IMP-J spacecraft will be equipped with both attitude sensing and attitude control subsystems. When these onboard subsystems are combined through ground processing, spacecraft attitude can be determined and controlled to satisfy mission requirements.

1.4.1 Digital Solar Sensor

A schematic representation of a digital solar sensor is given in Figure 1-2. Incident sunlight, passing through a slit on the top of a quartz block, falls on a gray-coded pattern on the bottom of the block and, depending on the angle of incidence, illuminates or fails to illuminate each of the photo-cell detectors in the pattern. The solar sensor also includes a command slit mounted perpendicular to the gray-coded reticle.

Assuming the detector is mounted such that the command slit is parallel to the Z axis and the Z axis is coincident with the satellite spin axis, the time of illumination of the command slit provides a measurement of the azimuth angle of the Sun and the gray-coded reticle provides a digital representation of the elevation angle of the Sun with respect to the spin axis. The times of successive solar crossings of the command slit yield the spin rate.

1.4.2 Horizon Detector

The horizon detector consists of a telescope and lens system focused onto a photodiode (Figure 1-3). The telescope is mounted such that the narrow field

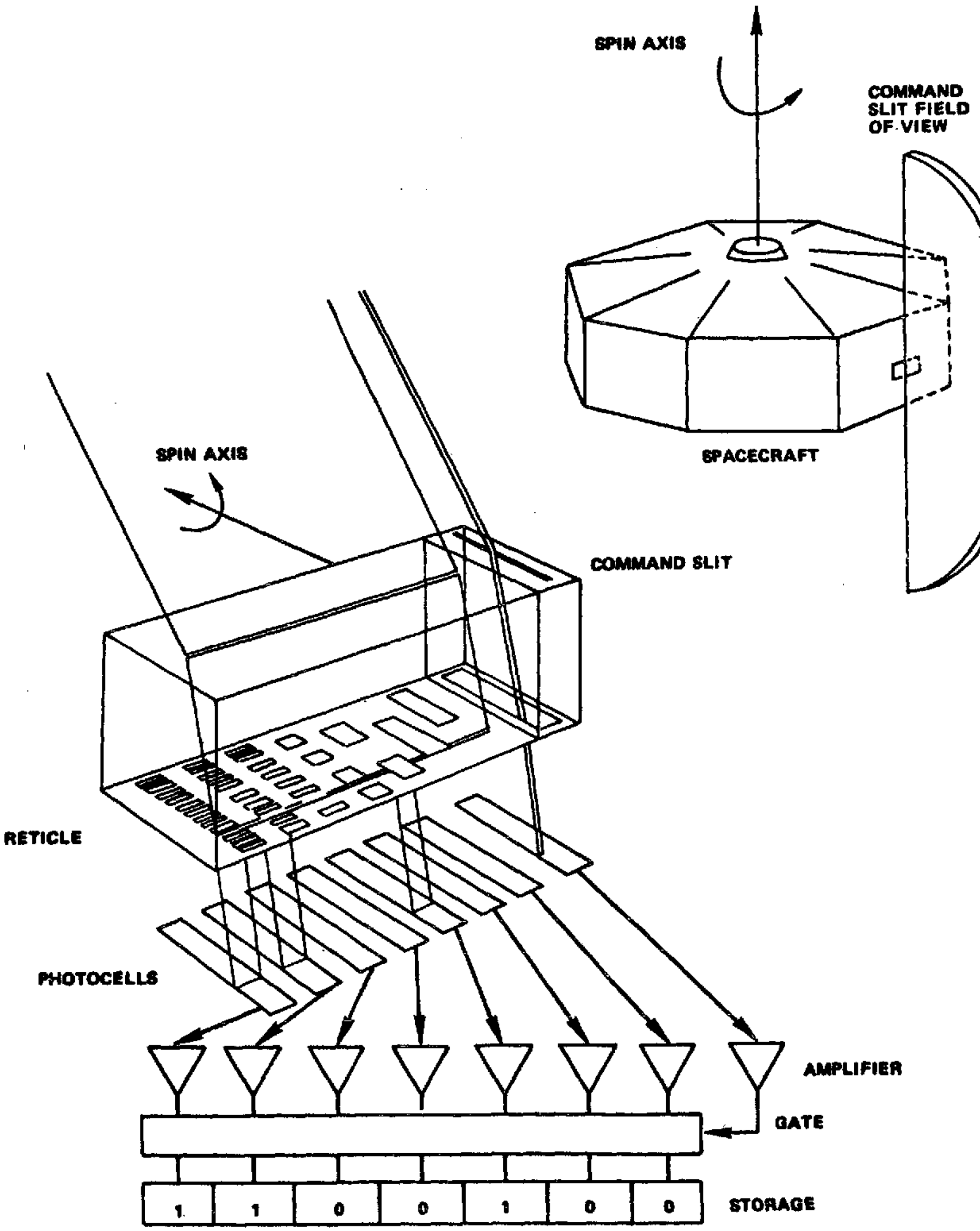


Figure 1-2. Schematic Representation of a Digital Solar Sensor

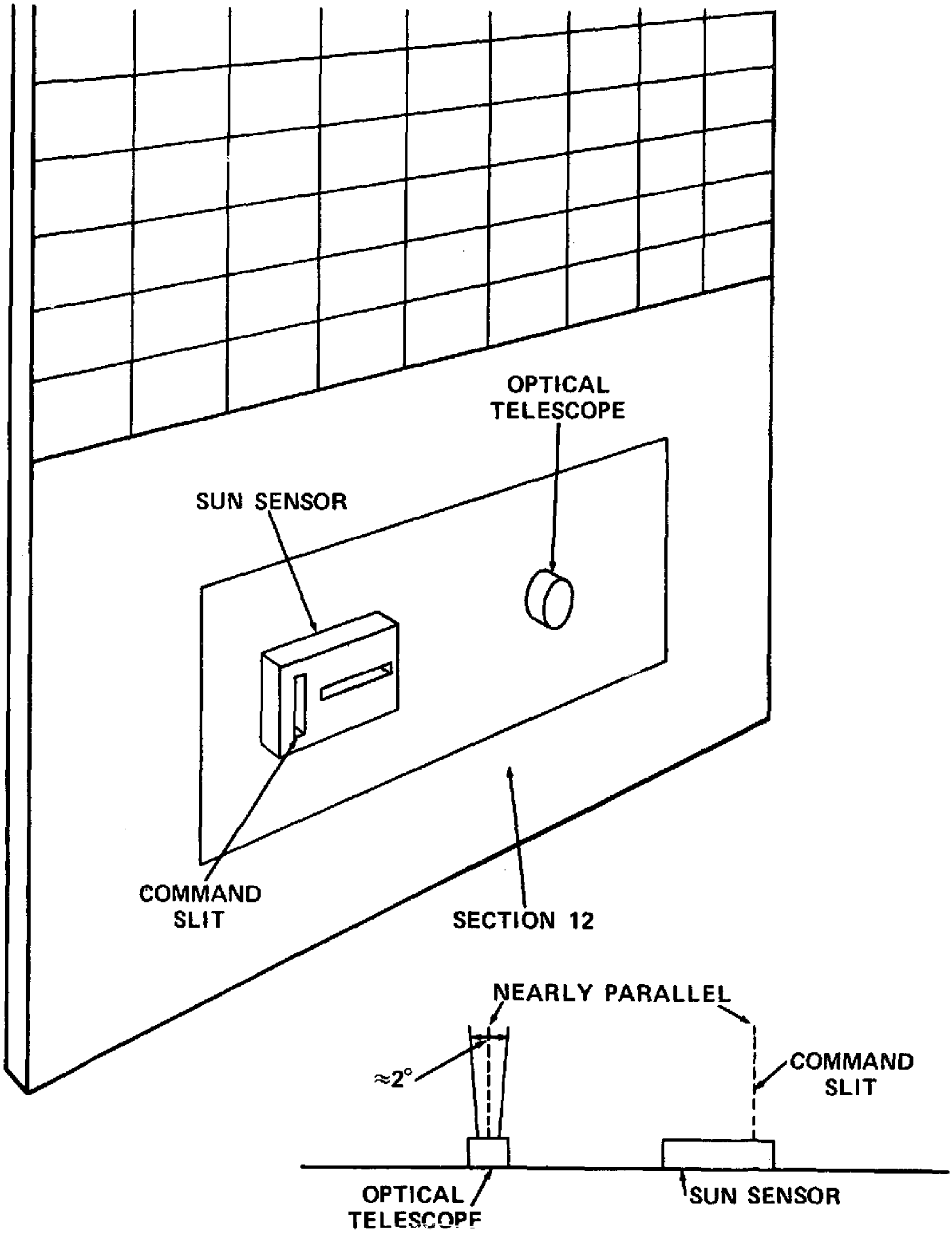


Figure 1-3. Optical Aspect Sensor Configuration

of view, 2.0 degrees, can scan 90.0 degrees from the spin axis. A positive output pulse is produced when the field-of-view of the sensor sweeps across the the discontinuity caused by the sunlit Earth against the background of interplanetary space. The time of this Earth-in pulse provides the angular distance to the sunlit edge of the Earth from the Sun in the sensor plane. An output pulse is also produced when the sensor sweeps across the discontinuity from the sunlit Earth to the background of interplanetary space. This Earth-out pulse is used in conjunction with the Earth-in pulse to determine an Earth width. This information, coupled with orbital information, is sufficient to determine the angle of the spacecraft spin axis with respect to the nadir direction.

1.4.3 Attitude and Spin Control System

The IMP-J Attitude Control System (ACS) is a cold-gas, monopropellant, pulsed-jet system whose major purpose is to achieve proper spin-axis orientation and to maintain the required spin rate. Located below the honeycomb shelf, the ACS consists of two spacecraft fuel tanks, each containing approximately nine pounds of Freon-14 fuel and a single regulator to maintain a working pressure of 40 psi. Solenoid valves permit expulsion of the fuel through expansion nozzles located 200 cm from the spin axis. Redundancy in the system is achieved through a series-parallel arrangement of four valves for each of the control and maneuver functions. To change attitude, the system is briefly actuated once during each spacecraft revolution until the spin axis precesses to the desired position. For spin control, commands of 72 seconds are executed, causing thrusting until a predetermined amount of angular momentum is imparted to the spacecraft. The commands can be broken down in smaller increments for final adjustment to spin rate.

SECTION 2 - MISSION ANALYSIS

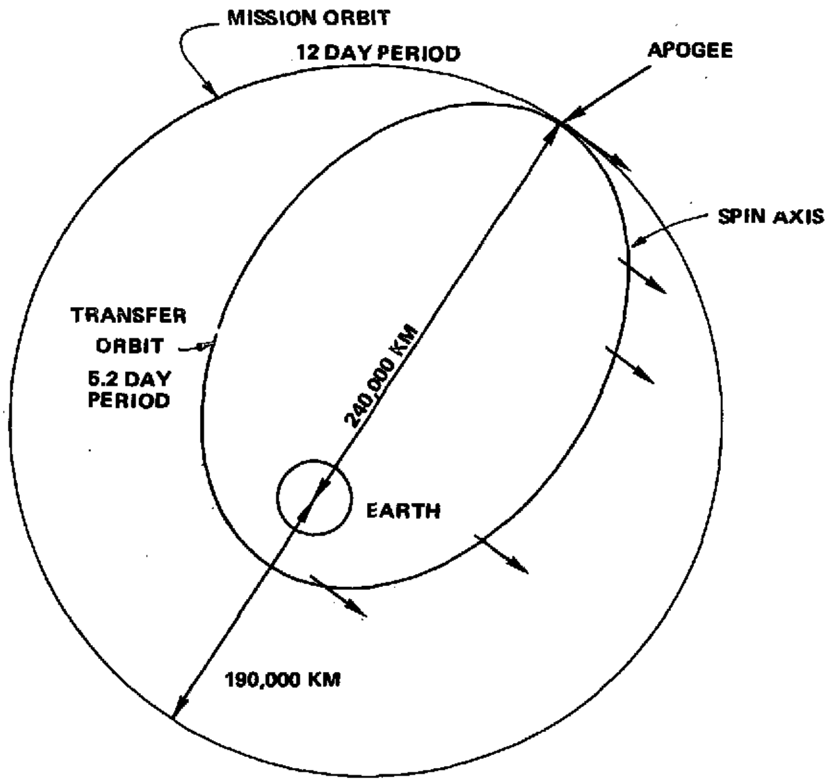
2.1 INTRODUCTION

This section describes the mission orbit, means of determining attitude, nominal optical aspect (OA) data, variations in OA data, and control maneuvers for the IMP-J mission. The IMP-J satellite will be placed in a near-circular orbit between 30 and 40 Earth radii from the Earth, with an inclination of 25.3 degrees to the equator. This orbit will be accomplished by first using a Delta launch vehicle to place the spacecraft in a highly elliptic transfer orbit, with a perigee altitude of approximately 200 km and apogee altitude of approximately 240,000 km. When the spacecraft reaches apogee, its solid propellant rocket motor will increase the speed of the spacecraft by approximately 940 meters/second, placing the spacecraft in the desired circular orbit.

The geometry of the transfer orbit is shown in Figure 2-1. The small arrow on the orbit indicates the direction of the spin axis. At injection, the direction of the spin axis will be approximately 175 degrees from the apogee velocity direction, but the spacecraft will be nose down; i.e., since the spacecraft rocket motor will be pointed up at launch, when apogee is reached the rocket motor will be pointed so that its thrust is approximately in the positive velocity direction. After the apogee burn has been made and the spacecraft is in its mission orbit, the attitude will be the same as during the burn. Shortly after apogee, the spacecraft will be maneuvered such that the spin axis is parallel to the Ecliptic Pole. This position will provide the spacecraft with a 90-degree Sun angle for the rest of the mission.

2.2 NOMINAL MISSION PLAN

Shown in Figure 2-2 is the sequence of attitude determination and control events.

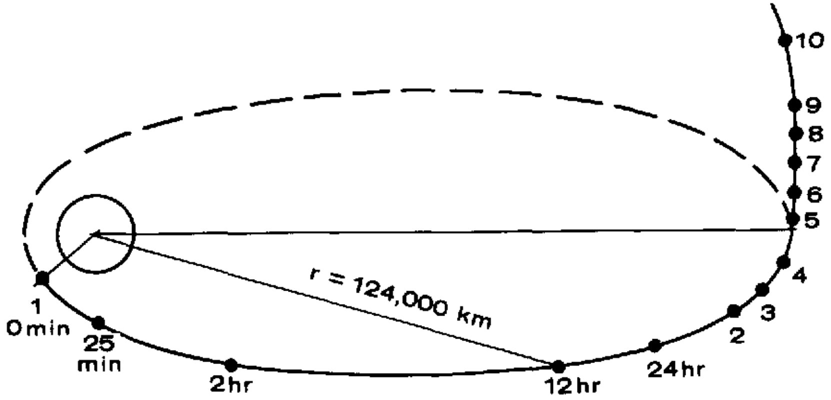


NOMINAL ORBITAL ELEMENTS

ORBITAL ELEMENTS	TRANSFER ORBIT	MISSION ORBIT
SEMIMAJOR AXIS	122805.31 KM	220041.6 KM
ECCENTRICITY	0.94646	0.09639
INCLINATION	28.705994 DEG	25.2846 DEG
MEAN ANOMALY	0.08807 DEG	222.309 DEG
ARGUMENT OF PERIGEE	120.1752 DEG	68.937 DEG
RIGHT ASCENSION OF NODE	247.997 DEG	266.5547 DEG
EPOCH	73, 10, 26, 02:30.0 YY, MM, DD, HH:MIN	73, 10, 28, 15, 30.0 YY, MM, DD, HH, MIN

Figure 2-1. Mission Geometry

IMP-J ATTITUDE AND CONTROL MISSION PLAN



	EVENT	TIME (GMT)	COMMENT
1	INJECTION	10/26/02:30 (Inj+0)	
2	RECEIVE 1st O.A. DATA	10/27/20:00 (Inj+41hr)	α = 92.21 δ = -12.82 BIASRE = 0.4
3	1st MANEUVER	10/28/01:00 (Inj+46hr)	α = 96.13 δ = -3.5
4	2nd MANEUVER	10/28/7:30 (Apo-7.5hr)	α = 100.0 δ = -4.0
5	KICK MOTOR FIRING	10/28/15:30 (M.O.+0)	
6	DESPIN TO 18 RPM	10/28/16:30 (M.O.+1hr)	
7	DEPLOY BOOMS	10/28/17:00 (M.O.+1.5hr)	
8	PARTIAL SEP MANEUVER	10/28/19:30 (M.O.+4hr)	TO RECEIVE O.A. DATA
9	COMPLETE SEP MANEUVER	10/28/24:00	α = 110.0 δ = -6.0
10	RECEIVE O.A. DATA	10/29/17:00	α = 90.0 δ = -66.7

Figure 2-2. Nominal Mission Plan

2.3 ATTITUDE DETERMINATION

The attitude of the spin axis of IMP-J is determined by processing the spacecarft ephemeris data along with the OA data transmitted to the ground from the spacecraft. The Interplanetary Monitoring Platform Attitude Support System (IMPASS) uses the OA data to determine the attitude.

Figure 2-3 illustrates the parameters measured and telemetered by the spacecraft. The parameters include the Sun angle, Earth-in time, Earth-width time, and spin period. The spin period is used to convert the Earth-in time and Earth-width time to angles.

The Earth-in time and Earth-width time are affected mainly by the threshold sensitivity of the OA sensor. This sensitivity causes early or late triggering and can produce errors in the Earth-in and Earth-width measurement.

The determination of attitude from the OA sensor is straightforward if the sensor modeling is correct or if the OA data is free from random or systematic errors. OA data is rarely received without errors. Therefore, the attitude determination problem is essentially a problem of circumventing random errors in the OA data and modeling system biases. The technique used to handle the random errors are discussed in Sections 4.6 and 8.6 of this report.

The main problem encountered in attitude determination is the biases in the data received from the spacecraft. Before the IMP Attitude Support System (IMPASS) can compute the correct attitude, the systematic errors in the data must be determined. These system biases are

1. The error in the rotation angle from the Sun to the horizons are modeled by the variables ABIAS1 and ABIAS2 for the first and second horizon biases, respectively. This error results from a misalignment of the Sun sensor and the optical telescope in the plane of rotation of the spacecraft.

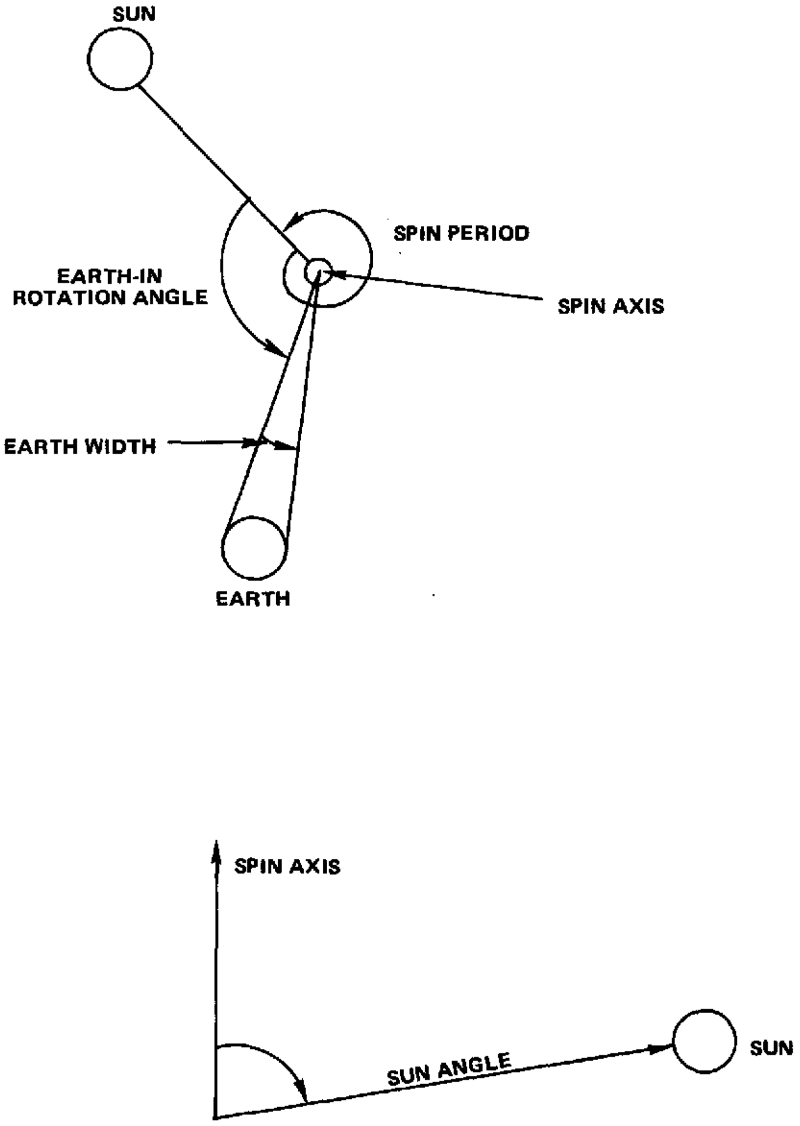


Figure 2-3. Attitude Determination Parameters

2. The OA telescope, which should be perpendicular to the spin axis, can have a slight mounting error. This error is modeled by the parameter SENANG.

3. The sensitivity of the sensor dictates the amount the viewed body (Earth or Moon) is within the field of view when the sensor triggers. The sensitivity of the sensor is modeled by the parameters BIASRE and BIASRM for the Earth and Moon, respectively. These biases affect the width of the viewed body. The more sensitive the telescope, the wider the viewed body will appear.

4. The Sun sensor has two possible errors.

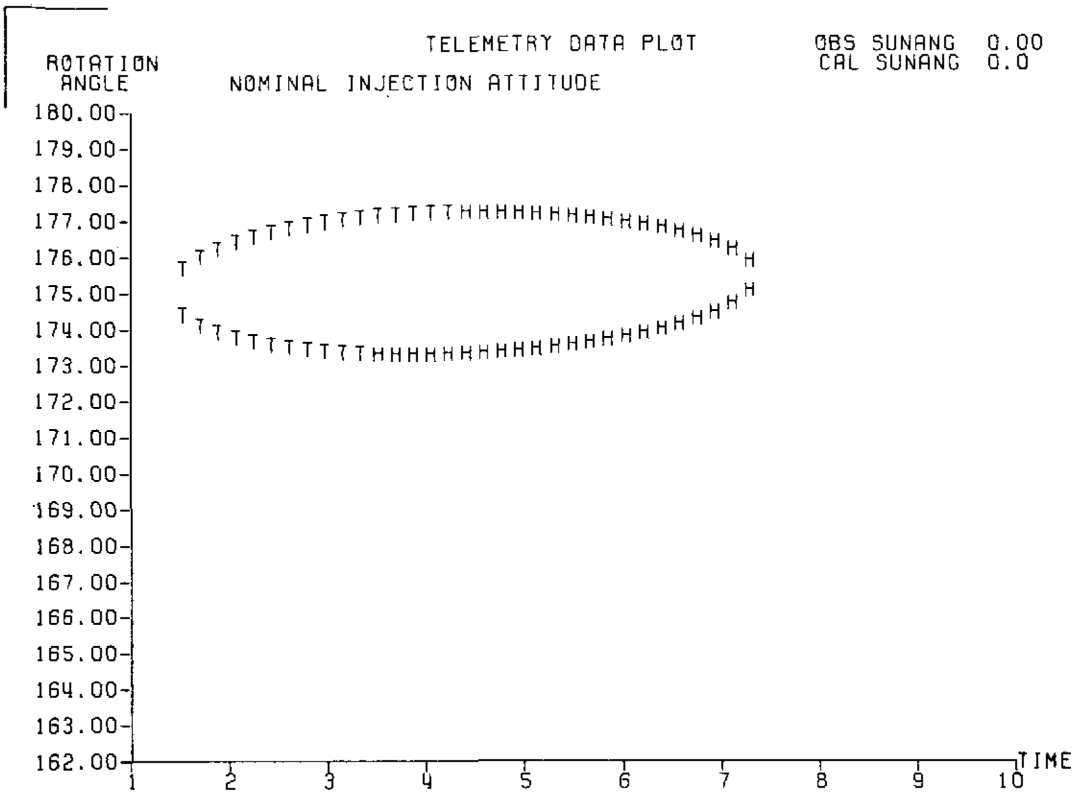
   a. The sensor is not perpendicular to the spin axis.

   b. A compromise is necessary due to the digital accuracy of the Sun sensor data. The Sun angle is measured in 0.5-degree increments. Therefore, the error can be as much as 0.5 degree. If the midpoint of the Sun angle increment is used, the maximum error is 0.25 degree. This is done by adding 0.25 degree to any Sun-angle reading from the telemetry data.

   The combination of the 0.25 degree to minimize the quantization error and the Sun sensor-mounting error is modeled by the parameter ANGDEL.

The Sun sensor and OA telescope mounting angles will be measured before launch in the test laboratory. During the mission the biases will be computed from the OA data and should agree with the measured values.

Figures 2-4 through 2-6 show the OA data availability during the transfer and mission orbits for the nominal attitudes. These figures show rotation angle, defined in Figure 2-3, as a function of time. The figures are outlines of the sunlit Earth as viewed from the spacecraft. The T's on the figure represent the terminator, and the H's refer to the horizon being lit.



1	73.300.19. 0. 0	6	73.301. 6.23.20
2	73.300.21.16.40	7	73.301. 8.40. 0
3	73.300.23.33.20	8	73.301.10.56.40
4	73.301. 1.50. 0	9	73.301.13.13.20
5	73.301. 4. 6.40	10	73.301.15.30. 0

KEY 00 CREATE CALCOMP TAPE
KEY 01 SCALE
KEY 02 RESTORE TIME SCALE
KEY 03 PREDICT DATA
KEY 04 TELEMETRY DATA DISPLAY
KEY 05 DISPLAY CONTROLLER

SCALE FACTORS
ROTMIN 162.00 ROTMAX 180.00
IPRED 1

PREDICTION VARIABLES
NPRED 50
APRA 92.210 APDEC -12.820
ABIAS1 0.0 ABIAS2 0.0
BIASRE 0.400 SENANG 90.00
TSTART 73.300.19. 0. 0. 0
TSTOP 73.301.15.30. 0. 0
ADJ + 0. 0. 0. 0. 0. 0

Figure 2-4. Optical Aspect Data Availability for Nominal Injection Attitude

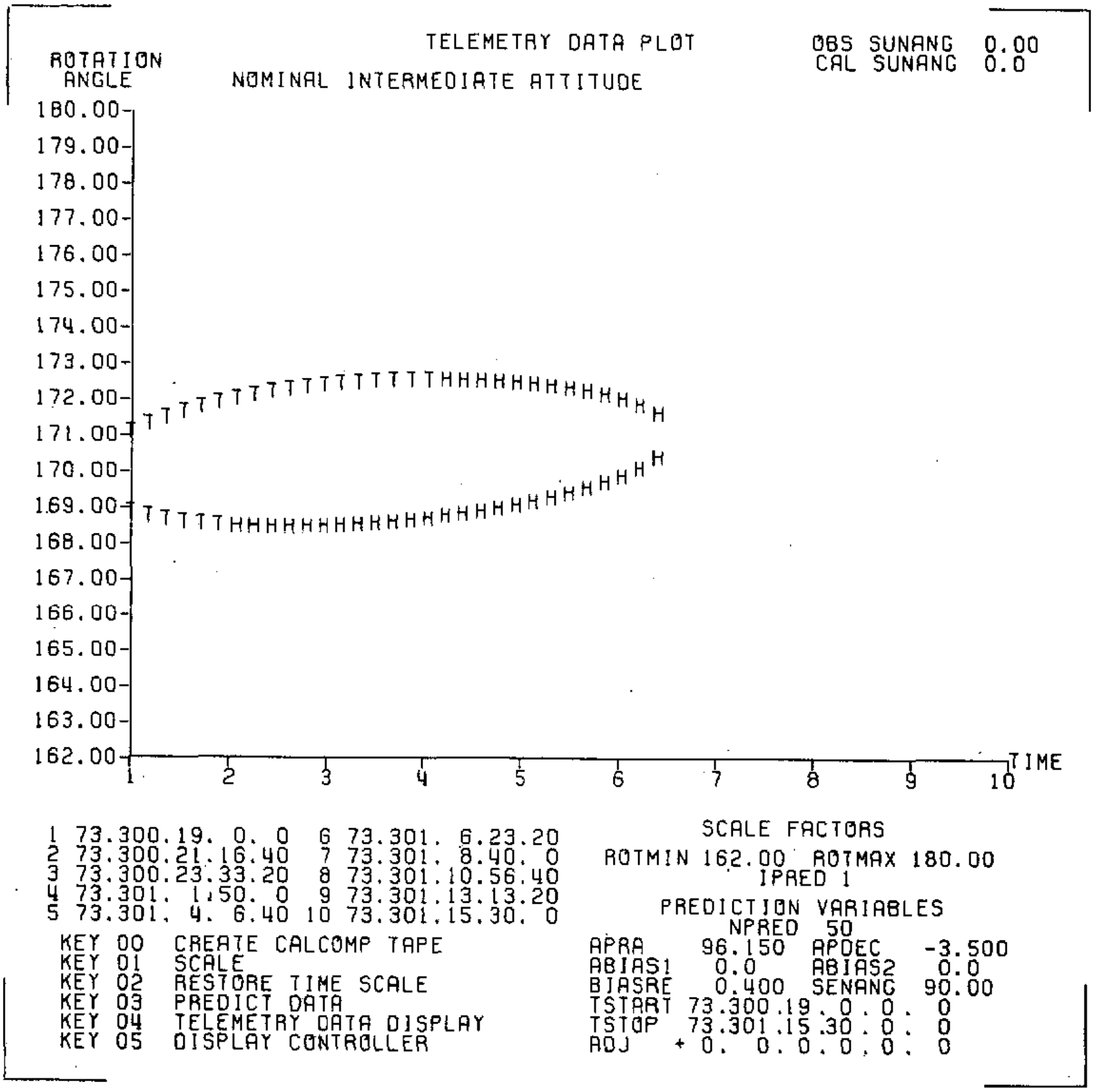


Figure 2-5. Optical Aspect Data Availability for Intermediate Attitude

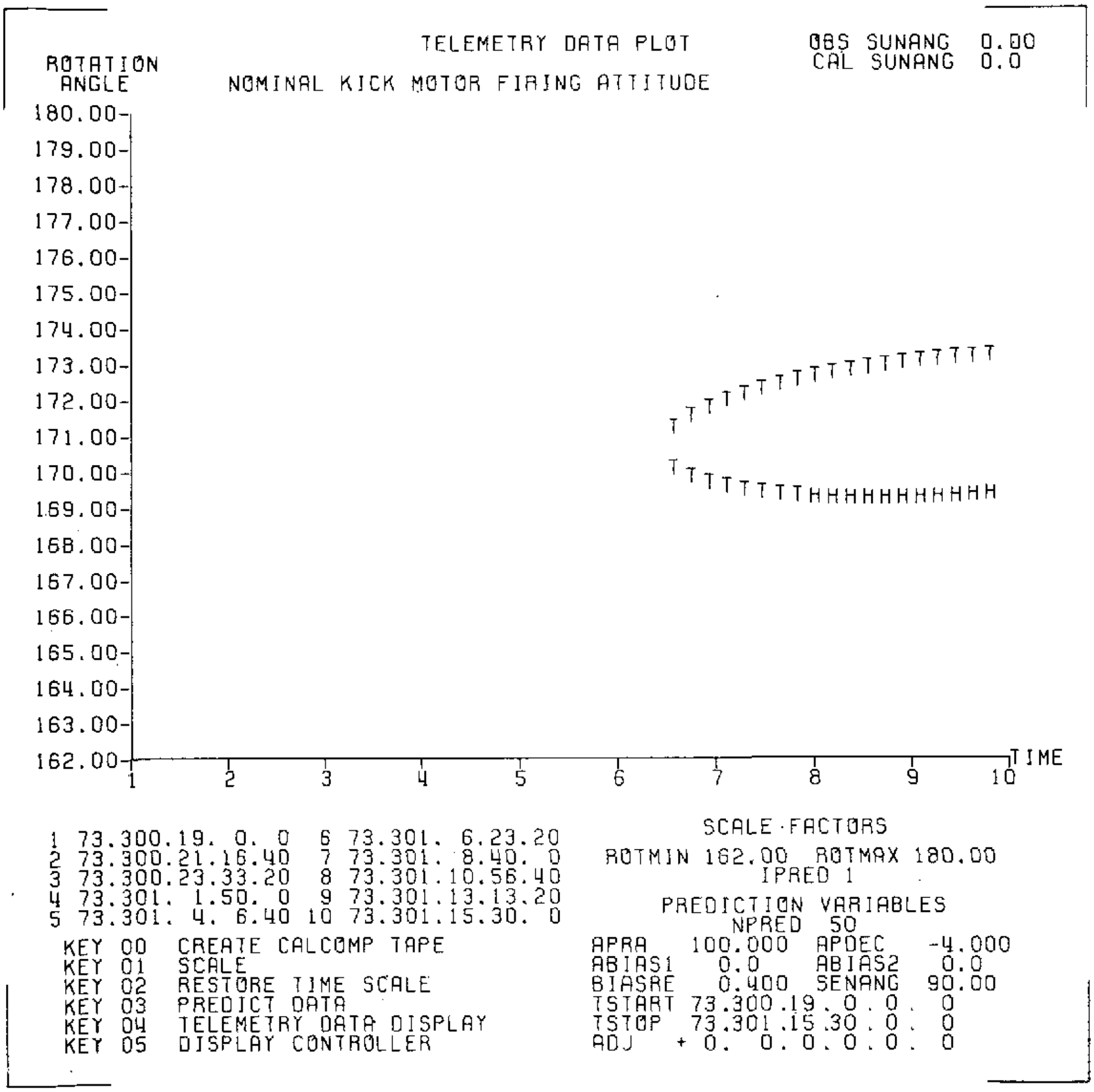


Figure 2-6. Optical Aspect Data Availability for Kick Motor Firing Attitude

Figure 2-4 shows the injection attitude with spin axis having right ascension (α) of 92.21 degrees and declination (δ) of -12.82 degrees. The values of the Earth bias, BIASRE, is set equal to 0.4 degree. The time is given below the graph in the format year, day of year, hour, minute, second. The figure is a reproduction of the IBM 2250 plot display that will be used during launch. APRA and APDEC are the right ascension and declination, respectively.

Figure 2-5 shows the intermediate attitude of 96.15 degrees right ascension and -3.5 degrees declination. Figure 2-6 shows the kick motor firing attitude of 100 degrees right ascension and -4 degrees declination.

Figure 2-7 shows the first time that OA data will be available at the South Ecliptic Pole (SEP) attitude in the mission orbit. The second time that OA data is available in mission orbit is twelve days later, as shown in Figure 2-8. Optical aspect data is not available from the Moon at the nominal attitudes during the transfer orbit, but it may be available during the first day of the mission orbit.

Coning, precession, and nutation are also possible causes of problems in determining attitude. The IMPASS system cannot determine an accurate attitude during these dynamic anomalies. The effect of these on spacecraft attitude determination is discussed in References 1 and 2.

2.4 ATTITUDE PERTURBATION

Four types of perturbation that occur during a mission affect the attitude of the spacecraft: aerodynamic forces, magnetic torques, gravity gradient, and solar pressure. None of these perturbation forces are expected to affect the attitude of IMP-J.

The aerodynamic force would only be a factor if the spacecraft did not leave the transfer orbit and made low-perigee passes. If it is decided that IMP-J should

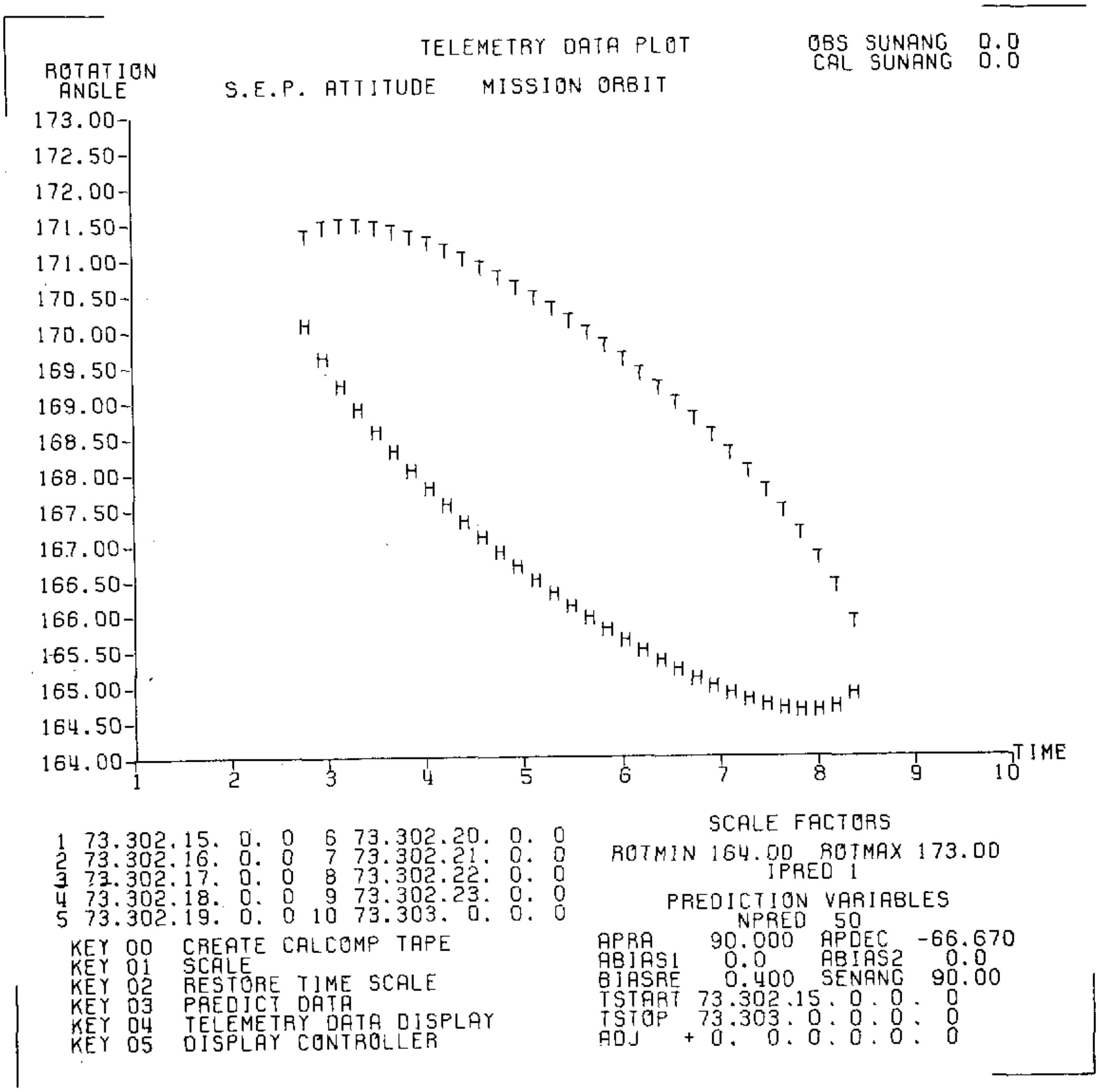


Figure 2-7. First OA Data Availability in Mission Orbit at SEP Attitude

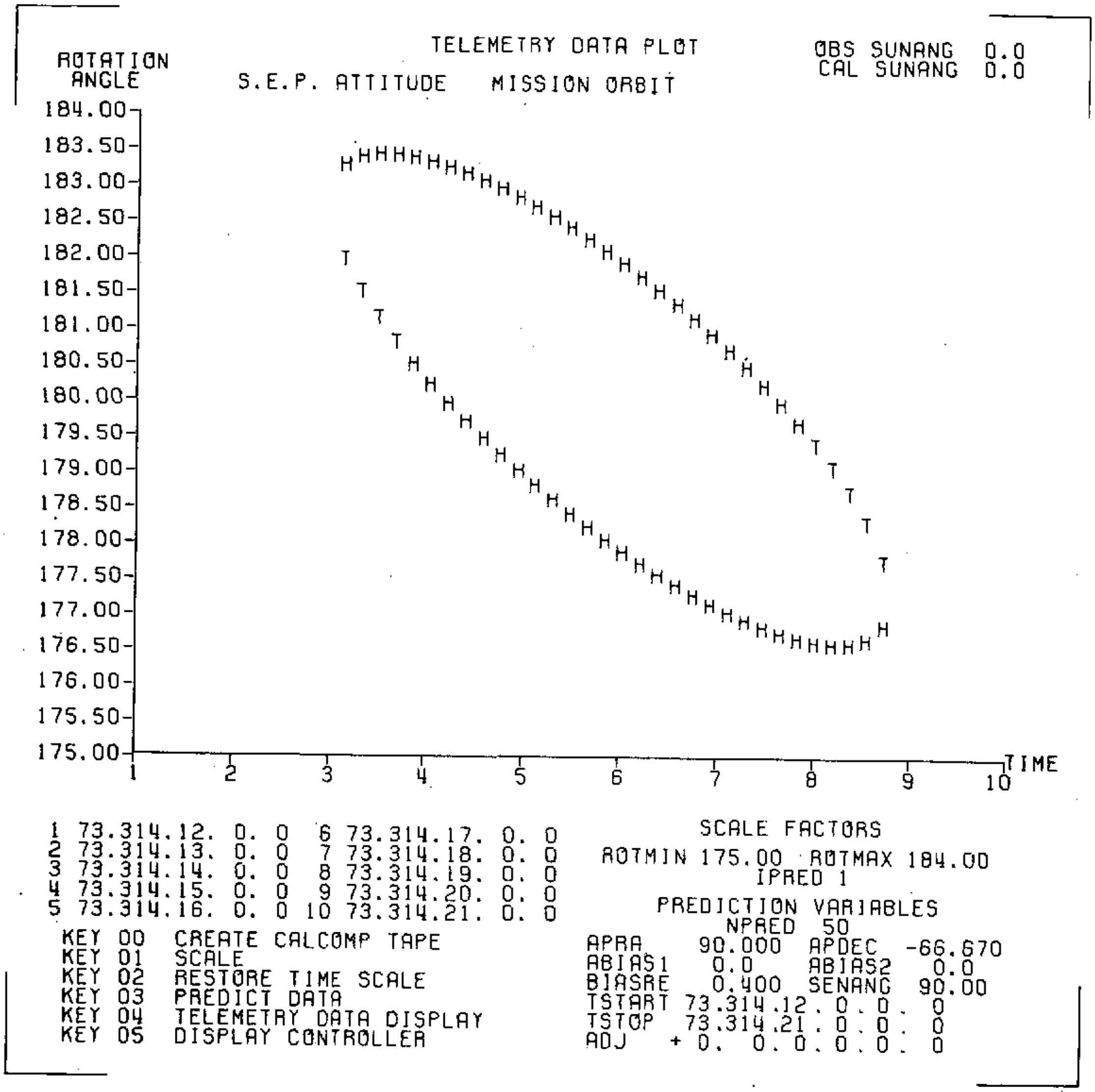


Figure 2-8. Second OA Data Availability in Mission Orbit at SEP Attitude

stay in the transfer orbit additional revolutions before apogee burn, the aerodynamic effects on attitude will be negligible because

- Spin rate is very high
- The spin vector is essentially along the perigee velocity vector
- Perigee altitude is high (estimated at 287 km)

Magnetic torques on the spacecraft would occur only if the spacecraft stayed in the transfer orbit. Even then, the change of the spacecraft attitude due to the magnetic torque would be negligible because of the high spin rate and because the magnetic dipole of the spacecraft is extremely small. The gravity gradient torque is also negligible due to the high spin rate and great distances from any central body. The solar pressure has essentially no effect on the spacecraft attitude in either the transfer orbit or the mission orbit.

2.5 ATTITUDE CONTROL SYSTEM ANALYSIS

The IMP-J ACS consists of a Freon cold-gas thruster system. This system is used to control the attitude and spin rate of the spacecraft. The thrusters are mounted on booms, which are folded along the sides of the spacecraft before the kick motor is fired. Then, the booms are unfolded until they are perpendicular to the spacecraft. The amount of angular motion or precession of the spin axis produced by the thruster is computed as follows:

$$\Omega = \frac{F \cdot \ell}{I_Z \omega}$$

where F = thrust magnitude (pounds)

ℓ = distance from spin axis to the thrsuter (feet)

I_Z = moment of inertia about the spin axis (slug-feet^2)

ω = spin rate (radians/second)

t = pulse duration computed as follows:

$$t = \frac{\beta}{\omega}$$

where β = angular pulse rate (radians).

Table 2-1 presents a summary of these variables used for determination of spin-axis precession in the following modes of spacecraft operation: booms folded, booms unfolded at the slow spin rate, and booms unfolded at the fast spin rate.

The change in spin rate is given by the following:

$$\dot{\omega} = \frac{F \cdot \ell}{I_Z} t = \frac{\text{torque time}}{\text{moment of inertia}}$$

Table 2-2 presents a summary of the variables for spin change.

The spin-axis precession commands are sent from the tracking station in units of eight pulses. Each pulse lasts for 22.5 degrees of spacecraft rotation. The pulse-on command is relative to when the OA system senses the Sun. The IMP-J spacecraft is designed to precess in only four directions: north toward the Sun; south away from the Sun; east in the direction of the Sun vector crossed into the spin vector (therefore, perpendicular to the Sun); and west, which is the negative of east. The north-south pulses are turned on perpendicular to the Sun, thus producing a torque toward or away from the Sun, and east-west pulses are turned on in line with the Sun.

The thrust pulse of the ACS will be displaced from the desired position by an amount in degrees approximately equal to the spin rate in rpm divided by 4. This error is in the direction of the rotation, causing each maneuver to be

Table 2-1. Spin Axis Precession Variables

PARAMETERS	BOOMS FOLDED	BOOMS UNFOLDED SLOW SPIN	EFM ANTENNAS DEPLOYED
Ω (DEG/COMMAND)	0.0524	1.32	0.053
F (LB)	0.13	0.13	0.13
ℓ (FT)	3.4	6.6	6.6
ω (RPM)	46.0	9.4	23
I_Z (SLUG-FT^2)	67.77	126.53	521.01

Table 2-2. Spin Change Variables

PARAMETERS	BOOMS FOLDED	BOOMS UNFOLDED SLOW SPIN	EFM ANTENNAS DEPLOYED
ω (RPM/COMMAND)	2.9	4.8	1.16
F (LB)	0.13	0.13	0.13
ℓ (FT)	2.12	6.6	6.6
I_Z (SLUG-FT^2)	67.77	126.53	521.01

biased slightly in the next direction, i.e., a north maneuver is slightly west of north and west is slightly south of west. This occurs because the OA sensor signals the thrusters when the pulse should start; however, there are delays in the solenoid valve and thrust buildup and decay. There is a 15-millisecond delay associated with the opening of the solenoid valves. The thrust requires 60 milliseconds for buildup. Approximately 22.5 milliseconds are required for the valve to close. This is followed by 60-millisecond thrust tail off. All of these result in a shift in the center of the thrust pulse to be 11.5 degrees later at 46 rpm and 3 degrees later at 12 rpm. At the higher spin rate, the delay causes a significant change in the direction of the maneuver. This anomaly is modeled in the control computation.

There will be two control maneuvers before kick motor firing. After kick motor firing there will be a spin down before boom deployment, a precession maneuver to the ecliptic plane, a trim maneuver for this attitude, and then a spin-up maneuver before antenna deployment. The precession commands will be sent at 25.5-second intervals. The total number of commands required for precession from the injection attitude of 92.2 degrees right ascension and -12.8 degrees declination to the nominal firing attitude of 100 degrees right ascension and -4 degrees declination will be 300. This means that the total maneuver time before kick motor firing will be approximately 2 hours. The total time required to send the command to maneuver to the south ecliptic pole will be approximately 1-1/2 hours.

SECTION 3 - ATTITUDE SUPPORT SYSTEM

3.1 SYSTEM OVERVIEW

The IMP-J Attitude Support System consists of three subsystems (Figure 3-1): the Telemetry Data Handling Subsystem, the Attitude Determination Subsystem, and the Multi-Satellite Attitude Prediction (MSAP) Subsystem. The functions of these subsystems are discussed in the following subsections.

3.1.1 Telemetry Data Handling Subsystem

The Telemetry Data Handling Subsystem provides a direct interface between the raw telemetry data being transmitted from the XDS 930 computer in the Multi-Satellite Operations Control Center to the IBM S/360-95 and/or -75 at Goddard Space Flight Center via a direction data link facility. The subsystem places the data into a satellite-dependent data set on disk.

3.1.2 Attitude Determination Subsystem

The primary function of the Attitude Determination Subsystem is to determine attitude accurately from the raw telemetry data obtained from the spacecraft. Specific objectives of the subsystem are to

- Provide necessary interface with the Telemetry Data Handling Subsystem and the MSAP Subsystem
- Approximate the spacecraft attitude using the quick-look procedure available in the optical aspect attitude determination module
- Determine the spacecraft attitude to within two degree of accuracy using the differential correction procedure available in the optical aspect attitude determination module
- Provide an interactive graphics capability via the IBM 2250 Display Unit to monitor and control the subsystem flow

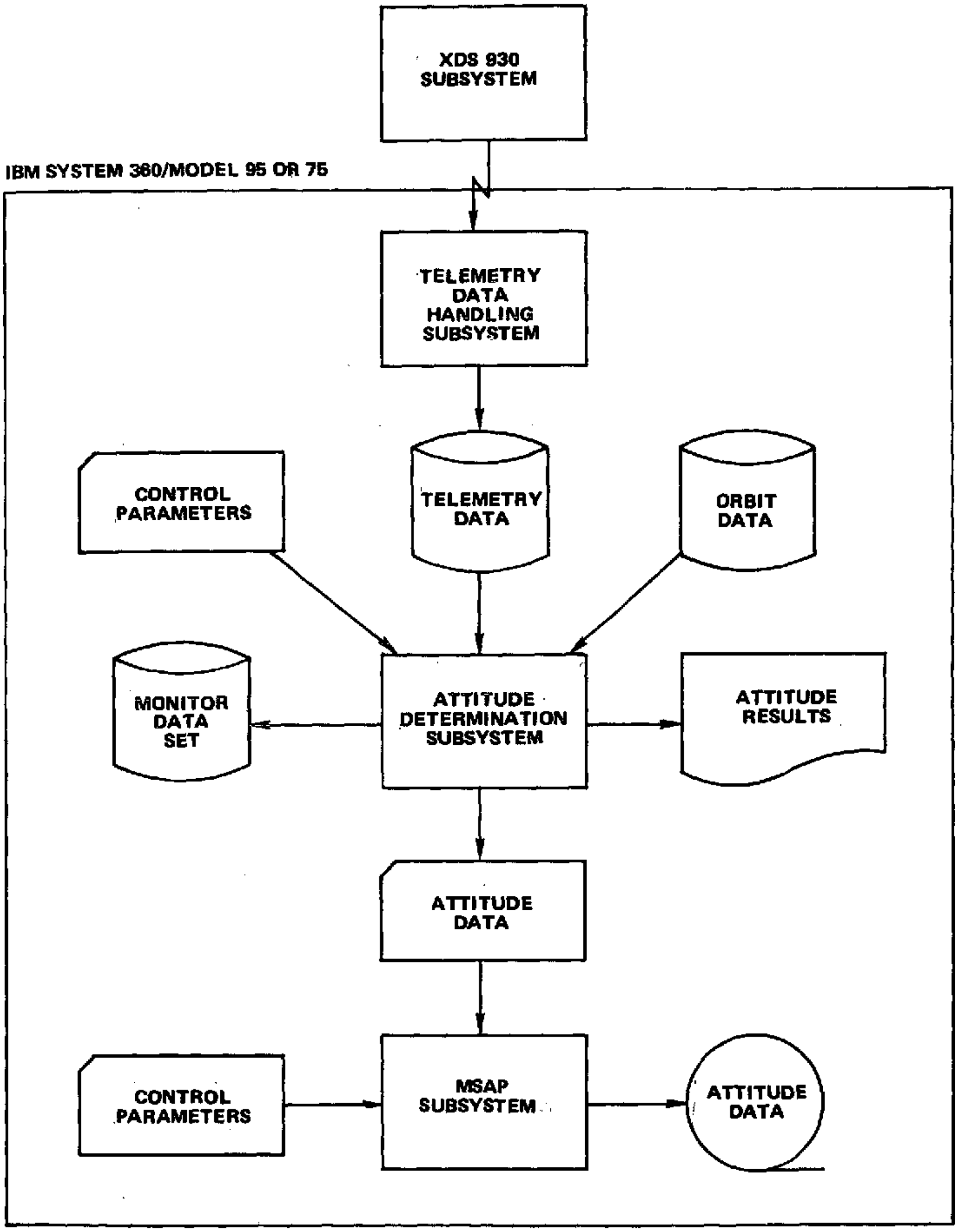


Figure 3-1. IMP-J Attitude Support System Overview

- Provide attitude determination results in a readily usable format for evaluation

3.1.3 MSAP/IMP-J Subsystem

Control of the IMP-J spacecraft is accomplished through the MSAP/IMP-J Subsystem. Since several onboard experiments require that the spin axis be normal to the Sun line, MSAP will be used to reorient the spacecraft's spin axis parallel to the ecliptic south pole while the spacecraft is in the mission orbit.

3.2 ATTITUDE SUPPORT SYSTEM REQUIREMENTS

The major requirements for the IMP-J Attitude Support System are to:

- Provide an initial attitude determination after separation from the third-stage launch vehicle
- Generate attitude control commands to orient the spacecraft spin axis to the desired attitude for the apogee kick motor firing
- Provide an attitude determination for the spacecraft following the firing of the kick motor
- Generate attitude control commands to orient the spacecraft spin axis normal to the ecliptic plane
- Provide an attitude determination for the spacecraft when the final orbit and attitude have been achieved.

SECTION 4 - ATTITUDE DETERMINATION SUBSYSTEM

4.1 SUBSYSTEM OVERVIEW

The IMP-J Attitude Determination Support (IMPADS) Subsystem processes optical aspect data telemetered from the IMP-J spacecraft and calculates the attitude of the spacecraft's spin axis. A conversational graphics interface is provided between the user and the IMPADS Subsystem to enhance the system's support capabilities. The subsystem is programmed in FORTRAN IV and ALC languages and is intended to be operational on the IBM S/360-95 or -75 computer at GSFC.

The IMPADS Subsystem consists of four main modules (Figure 4-1).

- IMPEXC--executive control module
- IMPTEP--telemetry preprocessor module
- GRADIS--graphic display module
- OA--optical aspect attitude determination module

The primary function of the executive module, IMPEXC, is to provide control of program initialization, program flow, and program output. In addition, IMPEXC establishes all necessary interfaces with the subsystem's other modules.

The telemetry preprocessor, IMPTEP, reads, unpacks, converts (to engineering units), and validates the telemetry data. IMPTEP reads the raw telemetry data from a satellite-dependent disk file generated by the Telemetry Data Handling Subsystem or from a magnetic tape generated on the XDS 930 computer. Telemetry data in engineering units may also be input via any card-image device.

The display module, GRADIS, provides communication between the IMPADS Subsystem and the user via a 2250 Display Unit. The module displays the optical aspect and control information input by the preprocessor and displays the

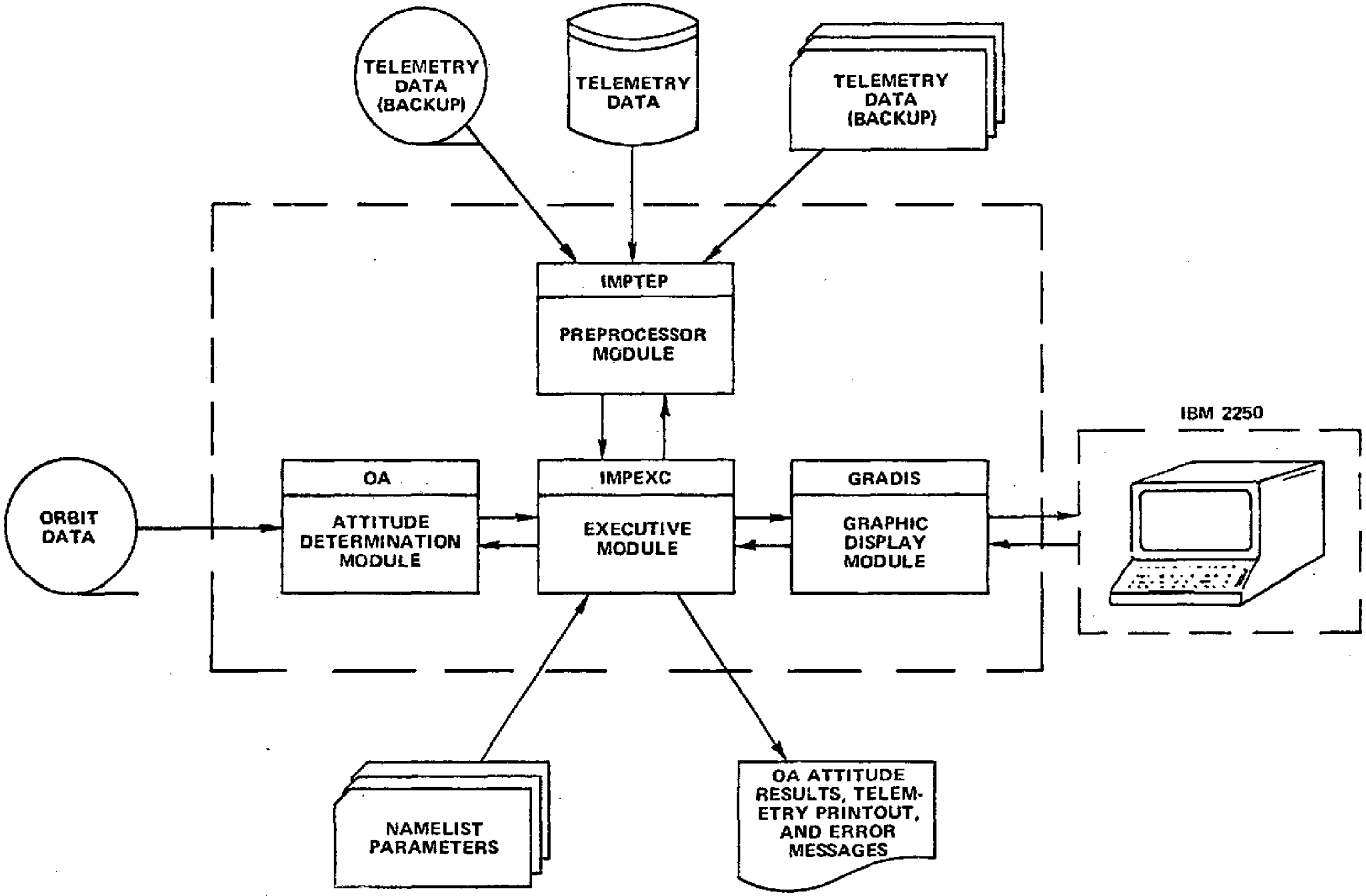


Figure 4-1. Attitude Determination Subsystem Overview

results of attitude calculations returned by the OA module. GRADIS provides the capability of retrieving and concatenating selected intervals from the available telemetry data and excluding data samples within an interval from attitude calculations. The module provides the user with a means of respecifying or overriding NAMELIST parameters during program execution. In addition, GRADIS produces the following displays:

- Telemetry data display
- Plot of observed and predicted telemetry data versus time
- Alpha versus delta display
- Plot of residuals (o-c) of Sun angles, nadir, and dihedral angles
- Summary of differential correction
- Results of quick-look calculation

The function of the OA module is to determine the attitude of the spacecraft's spin axis (expressed in right ascension and declination) from Sun-sensor and Earth-scanner telemetry data.

4.2 ATTITUDE DETERMINATION SUBSYSTEM DATA FLOW

The IMPADS Subsystem can be initiated through the card reader, the GTS system, or the CRJE system. Input parameters for both IMPADS and the OA module are read via the NAMELIST feature through some device containing card images: cards, magnetic tape, CRJE file, or disk file. The two NAMELISTs can be read from different devices.

Input telemetry data (sample time, Sun and Earth times, Earth width, spin rate, Sun angle, and gas pressures and temperature) is read by the preprocessor module, IMPTEP. In the primary mode of operation, this data is read from a bit string on the satellite-dependent disk file IMPFILE, which is generated by the Telemetry Data Handling Program on the IBM S/360-95 computer. When this disk file is not available, several backup modes of operation can be used: a tape generated by the XDS 930 in the same format as the satellite-dependent disk file; cards punched from the listings on the XDS 930 printer; or card images on a magnetic tape, CRJE file, or disk file. Since the input unit number is a NAMELIST variable, more than one tape can be used as the input. The particular mode of operation being used is defined through the NAMELIST IMPNAM and the JCL.

When the 2250 display is being used, the operator can

- Set flags to reject or accept data
- Request the method of attitude determination desired and specify the data to be used for the determination
- Specify the interval of data to be preprocessed
- Alter the time check parameters, the maximum spin rate, and the angle adjustment factors
- Terminate the run

When the display is not in use, the type of attitude determination, data used for attitude determination and preprocessing, time check parameters, maximum spin rate, and angle adjustment factor are input through the NAMELIST IMPNAM.

The following reports are optional when the IMPADS Subsystem is executed: NAMELIST reports, error message reports, attitude determination reports containing both the input and the results, a telemetry data report containing all attitude and control information for each preprocessed sample, and a listing of all records read when the data is included in card images.

The data flow for all modes of operation is illustrated in Figure 4-2.

4.3 EXECUTIVE MODULE

The IMP-J Executive Module (IMPEXC) is the main executive routine controlling program flow between the modules of IMPADS. IMPEXC provides the necessary internal data management and housekeeping functions within IMPADS and issues

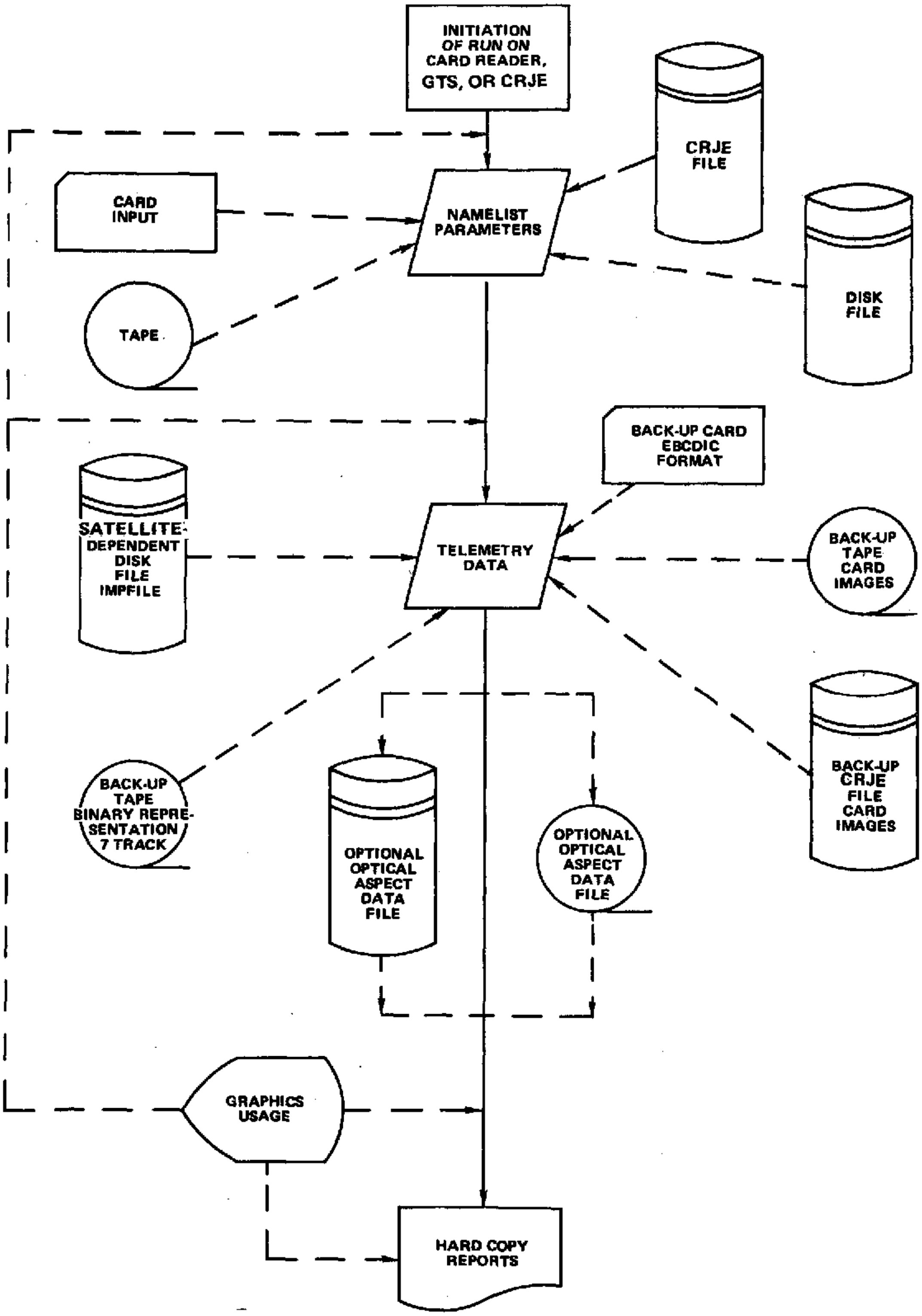


Figure 4-2. IMPADS Subsystem Data Flow

calls to IMPTEP and GRADIS when these modules are required to perform their respective functions. IMPEXC also generates the interface with the OA module for the attitude calculation given in right ascension and declination through quick-look, differential correction, or residual edit calculations. In addition, IMPEXC generates the printed summary reports listing information from the NAMELIST parameters, optical aspect data used for the attitude calculations, output returned by the OA module, and control and attitude information obtained from the telemetry data.

4.4 TELEMETRY PREPROCESSOR MODULE

The preprocessor module (IMPTEP) performs the following functions:

- Converts, when necessary, the time interval of data to be processed to GMT milliseconds of year
- Searches data records for the correct interval specified in either record numbers or times
- Reads the telemetry data
- Converts, when necessary, telemetry data to the required engineering units
- Places the converted data into buffers common to all modules of the IMPADS Subsystem
- Flags data records with erroneous transmission rates
- Flags data values out of sync
- Flags data records with times out of range
- Flags data records with erroneous IDs
- Flags data values out of range
- Adjusts times for an interval of data as requested

4.5 GRAPHICS MODULE

The graphics display (GRADIS) module provides the user with a conversational interface with the IMPADS Subsystem. This interface provides the following capabilities:

- Variable FORTRAN unit numbers for the input telemetry data set and ephemeris data set
- Input data screening
- Data rejection
- Manual and uniform adjustment of telemetry data
- Attitude determination requests
- Automatic quick-look attitude determination requests
- System and optical aspect (OA) NAMELIST initialization
- Plotting of telemetry data and attitude solution results
- Program termination control

The graphics display module provides a means of communication between the IMPADS Subsystem and the user via the 2250 Display Unit. The module can generate telemetry and control data displays, three attitude solution displays, system and OA NAMELIST displays, and plot displays of telemetry data and OA attitude results. GRADIS also includes an option display called the Display Controller, which centralizes selection of the nine other displays. From the Display Controller, the 2250 operator may transfer to any other graphic display in the module. In addition, he may select telemetry data intervals, write to the monitor data set, or terminate the run.

The graphic display module allows the user to reinitialize system and OA NAMELIST parameters as often as desired. GRADIS also permits the user to enter telemetry and control data into the data arrays and modify the data for use by the OA module. Both quick-look and differential correction attitude determination requests may be implemented after the data in the arrays has been properly edited. Two additional features, an automatic data processing mode and an automatic quick-look mode, are included in the module for use when telemetry data is being received via the data link in near real time. The GRADIS user also has access to four plot displays, one of the telemetry Earth-in and Earth-out rotation angles, one of the most current differential corrections solution results versus time, one of alpha versus delta, and one of the residuals versus time. Finally, GRADIS allows the user to monitor current error conditions in the IMPADS Subystem by displaying diagnostic error messages on the various displays.

The GRADIS module consists of the following 17 subroutines:

Subroutine	Description
GRADIS	Graphics control subroutine, which interfaces with the executive module
DISCON	Display controller subroutine, which presents a list of all display options
OADISP	Optical aspect NAMELIST display subroutine, which produces a display of selected OA NAMELIST parameters
SYSDIS	System NAMELIST display subroutine, which produces a display of IMPNAM NAMELIST parameters
TELDIS	Telemetry data display subroutine, which produces a display of telemetry data
TELPLT	Telemetry data plot display subroutine, which produces a plot of telemetry Earth-in and Earth-out times versus rotation angles
CONDIS	Control data display subroutine, which produces a display of control information
DCDISP	Differential correction solution display subroutine, which produces a display of computed attitudes obtained from the OA module

Subroutine	Description
DCPLOT	Differential correction plot display subroutine, which produces a plot of alpha and delta versus time for the most recent differential correction
SQLDIS	Single frame quick-look solution display subroutine, which produces a display of single frame quick-look solutions obtained from the OA module
BQLDIS	Block quick-look solution display subroutine, which produces a display of a block of quick-look attitudes solutions obtained from the OA module
PRODIS	Processing status display subroutine, which produces a display indicative of the operator request currently being executed
AVDPLT	Attitude solution plot display subroutines, which produces a plot of alpha versus delta for the most recent differential correction or block quick-look solutions
RESPLT	Residual plot display subroutine, which produces a plot of the residuals from calculations of Sun angles, nadir angles, and dihedral angles versus time
CMPRES	Compress telemetry data subroutine, which deletes flagged samples from data arrays
DELETE	Delete subroutine, which flags telemetry data samples to be deleted
REPLAC	Replace data subroutine, which replaces a data array item with the contents of a work area
UNIFRM	Uniform data adjustment subroutine, which adjusts all entries in an array by a fixed amount

The IMPADS graphics capability is an option which the subsystem user may specify via the NAMELIST. When graphics usage is not indicated by the appropriate NAMELIST parameter, control is never passed to GRADIS.

4.6 OPTICAL ASPECT MODULE

The optical aspect attitude determination module is designed to handle data from a single onboard horizon detecting optical sensor in conjunction with a solar aspect detector. The measurements available from the sensor complement are the time that the Sun was sensed, the time that the Earth was sensed, the width of the Earth scan in units of time, and the spin axis-Sun direction cone angle. Only horizon crossings are processed for the attitude determination calculations; therefore, terminator crossing data must be logically rejected. The module has as its primary input a file of preprocessed telemetry information, consisting of a string of sensor output times directly transformable to attitude information and an instrument-measured solar aspect read-out. This information is made available to the module in blocks, the extent of which is specified at execution time. The attitude calculation is based on single horizon crossing data. Also incorporated into the module is a least-squares differential correction procedure based on the generalized cones program GCONES. The output of the deterministic procedure for attitude determination is passed to the GCONES procedure as an initial estimate if a differential correction is desired. The central body for the attitude calculations will be considered as either the Earth or the Moon. In most cases, the module will function without benefit of an initial a priori estimate of the spacecraft attitude. Data flow for the Optical Aspect Module is given in Figure 4-3.

4.6.1 Analytical Techniques

4.6.1.1 Sun Angle Processing

The purpose of the Sun angle processing is to smooth Sun angles and reject outliers and invalid data. Any telemetry frame with a negative Sun time or with a Sun angle less than zero or greater than 180° is rejected. The remaining frames are used to compute a linear least-squares fit of the Sun angles versus the Sun times. The standard deviation of the Sun angles from the fitted

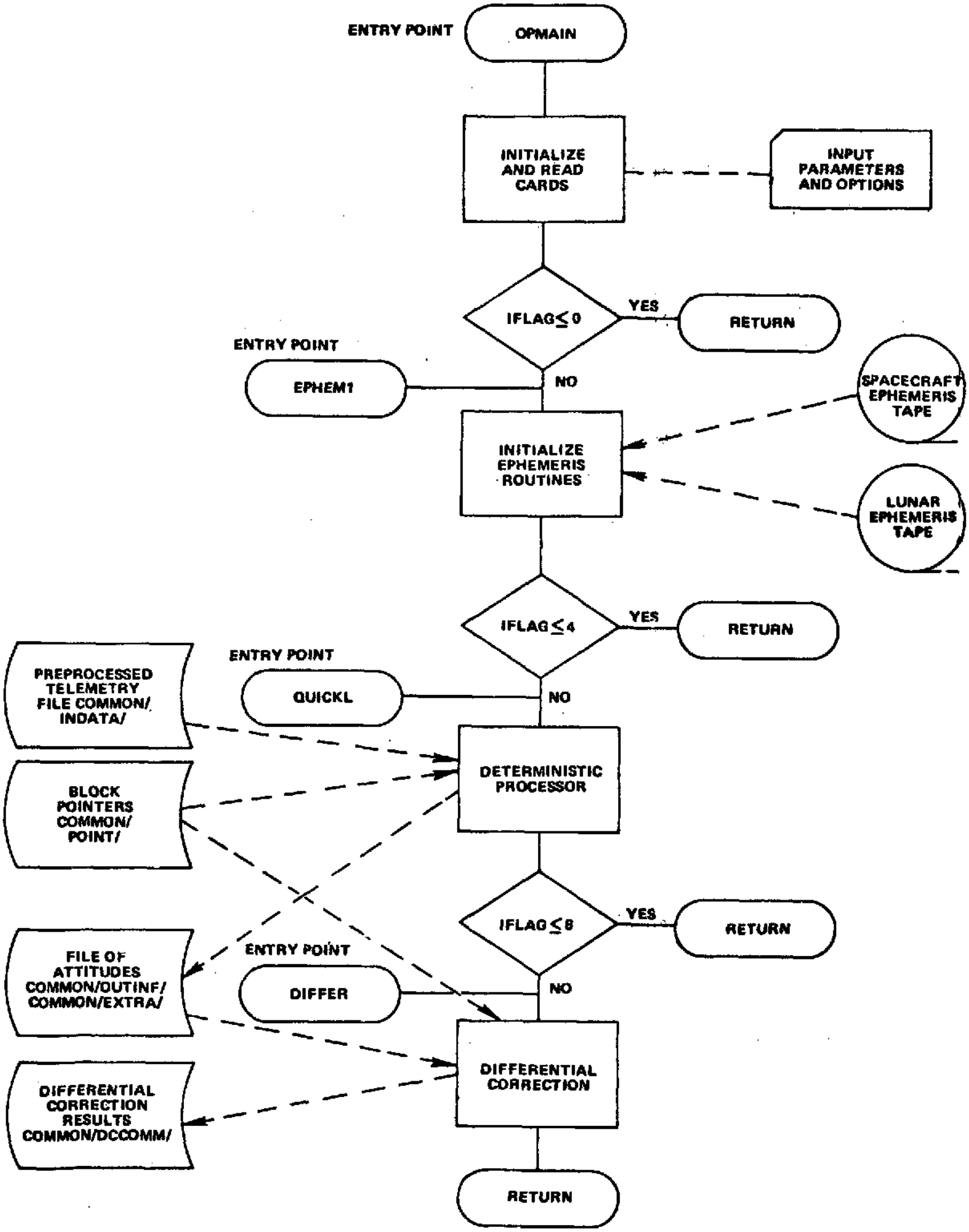


Figure 4-3. Optical Aspect Module Data Flow

curve is computed. Spurious values are rejected and the computation of the linear fit is repeated.

After the linear fit is complete, the smoothed Sun angle corresponding to each valid Sun time is saved for use in the differential correction. In addition, the Sun angle is evaluated at each sensor triggering time and saved for use in the single frame processing. Thus the data can be considered to have been simultaneously measured at the horizon sensor triggering time.

The utility of this procedure depends on a number of factors. The user may wish to turn off the option to smooth Sun angles if the spacecraft is undergoing nutation or if for any reason the Sun angle is varying more rapidly with time than would be expected to result from a constant attitude.

4.6.1.2 Single Horizon Crossing Computation

As many as two possible attitudes are calculated from a single horizon crossing event, with the corresponding nadir angles and dihedral angles.

Three spherical triangles must be solved to obtain the required nadir angles and corresponding dihedral angles (Figure 4-4). Each nadir angle defines a cone around the vector from the spacecraft to the central body, and each Sun angle defines a cone around the Sun vector. A utility routine is used to compute two attitude vectors defined by the intersection of these two cones. One of the attitude vectors corresponds to a dihedral angle greater than pi and the other attitude corresponds to a dihedral angle less than pi. The computed dihedral angle must be used to choose between the two attitude vectors obtained for each nadir angle in order to avoid a four-fold ambiguity. The fact that a two-fold ambiguity remains will be shown below.

The first spherical triangle is formed by the spin axis, the Sun vector, and the horizon crossing point, as shown in Figure 4-5.

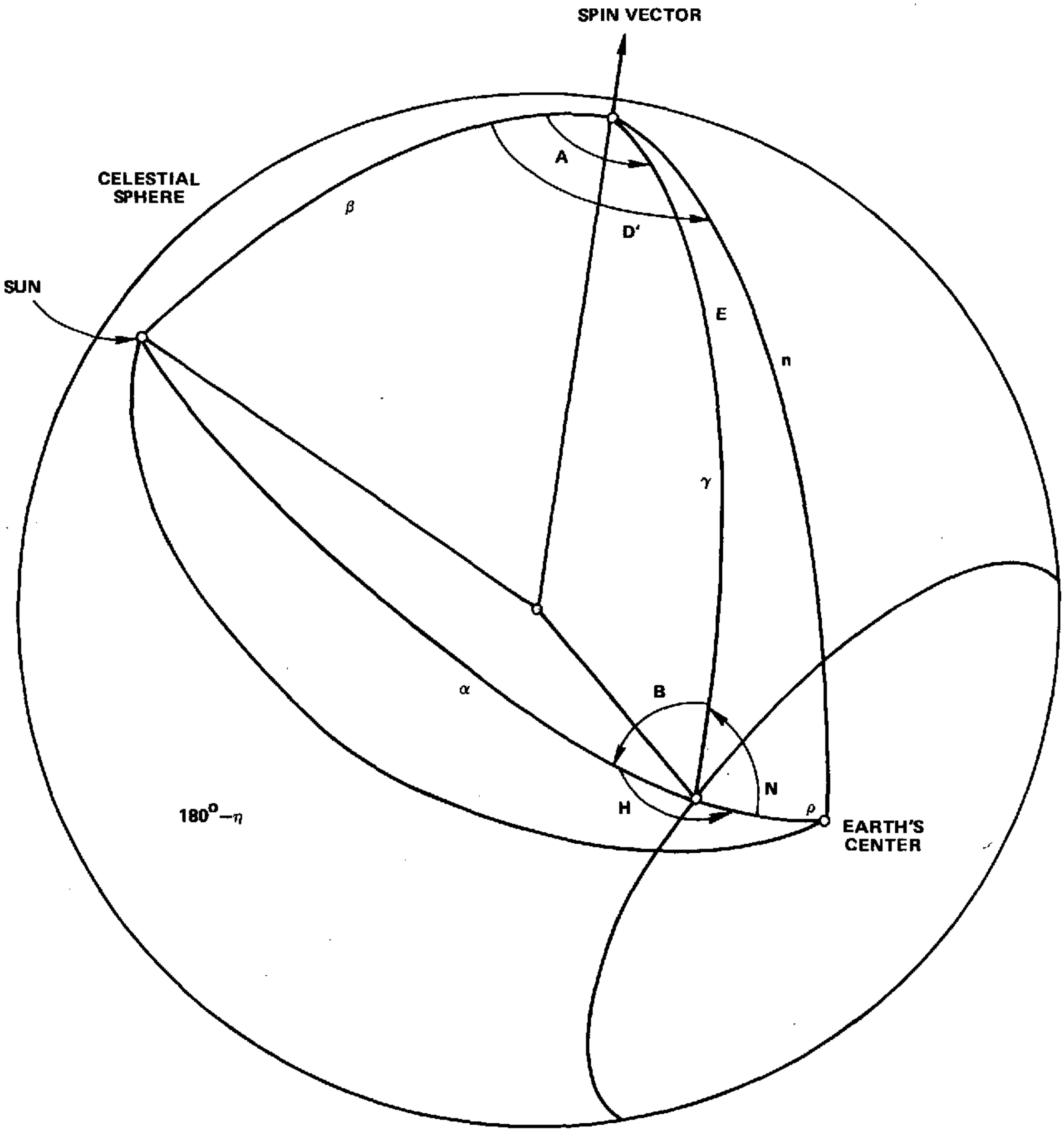


Figure 4-4. Single Horizon Crossing Geometry

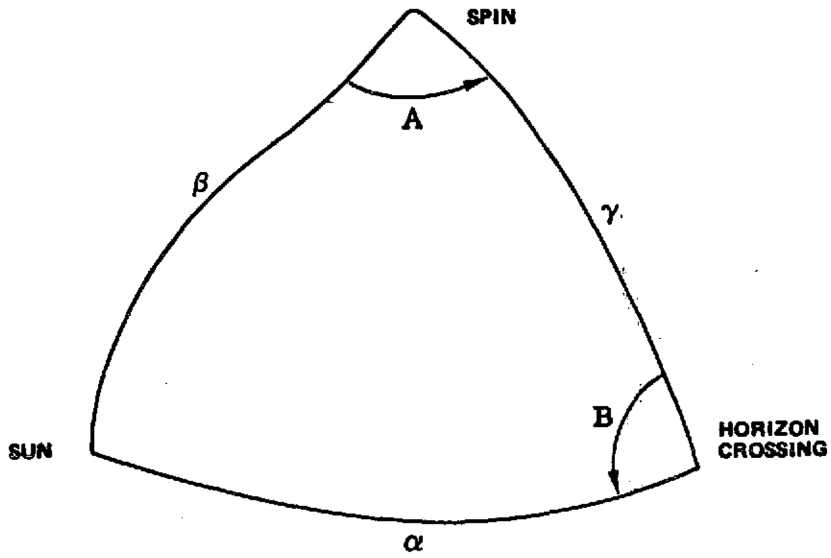


Figure 4-5. Horizon Crossing Spherical Triangle

The Sun angle, β , the sensor mounting angle, γ , and the spacecraft rotation angle from the Sun, A , are given so that α and B can be determined. From the law of cosines,

$$\cos \alpha = \cos A \sin \beta \sin \gamma + \cos \beta \cos \gamma$$

If abs $(\cos \alpha) > 1$, no solution is possible. Otherwise

$$\alpha = \arccos (\cos \alpha)$$

and

$$\cos B = (\cos \beta - \cos \alpha \cos \gamma)/\sin \alpha \sin \gamma$$

Again if

$$\sin \alpha \sin \gamma = 0 \text{ or abs } (\cos B) > 1$$

no solution is possible. Otherwise

$$B = \arccos (\cos B)$$

Putting the solution into the proper quadrant, if the dihedral angle A is greater than pi, it is necessary to set

$$B = -B$$

The second spherical triangle is formed by the Sun vector, the horizon crossing point, and the vector to the center of the central body, as shown in Figure 4-6.

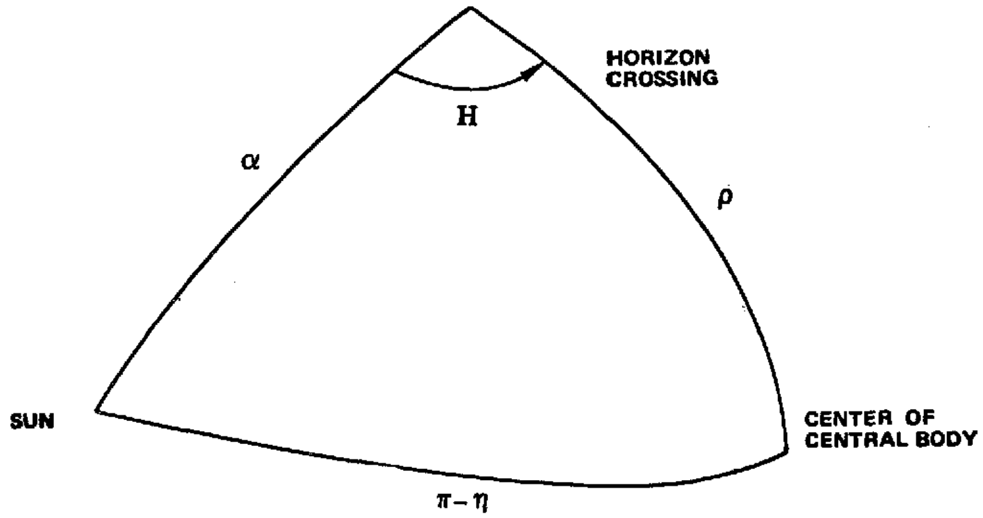


Figure 4-6. Sun Vector, Nadir Vector Spherical Triangle

The angular radius of the central body, ρ , and the arc length from the Sun to the center of the central body, $(\pi - \eta)$, can be computed from spacecraft and solar ephemeris. The angle α is as computed above. Calculations for H are

$$\cos H = \left(\cos(\pi - \eta) - \cos \alpha \cos \rho\right) / \sin \alpha \sin \rho$$

If

$$\sin \alpha \sin \rho = 0 \quad \text{or} \quad \text{abs}(\cos H) > 1$$

no solution is possible. Otherwise, there are two solutions for H , as follows:

$$H_1 = \arccos(\cos H)$$

and

$$H_2 = -H_1$$

The remaining calculations must be performed for $i = 1$ and $i = 2$. If solutions are obtained for both cases, then there will be two possible attitudes, two nadir angles, and two dihedral angles for this crossing. If one of the solutions is rejected in the computations which follow, then there may be a single unambiguous attitude, nadir angle, and dihedral angle for the crossing.

The third spherical triangle is formed by the spin axis, the horizon crossing point, and the center of the central body, as shown in Figure 4-7.

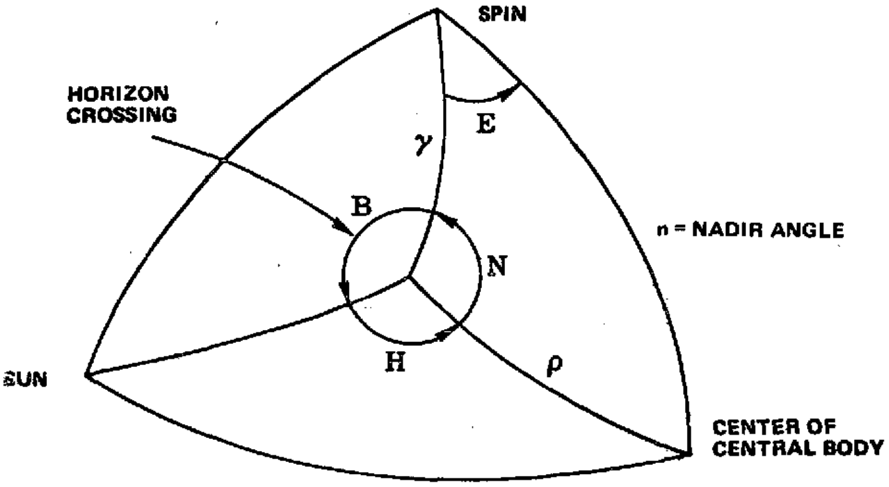


Figure 4-7. Schematic Drawing of the Three Relevant Spherical Triangles

The angles ρ , the central body angular radius computed from ephemeris information, and γ , the sensor mounting angle, are known. Therefore, the angle N can be computed, since

$$B + H_i + N = 2\pi$$

That is, the angles around a point on the celestial sphere sum to 360°. So

$$N = 2\pi - B - H_i - 2\pi k$$

where $k = 0, 1, \text{ or } 2$

k can be chosen such that

$$-\pi < N \leq \pi$$

For an Earth-in crossing, N must be positive, and for an Earth-out crossing, N must be negative. If the sign of N is incorrect, the solution is rejected. Otherwise, the third spherical triangle is solved for n and E . Again, using the law of cosines

$$\cos n = \cos N \sin\gamma \sin\rho + \cos\gamma \cos\rho$$

If

$$\text{abs}(\cos n) > 1$$

no solution is possible. Otherwise, the nadir angle is computed from

$$n = \arccos(\cos n)$$

and the dihedral angle E is computed from

$$\cos E = (\cos \rho - \cos \gamma \cos n)/\sin \gamma \sin n$$

If

$$\sin \gamma \sin n = 0 \quad \text{or} \quad \text{abs}(\cos E) > 1$$

no solution is possible. Otherwise

$$E = \arccos(\cos E)$$

If this is an Earth-out crossing, E must be negative. Therefore, E is set equal to $-E$. The dihedral angle

$$D = A + E$$

is the meaningful angle in the attitude calculation. If $D < 0$, then

$$D = D + 2\pi$$

If $D > 2\pi$, then

$$D = D - 2\pi$$

Now the two possible attitude vectors may be determined, given the nadir angle, Sun angle, and the cone axes. From the first attitude vector obtained, the corresponding dihedral angle, D' , is computed. The dihedral angle D' is defined as the angle formed by the Sun vector, the spin vector, and the Earth vector. If D and D' are both less than π or both greater than π, then the choice of attitude vector is correct. Otherwise, the other choice is correct.

4.6.1.3 Single-Frame Processing

A single input frame consists of a Sun crossing time, an Earth-in time, an Earth-out time, a spin rate, and two processed Sun angles.

If the Sun time or the spin rate is less than or equal to zero, the frame is rejected. Otherwise, each sensor triggering time is processed as follows:

1. If the triggering time is less than or equal to zero, the triggering is rejected.

2. If the Sun angle is less than zero or greater than pi, the triggering is rejected.

3. The spacecraft rotation angle, A, is computed by

   $$A = (\text{triggering time} - \text{Sun time}) \times \text{spin rate}$$

4. The angle is checked against the sensor cutoff angle. If A is less than the cutoff angle or greater than 2π minus the cutoff angle, the triggering is rejected.

For each central body under consideration (Earth, Moon), the following steps are performed:

1. The lighting conditions on the central body are computed from ephemeris data. If the central body is dark or not visible, then this triggering is rejected. If a terminator is visible, the terminator flag for this frame is set. If data, while a terminator is visible, is not to be included in attitude calculations, the triggering is rejected. Otherwise, processing continues as for the sunlit case.

2. As many as two attitudes are computed for this crossing.

3. If no solution is obtained, then there is no possible attitude consistent with the assumption that this triggering resulted from a sunlit horizon crossing on this central body. Therefore, the crossing is rejected.

4. If the central body is fully sunlit, the processing of this crossing is complete. If a terminator is visible, then it must be determined whether this triggering resulted from a terminator crossing.

If a terminator is visible, it must be determined whether the terminator is intersected by the spacecraft sensor scan, i.e., whether a particular triggering of the sensor was in fact a horizon crossing or a terminator crossing. The procedure for this determination is a recursive one. For each attitude computed, the module, through use of the Optical Aspect Data Predictor, determines whether a scan of the central body with this attitude would have produced a sunlit horizon crossing for the in- or out-triggering. If the computed attitude is not consistent with the assumption that this triggering occurred at a sunlit horizon crossing, then this attitude is rejected.

Note that it is possible that this test will fail to reject a terminator crossing. This occurs when the attitude computed from a terminator crossing is so far from the true attitude that a scan with the erroneous attitude would give a sunlit horizon crossing at this triggering. When this occurs, there is one attitude consistent with the assumption that this is a terminator crossing and a second attitude consistent with the assumption that the triggering was a sunlit horizon crossing. Therefore, there is no deterministic procedure for recognizing this problem. However, when the problem occurs, the resulting computed attitude will in general have a large error. Since the error is large, the erroneous attitude is easily recognized and rejected in the block averaging procedure.

4.6.1.4 Block Averaging Procedure

Once each input telemetry frame has been processed, the best estimate of the attitude must be computed based on the single-frame results. Each input telemetry frame results in two output measurement frames, each of which may contain zero, one, or two attitudes. The ambiguities which could not be resolved on a single-frame basis can now be eliminated if the block of data is sufficiently large.

In computing the average attitude for a block, several cases must be considered. For each output frame with two attitudes, the attitude to be included in the average must be identified. The procedure is further complicated by the fact that in some frames both attitudes must be rejected as spurious. It should be noted that attitudes are considered independent of the source of the horizon triggering from which they were calculated; i.e., attitudes calculated from Moon crossings are given equal weight with those from Earth crossings.

The following procedure is used in the most general case, in which every output frame contains two attitudes and no a priori attitude is available: the first attitude from the first output frame is selected as a trial attitude. In each following output frame, the attitude from the pair that is closer to the trial attitude is selected. The average of the selected attitudes is computed and a residual edit procedure performed to compute the final standard deviation.

The trial attitude obtained by this procedure is further refined by the following iterative technique: the trial attitude is used to select one attitude from each pair and an average of the selected attitudes is computed as before. The average attitude is used as a new trial attitude and the sequence is repeated. The process terminates when the set of attitude selections remains identical for two successive iterations. Convergence normally occurs in two to three iterations.

If an accurate a priori attitude is available, the procedure is greatly simplified. The a priori attitude is used as a trial attitude and the iterative procedure is used immediately.

4.6.1.5 Differential Correction

To further refine the attitude calculations, the OA module has included an option for a least-squares solution for the attitude. This differential correction method utilizes subroutine GCONES with three types of attitude data: Sun angles, nadir angles, and dihedral angles.

After data ambiguities have been resolved and spurious values have been eliminated in the single-frame processing, the input arrays with the appropriate telemetry data are initialized for the call to GCONES. For each input telemetry frame with a valid Sun angle and Sun time, a smoothed value of the Sun angle and the Sun vector at the Sun time are passed to GCONES. Each Sun angle passed is assigned a unit weight, unless the user specifies a different weighting factor for Sun angles. Following the single-frame processing, the selected nadir angle, along with the central-body vector at the sensor triggering time, is passed to GCONES. Likewise, the selected dihedral angle is passed to GCONES, along with the central-body vector and the Sun vector at the triggering time. All nadir and dihedral angles are assigned a unit weight, unless the user specifies a different weighting factor.

Note that GCONES can be used to determine attitude from Sun angle data alone, assuming that Sun data is available over a sufficiently long period and that the attitude remains constant. This will be the system failure mode in the event of horizon sensor failure.

In addition , GCONES executes a residual edit of the data while processing the differential correction. With this technique, spurious values can be eliminated from the least-squares calculations.

4.6.1.6 Interactive Graphics Technique

The interactive graphics capabilities of the attitude system proved to be very valuable during the attitude determination support of the IMP-H mission. As discussed in the IMP-H Post-Launch Report (Reference 2), the Telemetry Data Plot display was used in the interactive mode to determine the biases in the spacecraft OA system. The system biases and the attitude are simultaneously determined by varying the bias parameters and attitude until the calculated data agrees with the telemetry data. By doing this at two or more attitudes, the ambiguities that exist at one attitude can be resolved. The same biases will not cause good fits at more than one attitude unless they are the correct biases.

4.6.2 Module Structure

The optical aspect module is called by the executive of the Attitude Determination Subsystem and can be entered through one of four entry points. The function of each entry is as follows:

1. OPMAIN (IFLAG, IERR)--Read NAMELIST parameters. If IFLAG. LE. 0, return. Otherwise, continue.

2. EPHEM1 (IFLAG, IERR)--Initialize ephemeris routines. If IFLAG. LE. 4, return. Otherwise, continue.

3. QUICKL (IFLAG, IERR)--Perform quick-look attitude determination on requested block(s). If IFLAG. LE. 3, return. Otherwise, continue.

   NOTE: A quick-look can consist of one or more blocks with any number of frames .GE. 1. Program flow is the same regardless of block size.

4. DIFFER (IFLAG, IERR)--Perform differential correction on requested block(s).

As indicated in Figure 4-3, program flow always proceeds from one entry point to the next, in order. Therefore, the choice of an entry point determines the point in the program when execution will begin, and the choice of IFLAG determines the point when the system will return to the monitor.

Input for the OA module consists of the following:

- Card input control parameters and options input through the FORTRAN NAMELIST option
- Spacecraft ephemeris tape--an EPHEM tape containing the orbital position of the satellite
- Lunar ephemeris tape--a tape describing the position of the Moon during the time interval being considered

If the spacecraft and/or lunar ephemeris tapes are not available, orbital elements can be substituted.

Outputs from the module consist of printouts of input data and attitude results from the attitude determination calculation. The output is controlled by an indicator allowing for different levels of output to be printed.

SECTION 5 - MSAP/IMP-J SUBSYSTEM

5.1 SYSTEM OVERVIEW

Attitude prediction and control is provided by the MSAP/IMP-J subsystem. It is assumed for purposes of prediction that the spin axis vector is coincident with the satellite angular momentum vector, which changes according to the total disturbance torque acting on the satellite body frame. Motion caused by magnetic, gravity-gradient, solar-radiation pressure, and aerodynamic perturbations can be considered; however, because the torques on IMP-J will be negligible for nominal orbit constraints, these torques may be deactivated for prediction purposes, permitting speedier control calculations.

Any control recommendations made by MSAP/IMP-J for reorientation maneuvers are basically formulated on the principle of minimal gas usage. Options are available for specific attitude reorientations or to be instantaneous perigee velocity vector. Commands are also generated for spin control.

The task of calculating the reorientation control and spin control is achieved by using a simulation of the gas torquing system.

5.2 SYSTEM STRUCTURE

Data flow for the MSAP/IMP-J subsystem is shown in Figure 5-1. The initial attitude point used in the prediction may be output from the IMPADS system or from a previous MSAP run.

Torque computations can be eliminated for perturbations that are determined to be insignificant for the prediction needed. If a prediction is not desired at all, the control module can be run alone. Contingencies are also provided for certain satellite hardware malfunctions that may cause inaccuracies in prediction and control.

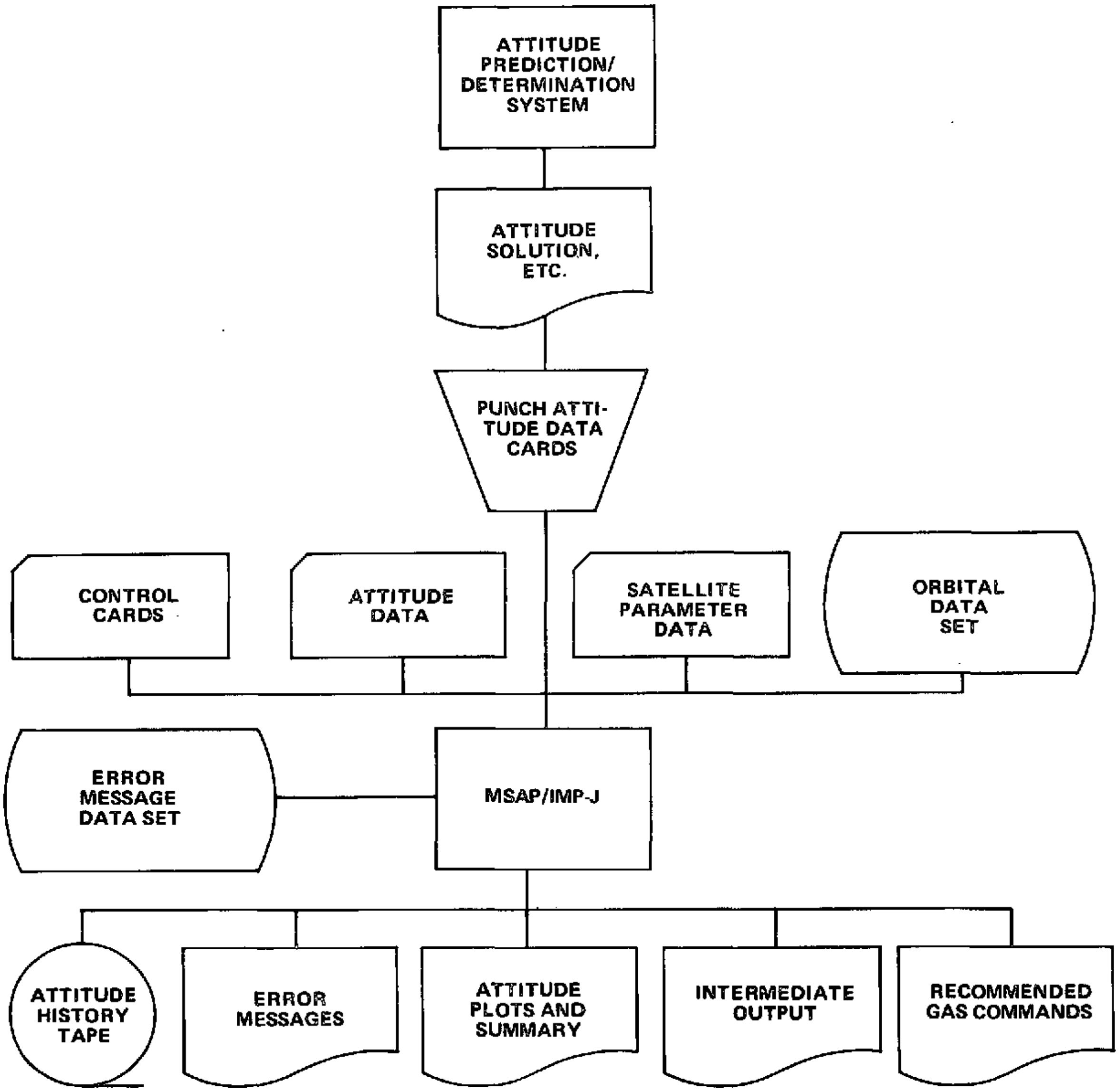


Figure 5-1. MSAP/IMP-J Data Flow

The basic prediction algorithm updates the spin axis vector at specified points in time over the simulation period. When control recommendations are needed, program control is passed from the predictor module to the control module. At user option, the recommendations generated can be applied by the predictor upon return of program control or printed out for user consideration only.

5.2.1 MSAP/IMP-J Input

All input data is read from four FORTRAN NAMELISTs. Parameters that are characteristic of the general (i.e., satellite independent part of) MSAP system comprise a separate NAMELIST, MSAPIN. These include the simulation time; the initial attitude parameters, right ascension, declination, and spin rate; data set reference numbers used for input and output purposes; time parameters from which both program efficiency and predictor accuracy can be controlled; and the orbital elements required by the orbit generator.

IMP-J related parameters, specifically the satellite physical data, initial state parameters, and control designators for the IMP-J control subsystem, are contained in the IMPIN NAMELIST.

Time-repetitive data is input under the HISTRY NAMELIST. IMPIN contains parameters that remain constant throughout a prediction, and HISTRY contains parameters such as moments of inertia, spin rates, gas state parameters, commands, attitudes, and other parameters that may vary during a single run. More than one HISTRY NAMELIST can exist for one set of input data, making it easier for the user to simulate over consecutive blocks of time.

Data for each control command input through the CNTRLN NAMELIST consists of the start time, the number of executions per individual command time, the time increment between command times, the number of commands, and the final attitude desired. Individual reorientations may also be input by specifying actual latitude and longitude commands used. This permits the user to describe fully any command needed or any historical command that may be used in analysis of the system.

5.2.2 MSAP/IMP-J Output

System output consists of a formatted listing of input data, an attitude parameter output, control data output, intermediate data output, error analysis of predicted maneuvers versus those observed, and error messages. The attitude parameters, in the form of listings and plots, and the input data comprise the standard output unit. All other data is printed out on option according to user requirements.

Besides the standard attitude parameters (right ascension, declination, and spin rate) two other parameters, zeta and gamma, are included in the summaries and plots. Zeta is the angle between the current attitude and the desired attitude to be reached by a set of commands, and gamma is the Sun angle.

Detailed accounts of control in terms of intermediate attitude and gas state parameters after each command or gas pulse can be printed at option. When less than a full command is necessary, appropriate information concerning the fractional command is included. At user option, a magnetic tape can be generated that summarizes an attitude prediction at designated intervals for use by the optical aspect data prediction program. The tape is headed by a satellite identification, a start time for the prediction, and an increment time between attitude points. The spin axis attitude is given in the form of direction cosines in geocentric inertial coordinates.

The user can view intermediate data from any part of the system by specifying intermediate output. Various levels of output provide the user with the desired details needed in the analysis.

5.3 MSAP/IMP-J SUBSYSTEM RESOURCES

MSAP/IMP-J is a complex subsystem made up of several modules and data sets. To operate the subsystem, it is necessary to combine source modules from either source decks or from libraries that reside on the ATTDET disk file on the IBM S/360-95 at Goddard Space Flight Center. Below is a summary of the major resources and input data needed to operate the system.

1. Hardware required for the current implementation of MSAP/IMP-J consists of
   a. Card reader
   b. One nine-track tape drive (if an orbit tape is being used)
   c. Disk pack (ATTDET)
   d. Printer
2. Data sets and libraries required by MSAP/IMP-J are
   a. Cards--job control language and NAMELIST input
   b. MSAP library data sets
      (1) MSAP.ERROR.MESSAGES--error message data set
      (2) MSAP.TABLE.HRnn--atmospheric density tables

Approximately 350 K of core is needed for the MSAP/IMP-J subsystem.

SECTION 6 - OPERATIONS PLAN

6.1 ATTITUDE SUPPORT PLAN

The Attitude Determination and Control Section (ADCS) is responsible for the determination and control of the spacecraft attitude for the first month following the ecliptic orientation maneuver. Due to hardware and experimenter requirements, the following attitude constraints have been specified for the IMP-J satellite:

1. The spin axis-Sun angle must be maintained at 90 ± 25 degrees during the transfer orbit.
2. The spin axis must be oriented normal to the ecliptic plane during the mission orbit.
3. The transfer orbital lifetime must be at least one year.

To achieve the mission support requirements, an attitude software support system has been developed. This system is designed to operate in a near-real-time environment and to achieve an accuracy of two degrees in the calculation of spacecraft attitude.

The attitude support system will require input from both the Orbital Computations Engineer (OCE) and the Mission Analyst (MA). The OCE will provide the Attitude Computations Analyst (ACA) with a spacecraft orbit ephemeris based on the latest computed orbit a few hours after injection into the transfer orbit. Prior to the first apogee of the transfer orbit, the MA will provide the ACA with the desired attitude for firing the kick motor to achieve the desired mission orbit.

6.2 ATTITUDE CONTROL AND RELATED ATTITUDE PROCESSING

To satisfy the attitude constraints imposed by the mission requirements, the IMP-J satellite has been equipped with an optical aspect (OA) attitude determination system and an attitude and spin control system. During the early stage of the mission, the Quick-Look Utility Data Program (refer to Section 8.3 for description) will be used in conjunction with the wallboard to monitor the Sun angle and spin rate values. Fifteen minutes of data will be received every four hours for this purpose. Twenty hours before kick motor firing, OA data will be received if the spacecraft is in the nominal orientation. The IMPADS subsystem will be employed to determine attitude for about the first three hours, then the OABIAS system (Section 8.4) will be employed for about 1 hour in order to determine biases. The MSAP/IMP-J subsystem will be used to determine the maneuver to the intermediate attitude. The maneuver will require about 1.5 hours to execute. The intermediate attitude will then be verified with the IMPADS subsystem for 2 hours followed by the OABIAS system for 1 hour. MSAP/IMP-J will again be employed to compute the final maneuver. OA data will be received for 8 hours at the kick motor firing attitude. During this time IMPADS will be used as well as OABIAS to refine the attitude and bias determination.

After the kick motor firing, the spacecraft will be maneuvered to receive OA data. At this attitude only, the IMPADS system will be used to determine attitude. At the end of the OA data availability, the spin axis will be precessed normal to the SEP. Data will be received and processed about 26 hours after the kick motor firing. Data will not be received again for 12 days.

In support of the IMP-J mission, the following items will be generated by the Attitude Support System:

1. Attitude results. Attitude solutions will be generated by the Attitude Determination Subsystem through the processing of the telemetry data transmitted from MSOCC via a direct data link.

2. <u>Attitude control recommendations</u>. The recommended control commands generated by the MSAP/IMP-J Subsystem will be supplied to MSOCC.

3. <u>OA data availability with recommended data transmission schedule</u>. A summary will be generated listing the times when useful OA data should be available from the onboard attitude system.

4. <u>Aspect angle predictions</u>. During the transfer orbit and the first mission orbit, the predicted aspect angles (spin axis-Earth angle) will be supplied to MSOCC.

SECTION 7 - DATA HANDLING PROCEDURES AND FORMATS

7.1 DATA PROCESSING CAPABILITIES

The following capabilities are available or can be selected in performing the control of the IMP-J satellite:

- Input unit device for the telemetry data
- Decision for graphics usage
- Type of attitude determination when graphics is not used
- Angles for initial estimate of attitude
- Interval of data to be processed
- Time adjustment parameters
- Time rejection parameters
- Decision for optional optical aspect data file
- Conversion factors
- Date of run
- Prediction and Control

7.2 TELEMETRY DATA FORMATS

7.2.1 Satellite-Dependent Disk File

The satellite-dependent disk file, provided by the Telemetry Data Handling Subsystem, contains an individual record for each sample of data. Each record has two distinct segments: the header portion, which is written in XDS 930 BCD format, and the attitude data, which is in bit-string format. All records are unblocked and are zero-filled to give them a fixed length of 312 bytes. An end-of-file mark is always inserted at the end of the data. No control words are used on this disk file.

7.2.1.1 Record Format

Byte	Label	Description
1-2	01110--0 (binary representation)	Spacecraft ID
3	00000XXX (binary representation)	Data type indicator
4-6	XXXXXXXX (octal representation)	Pass record number
7-12	SSSSSS ƀƀ (XDS BCD)	Station name
13-15	XXXXXXXX (octal representation)	Orbit number
16-21	YY/MM/DD (XDS BCD)	Date of pass
22-24	HHMM (XDS BCD)	Time of pass
25-56	Attitude Data	See Section 7.2.1.1.1
57-312	Zero fill	

7.2.1.1.1 Attitude Data Format (Bytes 25-56 of Each Data Record)

Byte	Label	Description
25-29	Time (36 bits)	GMT in milliseconds of year
30-33	Quality bytes	One byte for each snapshot
34-35	Spin rate	Telemetry counts
36-37	Sun time	Telemetry counts
38-39	Earth time	Telemetry counts
40-41	Earth width	Telemetry counts
42-43	Solar angle	Gray code
44	Gas pressure high	Telemetry counts
45	Gas pressure low	Telemetry counts
46	EFM1 ANT + X	Telemetry counts
47	EFM2 ANT - X	Telemetry counts
48	EFM3 ANT + Y	Telemetry counts
49	EFM4 ANT - Y	Telemetry counts
50-51	Zero fill	

Byte	Label	Description
52	ACS tank temperature	Telemetry counts
53	00000000	Zero fill
54-56	Spacecraft clock	Telemetry counts

7.2.2 XDS 930 Backup Tape (Binary Representation)

The data on this 7-track XDS 930 backup tape is in identical format to that of the satellite-dependent disk file. However, at the end of each pass a single end of file is written on the tape. Additional data for other chronological passes may be stacked on this tape with four end of files inserted behind the last record of the last pass on the tape. See Section 7.2.1.1 for details on the record format.

7.2.3 Backup Attitude Data Cards

Attitude data cards are punched using the XDS 930 printer listings when the backup mode of operation is required. All data is punched in engineering units as found on the listings. Two cards can be punched for each sample, with attitude data on card 1 and control data on card 2. The card code, in columns 1 through 4, is the spacecraft ID; it is checked on each individual card. A subcode is punched in column 6 as a check that the two cards per record have not been accidentally separated. No header type record is included.

CARD 1--ATTITUDE DATA

Card Column	Format	Label
1-4	0111	Spacecraft ID--constant identifier
6	1	Card subcode--constant identifier
8-9	I2	Year - 1900 of sample
10-12	I3	Day of year of sample
13-14	I2	Hour of day of sample
15-16	I2	Minute of hour of sample

Card Column	Format	Label
17-18	I2	Second of hour of sample
19-21	I3	Milliseconds of second of sample
22-27	F6.2	Sun angle
29-34	F6.1	Sun time
36-41	F6.1	Earth time
43-48	F6.3	Spin rate
50-55	F6.1	Earth width
80	I1	Transmission rate

CARD 2--CONTROL DATA

Card Column	Format	Label
1-4	0111	Spacecraft ID--constant identifier
6	2	Card subcode--constant identifer
8-14	F7.2	Gas pressure high
16-22	F7.2	Gas pressure low
24-29	F6.2	Gas temperature
31-36	F6.2	EFM1 ANT + X Length Deployment
38-43	F6.2	EFM2 ANT - X Length Deployment
45-50	F6.2	EFM3 ANT + Y Length Deployment
52-57	F6.2	EFM4 ANT - Y Length Deployment
59-64	F6.2	EFM5 ANT + Z Length Deployment
66-71	F6.2	EFM6 ANT - Z Length Deployment

7.3 IMP-J ATTITUDE FILE

An optional output feature of the IMPADS subsystem is the attitude file. When an acceptable solution has been found, the operator goes to the graphic Display Controller and presses key 16. The most recent solution is placed in the attitude file in the following format.

Word	Type	Description
1	R*8	Orbit level
2	R*8	Year and day of year (e.g., 1972.121)
3	R*8	Millisecond of day
4	R*8	Right ascension of the spin axis (degrees)
5	R*8	Declination of the spin axis (degrees)
6	R*8	3 sigma for alpha (degrees)
7	R*8	3 sigma for delta (degrees)
8	R*8	Half angle error (degrees)
9	R*8	Spin rate (rpm)
10-35	R*8	Fill (9999999)

SECTION 8 - SYSTEM SUPPORT PROGRAMS

8.1 INTRODUCTION

Utility programs have been developed to aid in the support of the operational requirements for attitude determination, prediction, and control. These programs are briefly described in the following subsections.

8.2 OPTICAL ASPECT DATA PREDICTION PROGRAM

The Optical Aspect Data Prediction Program (Reference 3) predicts the time interval during which the Optical Aspect System onboard the satellite will provide useful data and predicts the quality of that data.

The program accepts as input an attitude tape produced by the MSAP Subsystem. This tape contains a set of records, each giving the components of the spin vector and the time. Additional input consists of ephemeris tapes that gives the spacecraft ephemeris and lunar ephemeris during the period of interest. Control parameters are read from cards.

For each input record, the program produces printed output describing the availability of solar aspect, lunar aspect, and Earth aspect data at the specified time.

Various options are available to alter the specifications given above; e.g., ephemerides may be generated internally from the orbital elements; the Moon may be ignored; a fixed attitude may be read from cards so that no attitude tape is required; and printout may be suppressed, depending on the optical aspect data conditions at each point.

8.3 QUICK-LOOK UTILITY PROGRAM

The functions of the Quick-Look Utility Program are to assist in the analysis of raw telemetry data which is input to the Attitude Determination Subsystem and to archive specified telemetry records on an independent data set for use later. The program can be operated in either a noninteractive or an interactive graphics mode, the latter utilizing an IBM 2260 Display Unit.

In support of the analysis of telemetry data, the Quick-Look Program prints out a hexadecimal and/or binary representation of all records contained within a specified time or record interval. In the interactive graphics mode, each record is analyzed for the availability of OA data and the results of this analysis are displayed on an IBM 2260 Display Unit. In addition, in the interactive mode, a hexadecimal display of any record is available for viewing. The purpose of both the displays and the program output is to assist the user in determining the times and quality of the telemetry data used in attitude processing.

The archiving of telemetry records is accomplished by copying specific records from an input telemetry data set to an output data set that contains records with useful attitude data. In the interactive graphics mode, the user can archive records from a 2260 display and therefore be assured of the quality of the attitude data.

Input to the program consists of a telemetry data set either from disk or tape. The latter can contain multiple files. The control parameters are specified through the FORTRAN NAMELIST feature. Output consists of hard-copy printout and, optionally, a tape or disk data set containing archived records.

8.4 TELEMETRY DATA SIMULATOR

The purpose of the Telemetry Data Simulator is to create a data set containing valid IMP telemetry data. The simulator utilizes the optical aspect simulator to generate attitude data, which is then converted into telemetry counts and inserted into simulated telemetry records. In addition, a data modification subroutine exists that generates degraded telemetry data for use during system testing. The output of the program consists of a tape or disk data set of telemetry records in a format that can be input to the Attitude Determination Subsystem.

8.5 2260 ATTACH PROGRAMS

Backup attitude determination, prediction, and control programs have been developed for the IMP-J mission. They are interactive graphics programs designed to provide quick estimates of optical aspect data availability times, spacecraft attitude, and control commands required for attitude maneuvers. These programs use an IBM 2260 Display Unit for reading input control parameters and for displaying results. The backup programs run under the attach mode of the Graphic Terminal System (GTS). By running under the attach mode of GTS, these programs execute and provide results immediately.

8.6 OPTICAL ASPECT BIAS DETERMINATION SYSTEM

The Optical Aspect Bias Determination System (OABIAS) (Reference 4) was developed to help improve the attitudes determined with optical aspect data by determining sensor biases. The biases which are modeled by OABIAS include

- Sun angle
- Sun sensor tilted with respect to the spacecraft z axis
- Horizon sensor mounting angle (colatitude)
- Horizon sensor azimuth relative to the Sun sensor
- Horizon sensor threshold
- Panoramic Attitude Scanner plane tilted with respect to the spacecraft z axis (related to RAE-B)
- Spacecraft position in the orbit track

In addition to these biases, OABIAS also determines attitude and spin rate. OABIAS was used successfully to support the launch of the Radio Astronomy Explorer-B (RAE-B) spacecraft (Reference 5). Since that support, the system was modified to model in-track orbit biases. This was found to be necessary in the OABIAS support of the Small Scientific Satellite-1.

The basic approach of OABIAS rests on the concept that, since there are usually many more different observations than there are unknowns, the solution for the unknowns is amenable to least-squares techniques. The least-squares technique implemented in OABIAS is a recursive one. That is, the estimate of the biases is updated after processing each observation rather than processing a batch of observations, updating the estimate at the end, and then iterating. The recursive approach was chosen because: (1) it requires less computer time since it does not require the inversion of large matrices; (2) it is ideally suited to real-time operation because of the manner of updating the state vector; and (3) it is more efficient than batch mode in the solution of highly nonlinear problems.

The fundamental effort in the application of least-squares is in correctly modeling the observables. In the case of optical aspect data, the observables consist of

- Sun angle
- Sun sighting time
- Horizon sighting time (sky to central body)
- Horizon sighting time (central body to sky)

OABIAS implements eight models of these observables and combinations of them. The models are

1. Sun angle
2. Sun sighting time
3. Horizon sensor line of sight projected on the nadir vector
4. Horizon crossing time
5. Dihedral angle between the Sun and the horizon with the spin axis as the pivot
6. Earth width

7. Nadir angle (assumes a point central body)

8. Dihedral angle between the Sun and the center of the Earth with the spin axis as the pivot

Because the IMP-J telemetry does not include the Sun sighting time, models 2, 3, and 4 cannot be applied. Because the Earth will be about 3 degrees wide at acquisition of OA data, model 7 cannot be applied. The presence of a terminator on the Earth would interfere with the application of models 6 and 8. However, if the terminator is sufficiently near the horizon (nearly full Earth), the effect would be minimized. Although the size of the Earth as seen from the IMP-J orbit is large enough to eliminate model 7, it is small enough to cause computational difficulties in model 5. The overall effect of these constraints is being assessed through extensive prelaunch simulations.

The OABIAS program executes under the control of the Multi-Satellite Attitude Determination (MSAD) System. It receives the optical aspect data from IMPASS via an intermediate data set. OABIAS parameters are input via the IBM 2250 graphics device. The problem flow is also controlled via graphics. A variety of data plots are used to monitor the convergence characteristics of the recursive estimator. The estimated biases can then be input to the IMPADS Subsystem.

REFERENCES

1. National Aeronautics and Space Administration, X-542-72-334, IMP-H Attitude Control Prelaunch Analysis and Operations Plan, G. D. Repass, T. J. Brown, and H. L. Hooper, August 1972

2. --, X-581-73-214, IMP-H Attitude Determination and Control Report, G. D. Repass, H. L. Hooper, July 1973

3. Computer Sciences Corporation, 9101-13300-02TR, Optical Aspect Data Prediction (ODAP) Program System Description and Operating Guide, M. Joseph, M. A. Shear, October 1972

4. --, 9101-09700-01TN, Preliminary Design of the Optical Aspect Bias Determination System (OBIAS), A. Dennis, M. Plett, July 1972

5. --, 3000-06000-01TM, OABIAS Test Results I, T. Shinohara, September 1973