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THE SPINSAT BUS: New and Old in a Small Package

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The SPINSAT Bus is a three axis stabilized, light weight, inexpensive spacecraft capable of orbit changing (elliptic to circular, co-planar, plus inclination adjustment), orbit maintenance (altitude adjustment to assure ground trace repeatability) and, in final operational orbit, accurate earth pointing (± 0.5 degrees). The Bus also provides thermal control, telecommunication, data storage and electrical power for modest sized payloads to perform a variety of low-earth orbit (LEO) missions of up to three years duration.

The baseline Bus version is a cylindrical shape (914.4 mm [36 inch] diameter by 902.2 mm [35.52 inch] height). The outer surface is covered with solar cells and for this mission, four extendable panels covered with solar cells supply 115 watts orbit average under the worst case (high noon) orbit. The baseline design is compatible with Scout or Pegasus launch vehicles. For the first SPINSAT mission, the maximum launch weight (on Scout) is 166.5 kg (367.1 lbs) including an altimeter payload of 43.1 kg (1.95 lbs).

This paper will define the mission analysis that drove the design, the mission sequence of events and give a brief description of the Bus systems. The present launch target date is March 29, 1990 from Vandenberg, launch complex SLC 5.

THE SPINSAT BUS

I. INTRODUCTION

The Office of Naval Research (ONR) has selected a modified "T-SAT" design satellite bus from the Intraspace Corporation of North Salt Lake City, Utah, for use in their Special Purpose, Inexpensive Satellite (SPINSAT) Program.

The "T-SAT" is a new type of satellite featuring the old concepts of simplicity and affordability to support both operational missions and experimental programs.

The initial mission is to carry an altimeter, designed and made by the Applied Physics Laboratory of Johns Hopkins University, into an orbit that exactly duplicates many of the parameters of, but is not co-planar with, the GEOSAT orbit.

The Scout launch vehicle, using the 1.067 m (42 inch) payload fairing, could put a maximum payload of 116.2 kg (256 lbs) into the desired 800 km (432 nmi) orbit at 108 degrees with an inclination accuracy of 0.34 degrees (1 sigma). To increase the useful weight on orbit, the old principle of staging and an elliptical parking orbit were chosen with the SPINSAT Bus containing on-board propulsion (a restartable "kick" motor). The American Rocket Company (AMROC) is supplying a hybrid rocket motor to raise the Bus to the desired altitude and provide for inclination correction. Hybrid motors are not new (solid fuel with a gaseous or liquid oxidizer) but have not been used in such a small thrust size (nominally 120 lbs of force). Thrust vector control will be provided by a rapid response gimbal system with the entire motor, rather than just the nozzle, being gimballed (both old and new).

Once in final orbit, the altimeter will be earth pointing with the Bus being three axis stabilized to ± 0.5 degrees by a bias momentum system similar to the TDRSS Attitude Control System. The orbit will be maintained (altitude) so that the ground trace of the orbit line of nodes will be within ± 1 km (± 0.63 nmi) of the Exact Repeat Mission (ERM) by use of high pressure cold gas thrusters from the Reaction Control System (RCS). The thrusters will have a thrust force of only 0.02 lbs each. Cold gas thrusters were used extensively for torquing during the early days of satellite design but seldom for propulsion. Monotonic torques (particularly gravity gradient and aerodynamic) in pitch will be integrated by the momentum wheels into changes in angular momentum (wheel loading). Unloading the wheels will be accomplished by using magnetic torque rods (an old design principle) interacting with the earth's magnetic field. A magnetometer will be used to determine polarity for the torque rods.

II. MISSION ANALYSIS

A. FINAL ORBIT ATTAINMENT

Parking Orbit -- Trajectory simulation by LTV (the Scout manufacturer) indicated that 166.5 kg (367.1 lbs) could be placed in a nominal 200 x 600 km (108 x 324 nmi) parking orbit. Allowing for 90% iso probability dispersion errors, a worst case orbit of 195 x 502 km (105.3 x 270.5 nmi) was assumed for velocity change (ΔV) analysis. (See Appendix A).

Orbit Change (Altitude Only) - (Appendix A)

Using the circular orbit velocity at 800 km (432 nmi) altitude as a reference (U_n = 7451.85 m/sec [24448.33 ft/sec]) yields the velocity change at perigee (ΔV_p) and apogee (ΔV_a) given in Figure 1.

Orbit Change (Inclination Only)

The orbit parameters will, in general, not permit velocity combining at apogee with inclination changing. Again, 90% (1.65 sigma) iso probability inclination dispersion errors are (from Scout) 0.569 degrees. The velocity change requirements for inclination change are:

$$\frac{\Delta V_{\Delta i}}{U_N} = 2 \sin \frac{\Delta i}{2}$$

where Δi = inclination change for i and U_N as given,
$\Delta V_{\Delta i}$ = 74.00 m/sec (242.79 ft/sec)

Propellant Consumed

$$\frac{Wpc}{Wo} = 1 - e^{\frac{-\Sigma\Delta V}{gIsp}}$$

Wpc = weight of propellant consumed
Wo = weight at start of ΔV
$\Sigma\Delta V$ = sum of velocity changes
g = gravity acceleration
Isp = fuel (propellant) specific impulse

For the AMROC OIM, Isp ≈ 280 sec

For: Wo = 166.5 kg (367.1 lbs)
ΔV = 320.88 m/s (1052.76 ft/s)
Wpc = 18.37 kg (40.49 lbs)

The data on Figure 1 shows a 10% contingency on propellant.

Conclusions

The velocity change and propellant requirements have been defined to attain the final circular orbit from the worst case elliptical parking orbit. In actual use, each velocity change would be subdivided into two or more intermediate orbits to prevent overshoot and propellant waste. Final orbit trim will be performed using cold gas thrusters.

B. LAUNCH CONDITIONS FOR MAXIMUM SUN ORBITS

Figure 2 shows the sun angle to the orbit plane (ß) for a set of launch conditions allowing initial full sun orbits for a period of up to 50 days. Similar results would occur for a summer solstice launch with proper initial conditions.

Figure 3 is an expanded view of the first 30 or 50 days following insertion into the nominal parking orbit.

Superimposed on these figures are the ß angles required for an all sun orbit at the specified orbit altitudes (ß = 62.69° for an 800 km [432 nmi] orbit).

ß max values ocur at i ± e_c where e_c is the inclination of the earth's spin axis from the ecliptic normal (23.45°). Thus for i = 108°, ß max, max 108° ± 23.45° or 84.55° (See Figure 2). For 108° +23.45° = 131.45° (48.55°) ß max, min would occur (See Figure 4). Under these conditions, there would not be a full sun orbit for years.

For launches at the equinoxes, the ß angle would be approximately 72° and several days of all sun orbits could occur.

III. SPINSAT BUS DESIGN

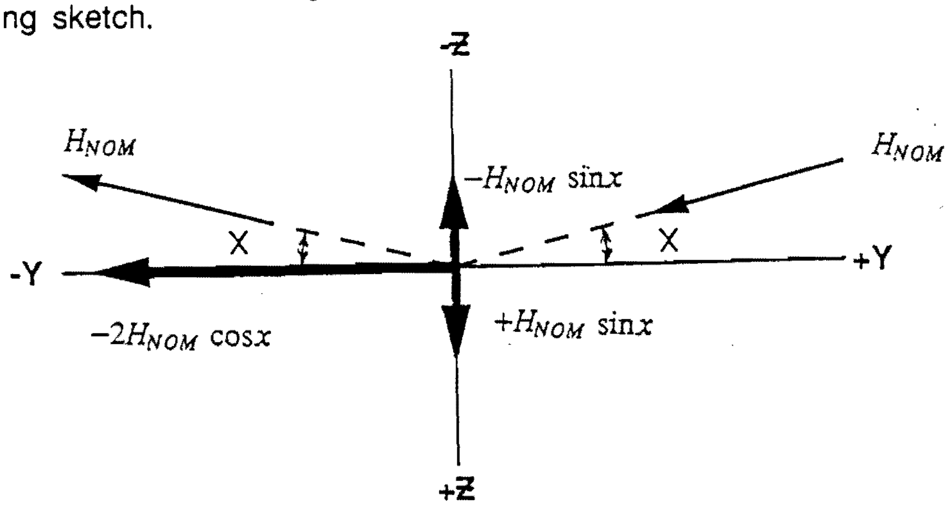
A. OVERVIEW

Figure 5 is an exploded view of the Bus. The altimeter antenna is mounted on the earth-facing end of the Bus (the yaw or + Ƶ axis). The four extendable solar panels are hinged near the + Ƶ face and fold out to 45° with respect to the + Ƶ axis. The solar cells are mounted on the inside of the panels so that near high noon (ß = 0°), they will pick up sunlight.

The OIM engine is mounted on the negative yaw (- Ƶ) face of the Bus.

In final orbit, the positive roll (+ X) axis is aligned with the velocity vector and the positive pitch (+ Y axis) forms a right angular set and is normal to the orbit plane.

The momentum wheels are aligned tilted approximately 14.5° towards the Ƶ axis so that their angular momentum vectors are indicated in the following sketch.



The large momentum component along the -Y axis provides the momentum bias needed. At constant speed on the wheels, the Ƶ components cancel.

B. WEIGHT BUDGET

See Table 1.

IV. SEQUENCE OF EVENTS

See Figure 6.

TABLE 1

WEIGHT BUDGET

	Sub Total Weight (kg)	Sub Total Weight (lb)	Item Weight (kg)	Item Weight (lb)
Structure	25.04	55.20		
Solar Panels (4)			6.53	14.40
OIM	34.20	75.40		
N_2			18.37	40.50
PBO (Fuel)			3.00	6.60
OIM Platform (Gimballed)	.63	1.40		
RCS	6.88	15.16		
N_2			2.09	4.60
C & DH	13.49	29.74		
RF System	10.73	23.66		
Payload	43.09	95.00		
ACS	9.8	21.77		
Thermal Tape, Paint	0.68	1.50		
Wiring Harness	2.27	5.00		
EPS	19.53	43.05		
Batteries (12)			10.34	22.80
Body Solar Cells			3.51	7.73
Panel Solar Cells			3.65	8.05
TOTAL	166.41	366.88		
LAUNCH WEIGHT	166.52	367.10		
MARGIN	0.10	0.22		

SPINSAT ORBIT ESTABLISHMENT FOLLOWING SEPARATION FROM SCOUT

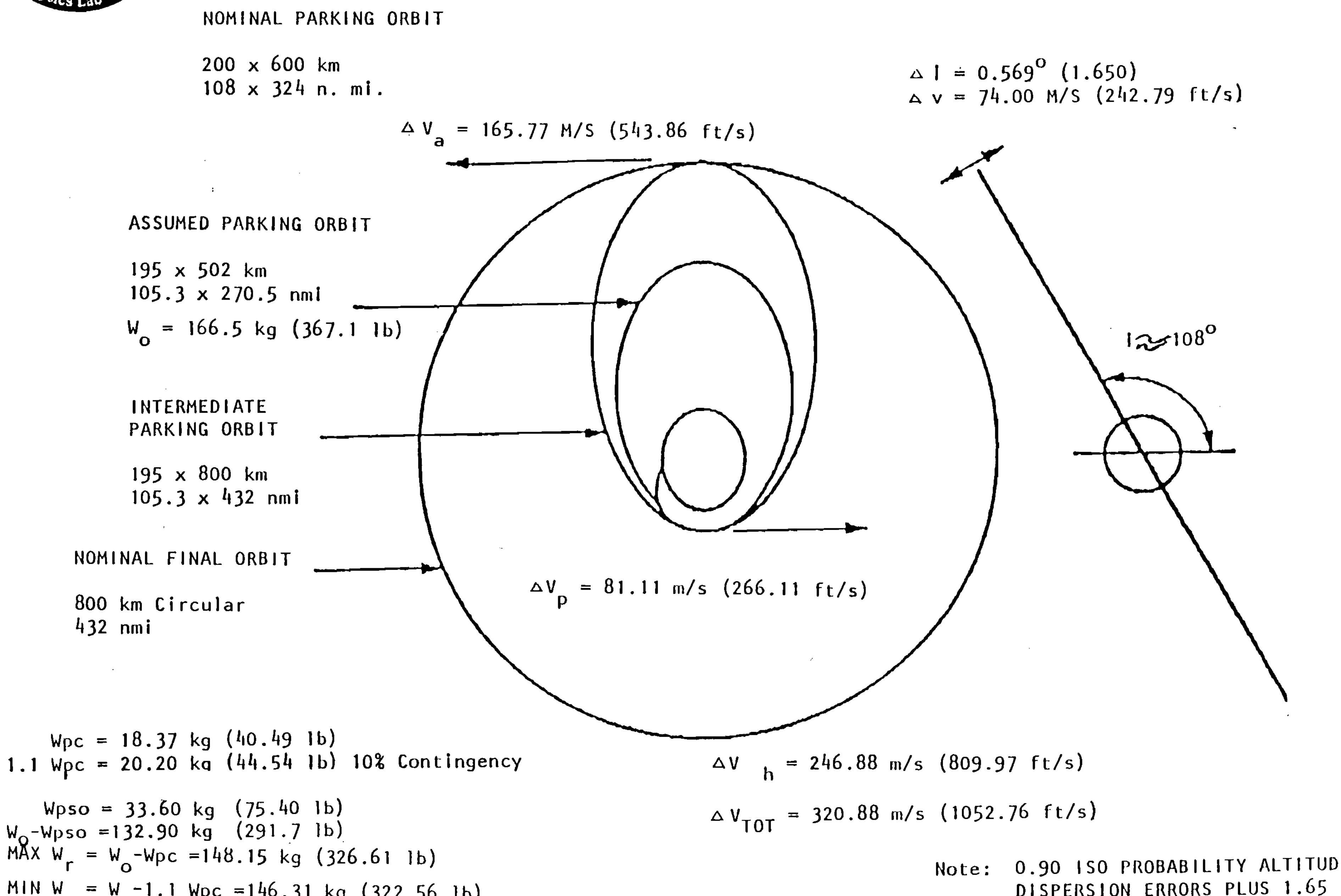


FIGURE 1

SUN ANGLE TO ORBIT PLANE (BETA) VS EARTH ORBIT POSITION (DELTA)

INITIAL FULL SUN ORBITS

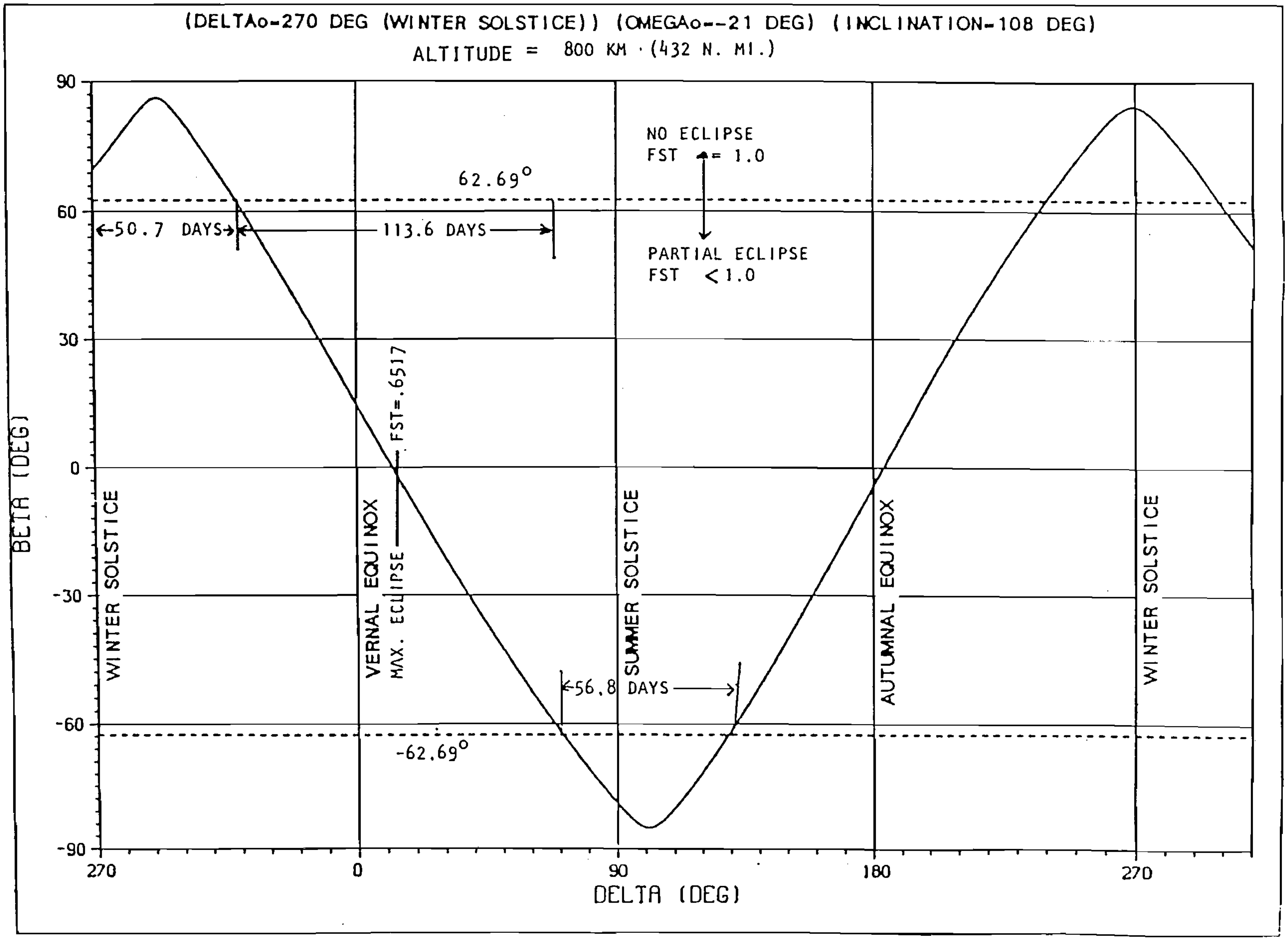


FIGURE 2

SUN ANGLE TO ORBIT PLANE VS EARTH ORBIT POSITION (DELTA)

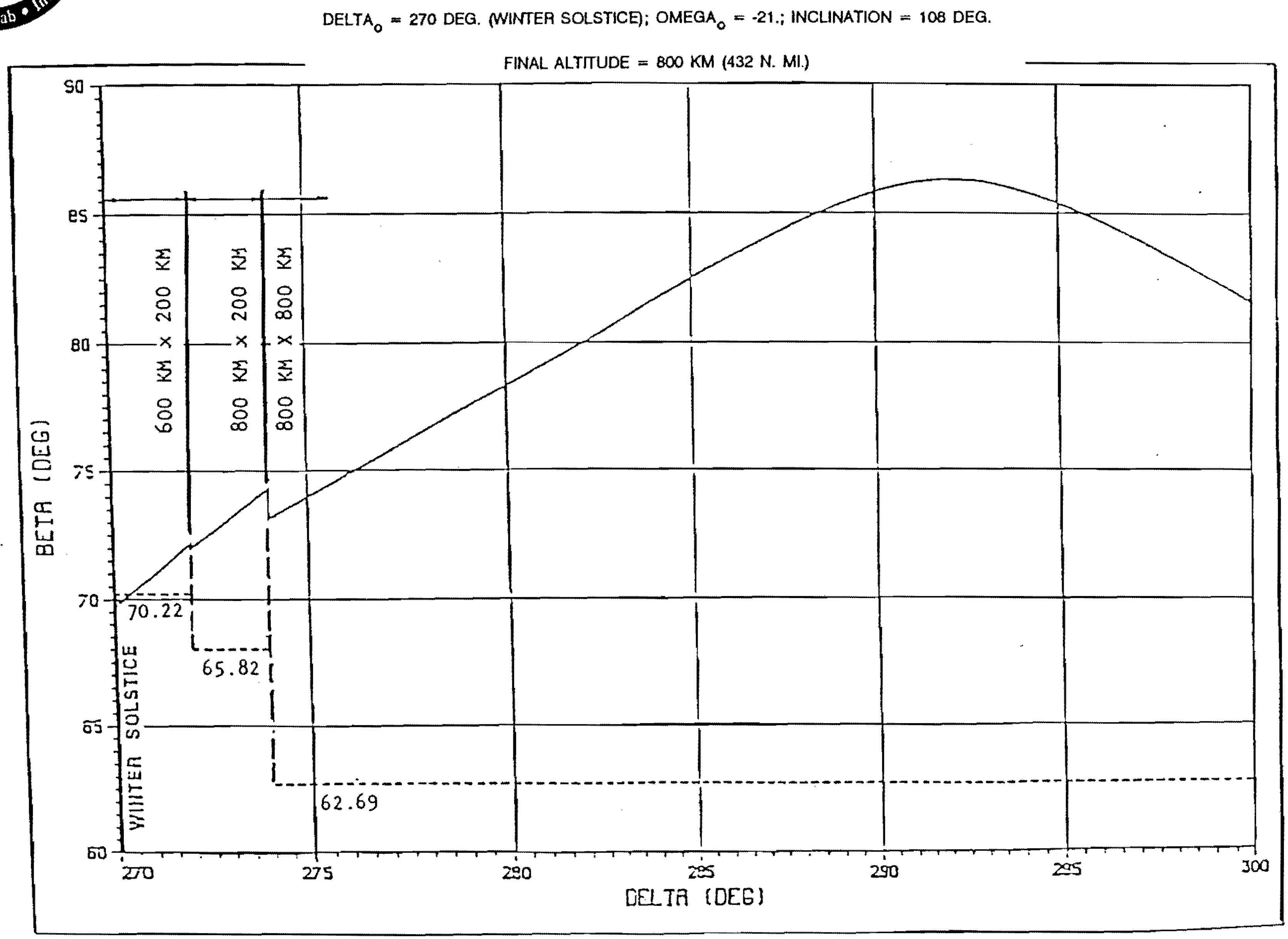


FIGURE 3

SUN ANGLE TO ORBIT PLANE (BETA) VS EARTH ORBIT POSITION (DELTA)

NO FULL SUN ORBITS

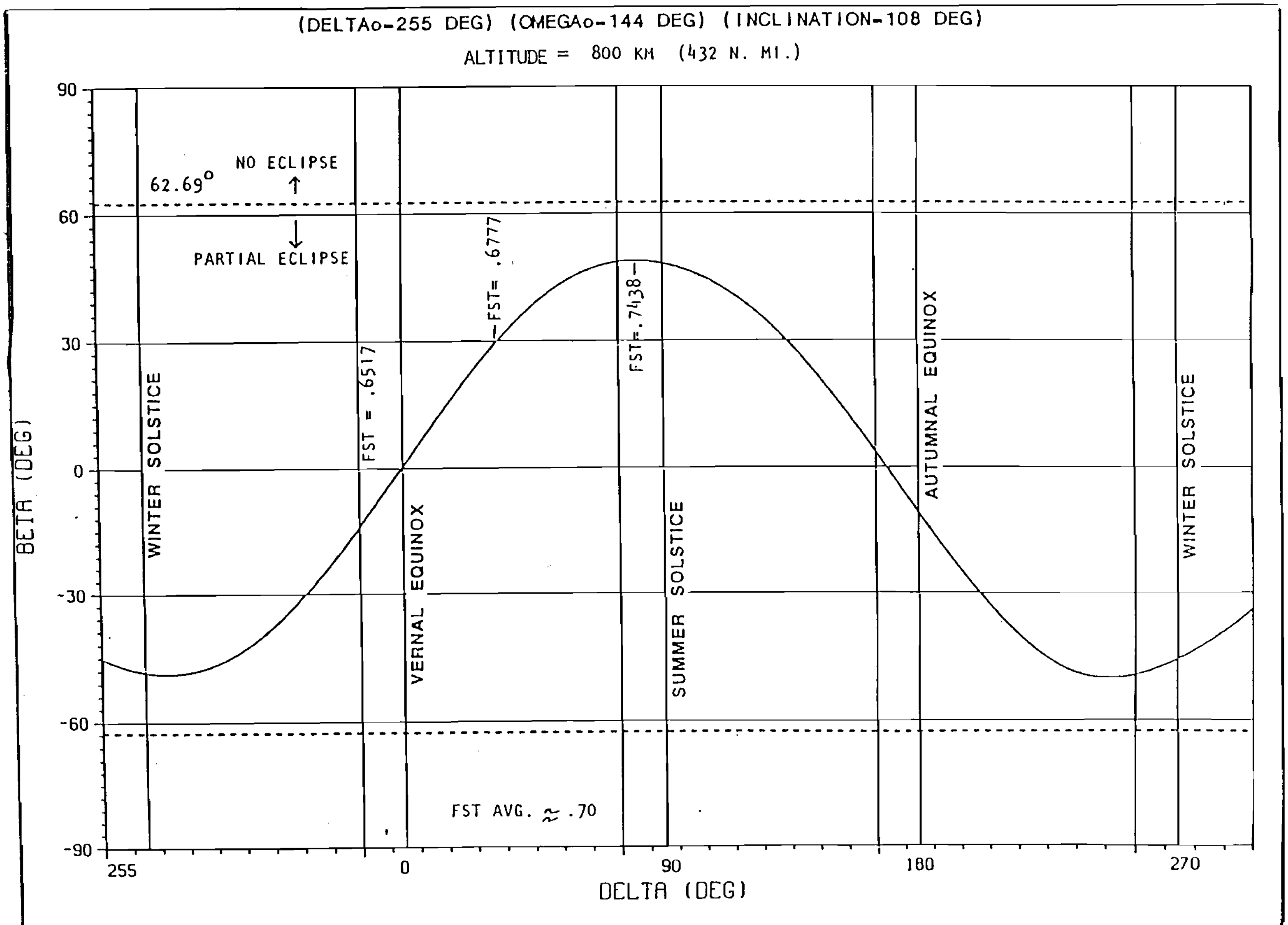


FIGURE 4

SALT SPACECRAFT CONFIGURATION

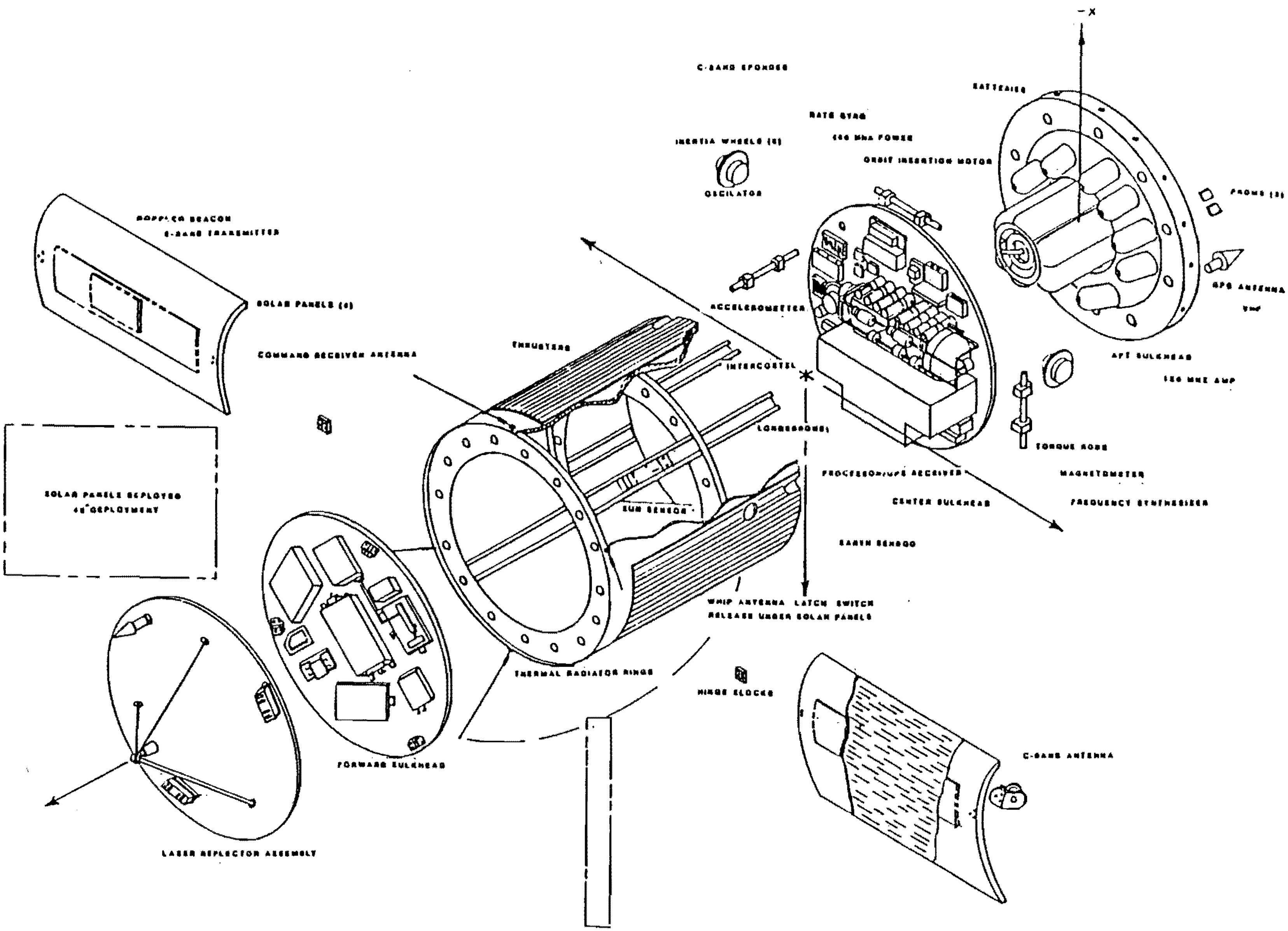


FIGURE 5

SEQUENCE OF EVENTS

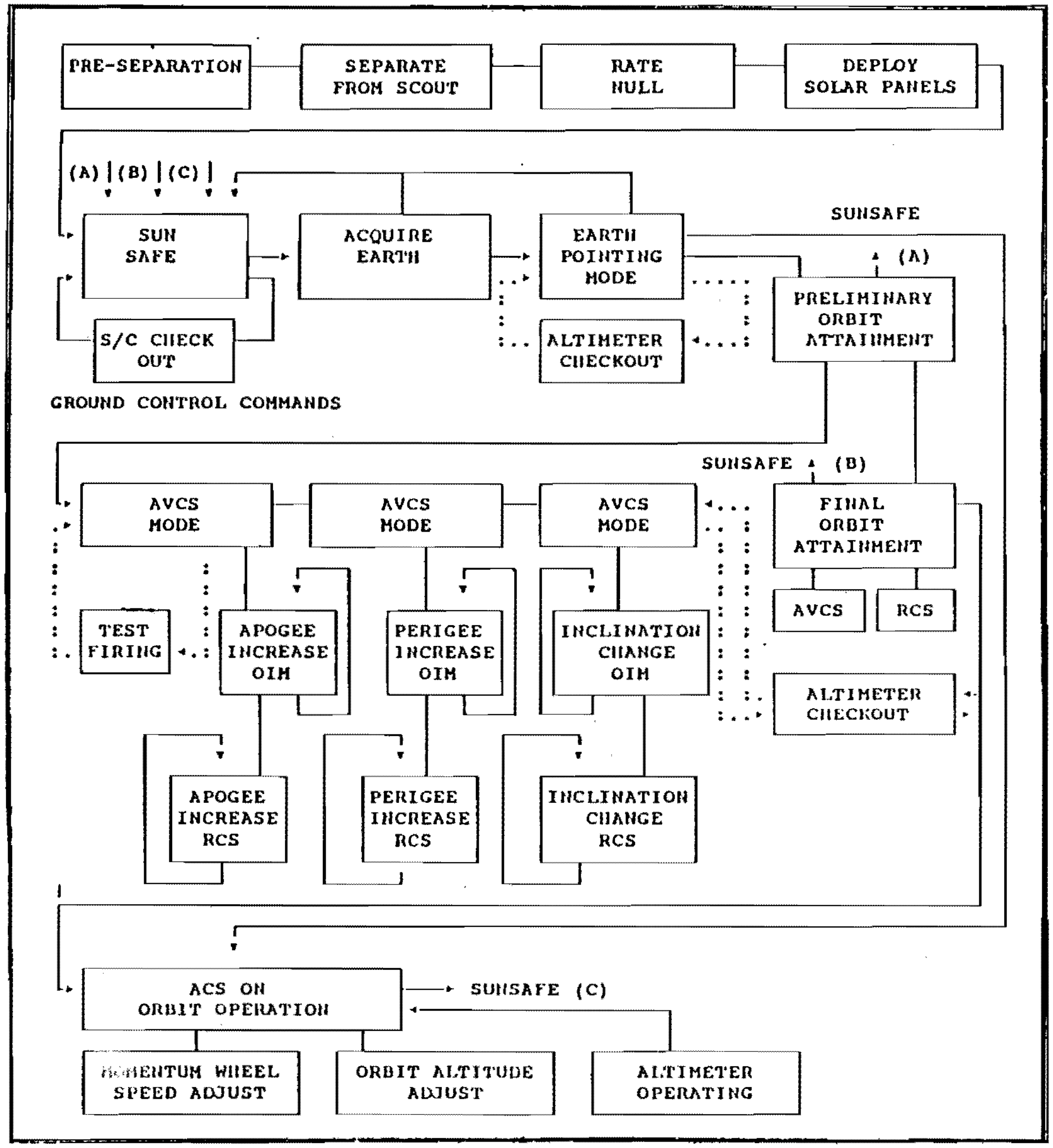


FIGURE 6

APPENDIX A

Orbit changing using minimum energy to transfer from an elliptic orbit to a final circular orbit with radius greater than the initial orbit semi-major axis:

Step 1: Add velocity (ΔV_1) at perigee to change to a transfer orbit whose apogee is tangent to the final ciruclar orbit.

Step 2: Add velocity (ΔV_2) at the new apogee to circularize the orbit at the desired altitude.

Step 1: $\frac{\Delta V_1}{U_N} = \sqrt{\frac{2}{\sigma_2(1+\sigma_2)}} - \sqrt{\frac{2}{\sigma_2(+\sigma_1)}}$

Step 2: $\frac{\Delta V_2 = 1 - \sqrt{\frac{2\sigma_2}{1+\sigma_2}}}{U_N}$

$V_1 \quad = \quad \sigma_1 = \frac{r_{pi}}{r_{ai}} \; i \; \sigma_2 = \frac{r_{p2}}{r_{a2}} = \frac{r_{pi}}{r_N} \; i\sigma_1 > \sigma_2$

$U_n \quad = \quad U_N = \sqrt{\frac{M}{r_N}}$ circular orbit velocity

R_p = perigee radius = Re + hp

R_a = apogee radius = Re + ha

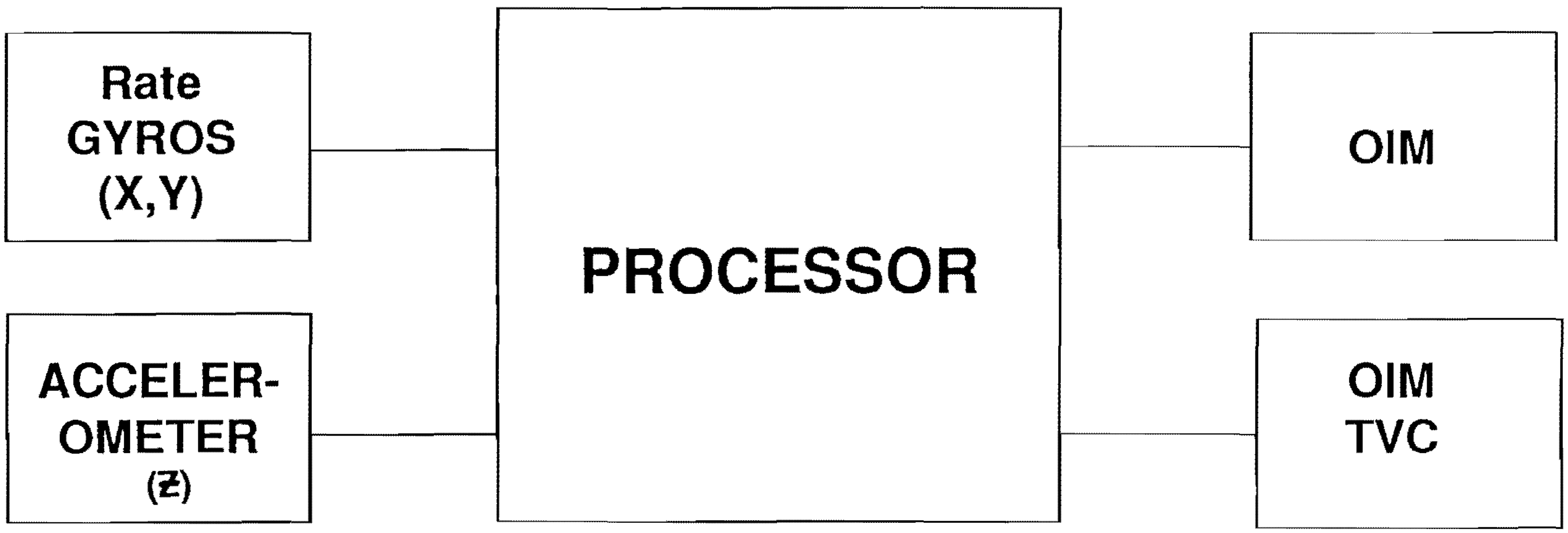
r_{Ns} = circular orbit radius = Re + h_n

R_E = Earth radius

Mi = gravitational constant

H_a = apogee altitude, hp = perigee altitude

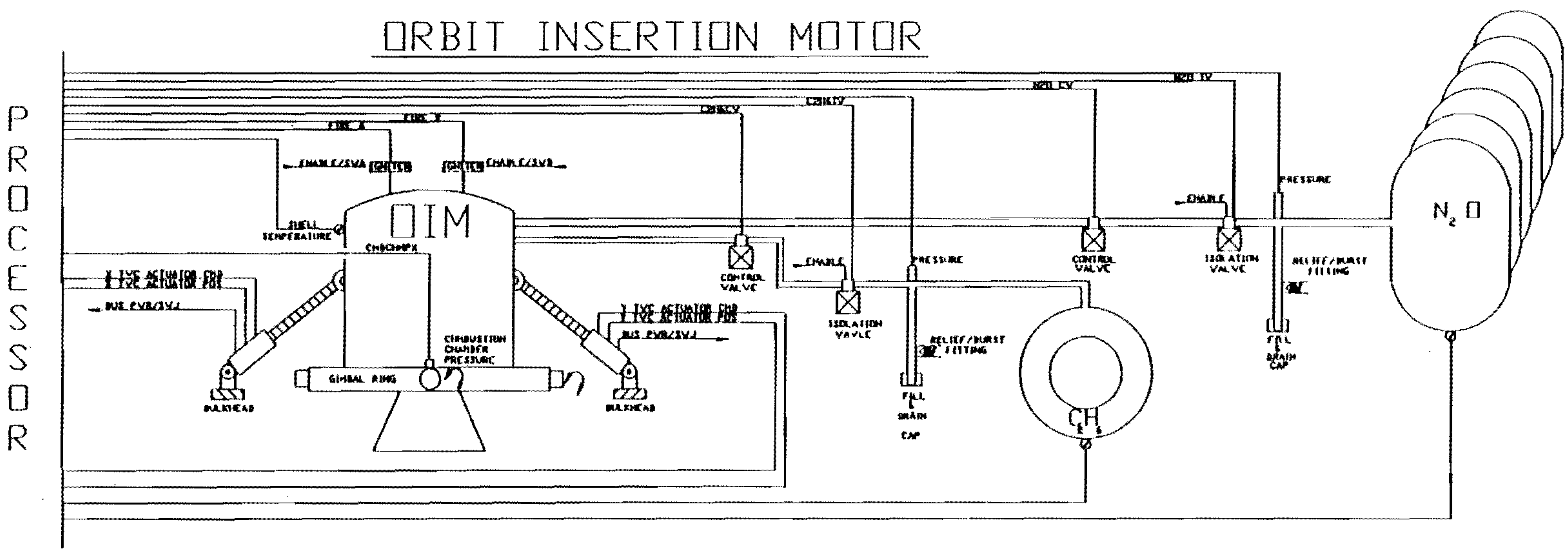
OIM SUBSYSTEM

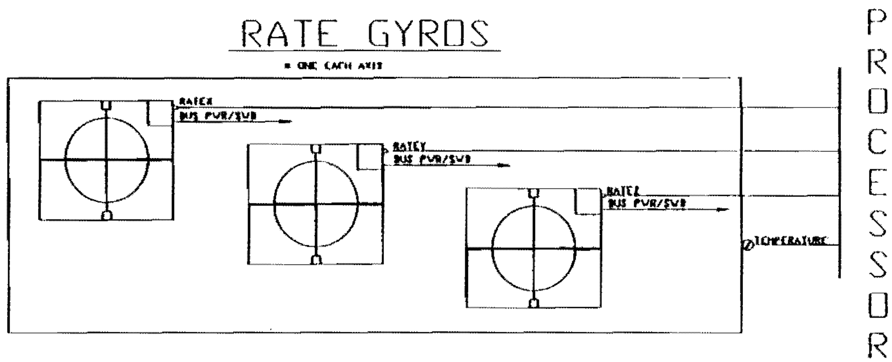


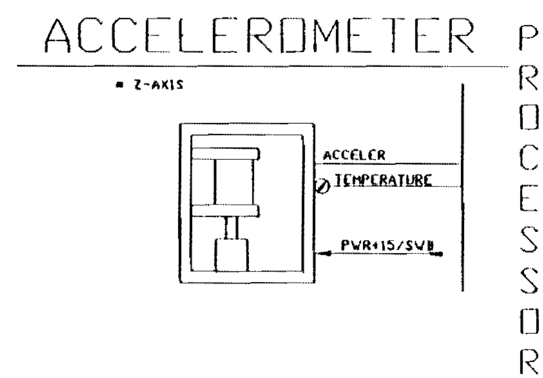
OIM SUBSYSTEM COMPONENT OVERVIEW

- ORBIT INSERTION MOTOR
  - RESTARTABLE - 25 TIMES
  - 40 - 150 LBS THRUST
  - TOTAL IMPULSE: 12,000 LB-SEC
  - 2 AXIS GIMBAL
- THRUSTER VECTOR CONTROL (TVC) ACTUATOR
  - 2 ACTUATORS - ALONG X AND Y AXIS
  - ±5° CONTROL RANGE
  - 10 BIT CONTROL FROM PROCESSOR
- ACCELEROMETER (Z AXIS)
  - INTEGRATED BY PROCESSOR TO SHUT-OFF OIM AT COMMANDED VELOCITY CHANGE
  - VELOCITY ACCURACY IS BETTER THAN 5 FT./SEC FOR 60 SECOND BURNS
- RATE GYRO'S
  - THE X AND Y RATE GYRO'S ARE INTEGRATED AND USED TO CONTROL THE TVC ACTUATOR
  - ANGULAR POSITION ACCURACY IS BETTER THAN ±5°

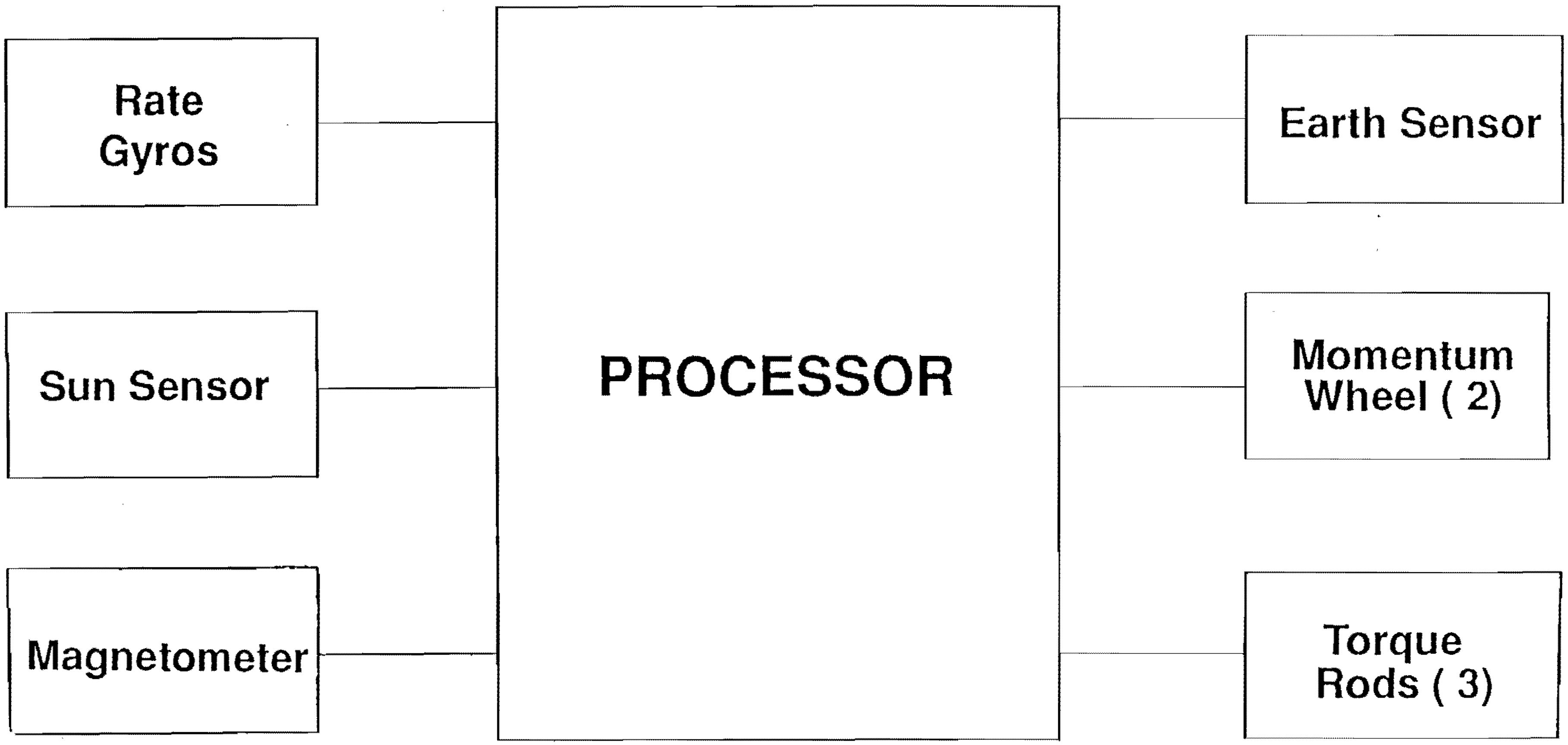
ORBITAL INSERTION MOTOR SCHEMATIC







ACS SUBSYSTEM



ACS COMPONENTS OVERVIEW

MOMENTUM WHEELS

- 2 WHEELS
- 1 FT-LB SEC EACH
- 0-5 OZ-IN TORQUE EACH
- 10 BIT CONTROL
- SPIN AXIS ± 15° OFF PITCH AXIS IN Ƶ DIRECTION
- ANGULAR MOMENTUM IS ALONG THE NEGATIVE PITCH AXIS
- COMMON CONTROL FOR TORQUE IN PITCH
- DIFFERENTIAL CONTROL TORQUE IN ROLL

EARTH SENSOR

- IR SENSOR
- 3 HEADS SPACED 120° AROUND SPACECRAFT
- 32 SENSING ELEMENTS PER HEAD
- 17 DEGREE FIELD OF VIEW FOR EACH HEAD
- ANGLED 69.3 OFF EARTH POINTING (Ƶ) AXIS
- COVERS 200KM TO 800KM ALTITUDE

ACS SUBSYSTEM COMPONENTS OVERVIEW CONTINUED

SUN SENSOR

- DUAL SLIT (0.05")SENSOR
- 5000 ELEMENTS/SLIT
- ±60° FIELD OF VIEW
- 1/4° ACCURACY
- 8 COARSE SENSORS - 8 QUADRANT COVERAGE

TORQUE RODS

- 3 UNITS, ONE IN EACH AXIS
- 2 COILS/UNIT
- ON/OFF CONTROL
- 15 A-M^2 DIPOLE MOMENT
- 0.05 A-M^2 RESIDUAL
- USED FOR MOMENTUM WHEEL DUMPS ONLY

MAGNETOMETER

- 3 AXIS MEASUREMENT
- 0-0.8 GAUSS RANGE
- 0.01 ACCURACY AND REVOLUTION

THERMAL CONTROL DIAGRAM