Reusable Reentry Satellite (RRS) System Design Study

PHASE B STUDY FINAL REPORT

APPENDIX E: Attitude Control System Study

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ITHACO

SAIC ®

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COMPANY
PRIVATE

RE-USABLE RE-ENTRY SATELLITE

(RRS)

ATTITUDE CONTROL SYSTEM

STUDY

Prepared for:

Science Applications International Corporation
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Under Subcontract Number 11-900286-39

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1.0   __INTRODUCTION__

ITHACO, Inc. has performed a study which consisted of a series of design analyses for an Attitude Control System (ACS) to be incorporated into the Re-usable Re-entry Satellite (RRS). The main thrust of the study was associated with defining the control laws and estimating the mass and power requirements of the ACS needed to meet the specified performance goals.

The analyses concentrated on the different on-orbit control modes which start immediately after the separation of the RRS from the launch vehicle. The three distinct on-orbit modes considered for these analyses are as follows.

Mode-1:   A Gravity Gradient (GG) three-axis stabilized spacecraft with active magnetic control.

Mode-2:   A GG stabilized mode with a controlled yaw rotation rate ("rotisserie") using three-axis magnetic control and also incorporating a 10 N-m-s momentum wheel along the (Z) yaw axis.

Mode-3:   A spin stabilized mode of operation with the spin about the pitch (Y) axis, incorporating a 20 N-m-s momentum wheel along the pitch (Y) axis and attitude control via thrusters.

To investigate the capabilities of the different controllers in these various operational modes, a series of computer simulations and trade-off analyses have been made to evaluate the achievable performance levels, and the necessary mass and power requirements.

2.0 MISSION DESCRIPTION

2.1 Mission Scenario

After the RRS has been injected into the operational orbit, the acquisition control mode of the ACS must stabilize the spacecraft and nullify any residual disturbances. Initial attitude information is provided to the ACS from the Initial Measurement Unit (IMU), which is used for guidance during the ascent and insertion phase. Once the RRS is properly oriented, the separation/deployment of the two spacecraft modules is accomplished via the use of astromasts. The nominal operational control scenario is then initiated.

Three modes of operation are invisioned: (1) gravity gradient, three-axis stabilized to provide a microgravity environment of less than 10^{-5} g, (2) gravity gradient stabilized with a controlled yaw rotation rate (rotisserie mode) for thermal balance, and (3) spin stabilized to impart an artificial gravity selectable between 0.1 to 1.5 g. All three modes of operation may be employed during a single mission and it is thus necessary to efficiently transfer control between modes.

Once the experimental phase has been completed, the RRS must be properly configured for the de-orbit maneuver. Prior to initiating the re-entry sequence, the spacecraft must be returned to the three-axis GG stabilized mode of operation. The astromasts are retracted in order to recombine the two spacecraft modules and the ACS provides the initial attitude parameters to the IMU for reentry guidance.

2.2   Spacecraft Configuration

The spacecraft configuration considered for the on-orbit control analyses is depicted in Figure 2.2-1. In order to simplify the magnitude of the dynamic and control simulations, the deployed configuration was used for all investigations, including the initial acquisition after separation from the launch vehicle. Designing the ACS for this more demanding deployed geometry insures that the ACS capability with the stowed configuration is more than adequate.

In the deployed configuration, the RRS consists of two large segments which are connected by three astromasts. As shown in Figure 2.2-1, the RRS extends approximately 3302 centimeters (1300 inches) in length and has a maximum diameter of 228.6 centimeters (90 inches). The spacecraft coordinate system and axis definitions are also depicted in Figure 2.2-1. The baseline orbital characteristics and the RRS mass properties are summarized in Table 2.2-1.

TABLE 2.2-1
SPACECRAFT MASS PROPERTIES AND ORBITAL CHARACTERISTICS

Inertia Matrix

(a)	Stowed Configuration	$I_x = 3340$ kg-m^2 (11414891.5 lb-in^2)
		$I_y = 3340$ kg-m^2 (11414891.5 lb-in^2)
		$I_z = 790$ kg-m^2 (2700262.05 lb-in^2)
(b)	Deployed Configuration	$I_x = 384916$ kg-m^2 (1315504824 lb-in^2)
		$I_y = 384916$ kg-m^2 (1315504824 lb-in^2)
		$I_z = 469$ kg-m^2 (1602807.5 lb-in^2)

Low Altitude Orbit
    Altitude   350 kilometers (189 nautical miles)
    Inclination   33.83°
    Eccentricity   0.0

High Altitude Orbit
    Altitude   897 kilometers (484 nautical miles)
    Inclination   98.0°
    Eccentricity   0.0

The GG stabilized mode and the spin mode have been studied in the lower inclined and lower altitude orbit, since the disturbance environment is expected to be more severe and the magnetic field components are least favorable. An ACS designed to accommodate these "worst case" conditions will perform satisfactorily at higher altitudes and higher inclinations, for both circular and elliptical orbits. Since the use of the rotisserie mode is of primary interest in polar orbits, this has been studied in the orbit with the higher inclination and higher altitude.
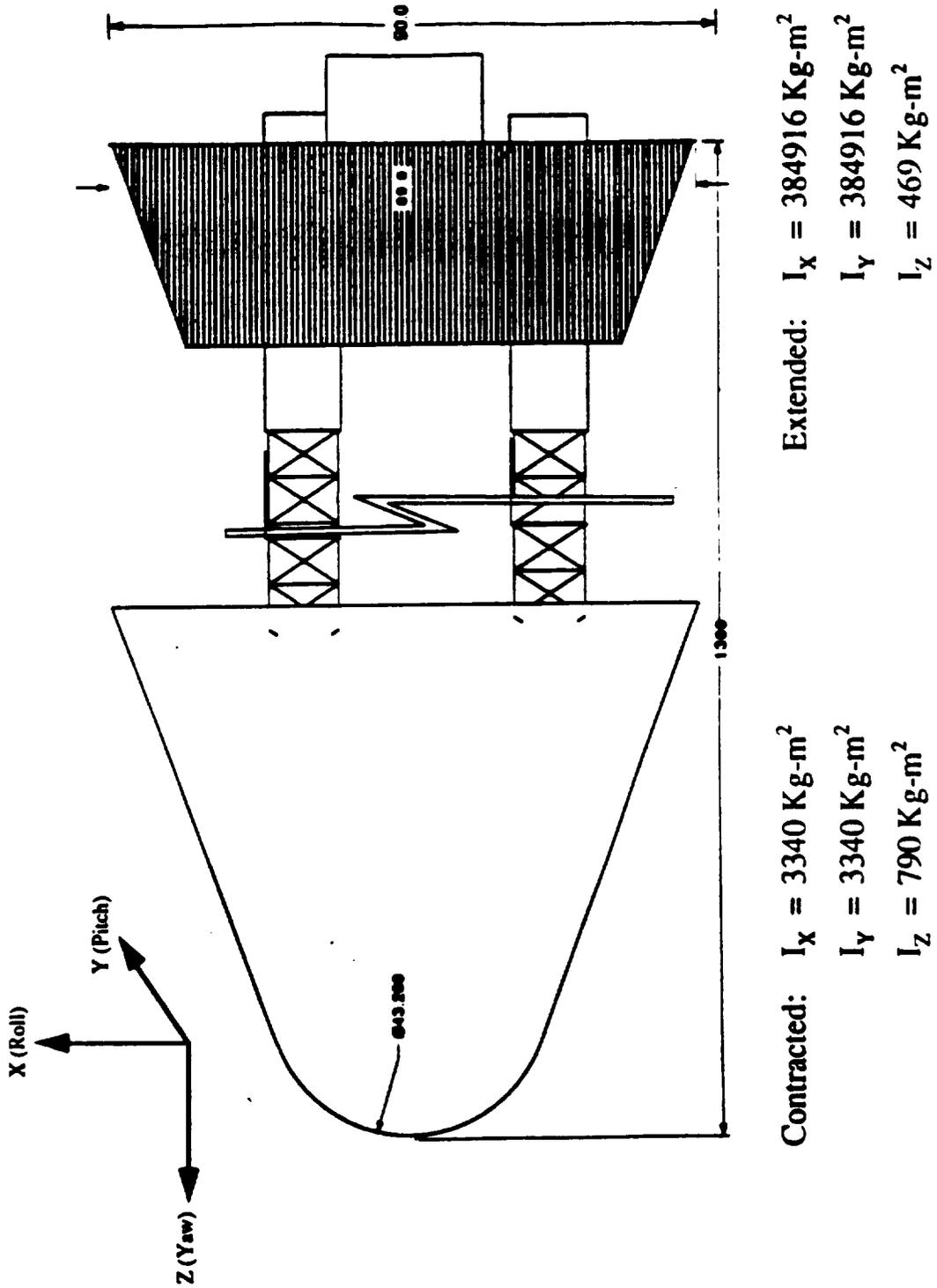
FIGURE 2.2-1
RRS COORDINATE AXES AND DEPLOYED CONFIGURATION

2.3   Disturbances

For the purposes of the current study, the spacecraft has been modelled as a rigid body, i.e., any flexible modes associated with the astromasts have not been included. The on-board propulsion system includes six propellant tanks which use bladder expulsion devices. Since the liquid is well contained, the affects of uncontrolled liquid motion (fuel slosh) are not significant and have not been modelled. The RRS vehicle design incorporates no other internal sources of uncompensated momentum.

Two sources of environmental disturbance torques have been considered - aerodynamic drag and solar pressure. For the purpose of modelling these disturbance torques, the spacecraft has been divided into five segments - two large masses and three astromasts. Based upon Figure 2.2-1, the effective cross-sectional areas of the lower and upper RRS segments have been estimated for exposure to the aerodynamic and solar pressure forces. For the three astromasts it has been assumed that the effective area exposed to the disturbance forces is 10% of that estimated if the astromasts were completely solid.

The mean air densities of $9.158e^{-12}$ and $5.245e^{-15}$ kg/m^3 have been used for the altitudes of 350 and 897 kilometers, respectively. A solar flux value of $4.5e^{-06}$ N/m^2 was used at both altitudes. Based upon these disturbance characteristics, the combined disturbance torques have been computed. For the orbit with the lower altitude and lower inclination, the net roll, pitch, and yaw disturbance torques for a single revolution are shown in Figures 2.3-1 thru 2.3-3, respectively.
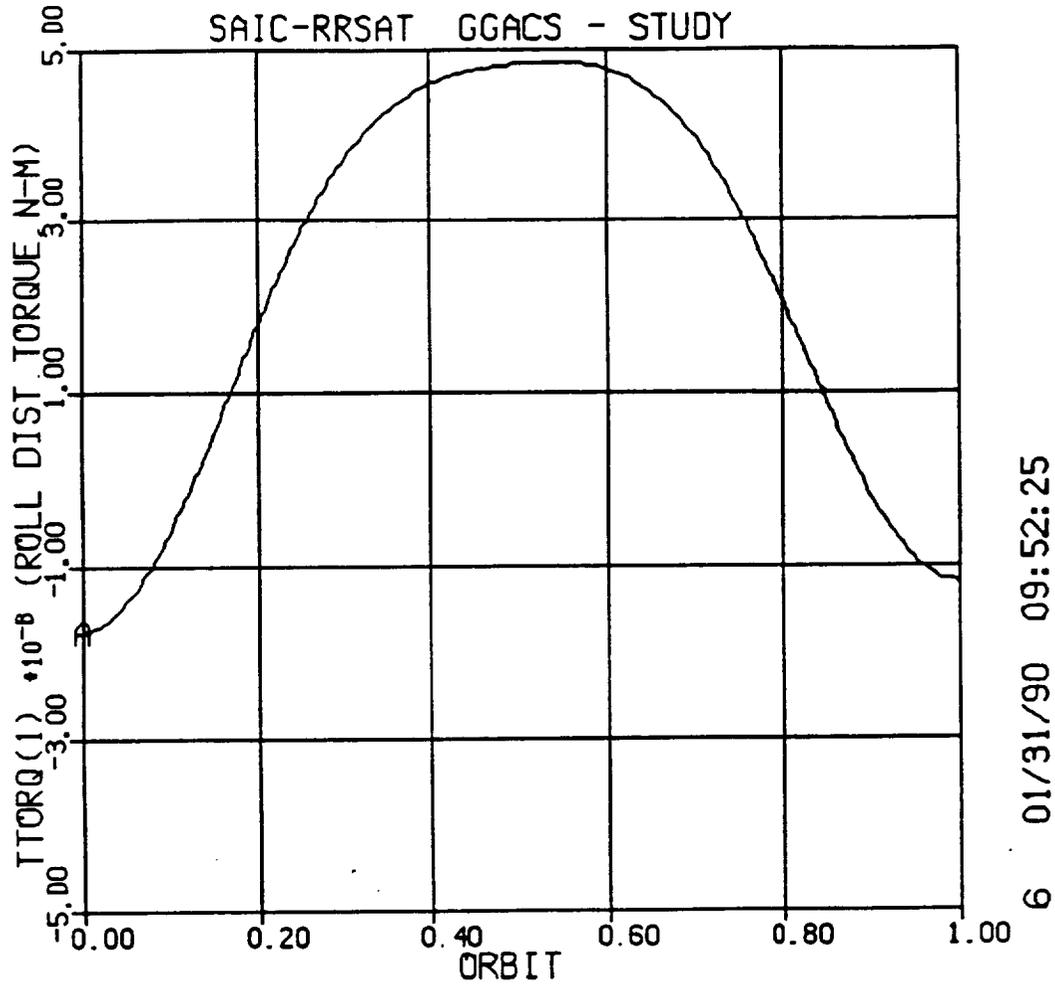
FIGURE 2.3-1
ROLL AXIS AERODYNAMIC AND SOLAR TORQUE
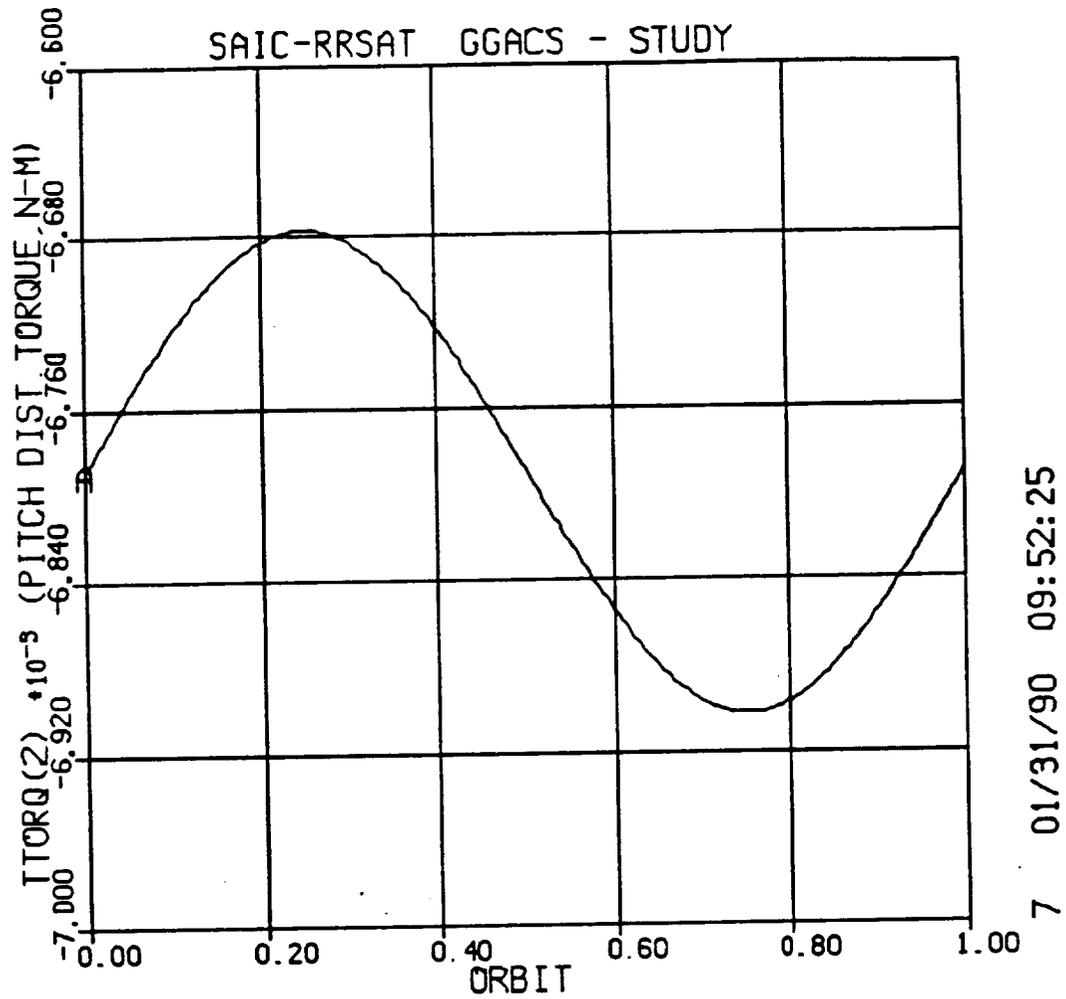
FIGURE 2.3-2
PITCH AXIS AERODYNAMIC AND SOLAR TORQUE
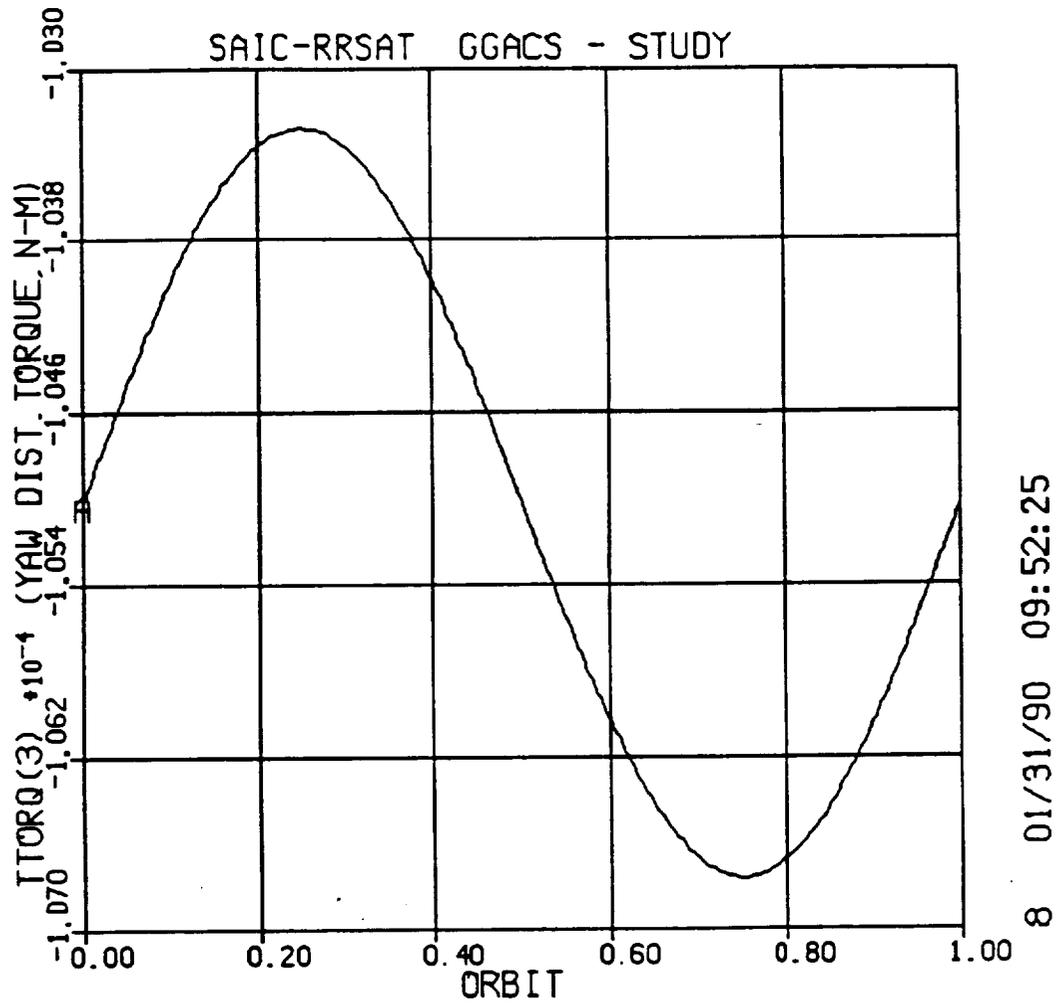
FIGURE 2.3-3
YAW AXIS AERODYNAMIC AND SOLAR TORQUE

3.0 SPACECRAFT DYNAMICS

3.1 Notation

The following notation is used through out the remaining text:

X,Y,Z	Body roll, pitch, and yaw axis, respectively, with X along the velocity vector, Y along the negative orbit normal, and Z toward the nadir, when no attitude errors exist in the three-axis stabilized spacecraft
$\underline{H}$	3 x 1 total momentum vector
$\underline{\omega}$	3 x 1 spacecraft body rate vector
$\underline{h}$	$\begin{bmatrix} 0 \\ h_2 \\ 0 \end{bmatrix}$, 3 x 1 wheel momentum vector
I	3 x 3 inertia matrix of the spacecraft
$\underline{T}_{gg}$	3 x 1 gravity gradient torque vector
$\underline{T}_d$	3 x 1 aerodynamic and solar pressure torque vector
$\underline{T}_c$	3 x 1 control torque vector generated by wheel and/or TORQRODs™
$\underline{q}$	4 x 1 quaternion vector associated with spacecraft attitudes
ϕ, θ, ψ	Euler roll, pitch, and yaw angles, respectively, with the order of rotations yaw-roll-pitch for the GG stabilized and spin modes. For the rotisserie mode (with yaw rotation control), the order of rotation is roll-pitch-yaw.
A(.)	direction cosine matrix
ω_0	orbital rate ·
$A_3(\psi)$	rotation matrix about the yaw axis by an angle ψ
$\dot{\phi}, \dot{\theta}, \dot{\psi}$	derivatives of ϕ, θ, ψ
$\underline{\dot{\omega}}$	angular acceleration of the spacecraft body

3.2 Equations of Motion

The dynamic equations of motion for the spacecraft are given by

$$\frac{d\underline{H}}{dt} + \underline{\omega} \times \underline{H} = \underline{T}_{gg} + \underline{T}_d - \underline{T}_c$$

where $\underline{H} = I\underline{\omega} + \underline{h}$

For the GG stabilized and spin modes of operation, the kinematic equations of motion of the attitude are given by

$$\frac{d\underline{q}}{dt} = \frac{1}{2} \Omega\,(\underline{\omega}_{sc}) \cdot \underline{q}$$

where $\Omega(\underline{\omega}_{sc}) = \begin{bmatrix} 0 & \omega_{zsc} & -\omega_{ysc} & \omega_{xsc} \\ -\omega_{zsc} & 0 & \omega_{xsc} & \omega_{ysc} \\ \omega_{ysc} & -\omega_{xsc} & 0 & \omega_{zsc} \\ -\omega_{xsc} & -\omega_{ysc} & -\omega_{zsc} & 0 \end{bmatrix}$

$$\underline{\omega}_{sc} = \underline{\omega} - A(\underline{q}) \cdot \begin{bmatrix} 0 \\ -\omega_0 \\ 0 \end{bmatrix}$$

$$A(q) = \begin{bmatrix} A_{11} & A_{12} & A_{13} \\ A_{21} & A_{22} & A_{23} \\ A_{31} & A_{32} & A_{33} \end{bmatrix} = \begin{bmatrix} q_1^2 - q_2^2 - q_3^2 + q_4^2 & 2(q_1 q_2 + q_3 q_4) & 2(q_1 q_3 - q_2 q_4) \\ 2(q_1 q_2 - q_3 q_4) & -q_1^2 + q_2^2 - q_3^2 + q_4^2 & 2(q_2 q_3 + q_1 q_4) \\ 2(q_1 q_3 + q_2 q_4) & 2(q_2 q_3 - q_1 q_4) & -q_1^2 - q_2^2 + q_3^2 + q_4^2 \end{bmatrix}$$

with $q_1^2 + q_2^2 + q_3^2 + q_4^2 = 1$

Assuming Z-X-Y (yaw-roll-pitch) Euler rotations, the attitude errors can be obtained as follows:

$$\phi = \arcsin\,(A_{23})$$
$$\theta = \arctan\,(-\frac{A_{13}}{A_{33}})$$
$$\psi = \arctan\,(-\frac{A_{21}}{A_{22}})$$

3.2 Equations of Motion (Cont'd)

For the rotisserie mode of operation, the following kinematic equations have been used with the roll-pitch-yaw (X-Y-Z) sequence of Euler rotations.

$$A(\psi, \theta, \phi) = \begin{bmatrix} \cos\psi \cos\theta & \cos\psi \sin\theta \sin\phi + \sin\psi \cos\phi & -\cos\psi \sin\theta \cos\phi + \sin\psi \sin\phi \\ -\sin\psi \cos\theta & -\sin\psi \sin\theta \sin\phi + \cos\psi \cos\phi & \sin\psi \sin\theta \cos\phi + \cos\psi \sin\phi \\ \sin\theta & -\cos\theta \sin\phi & \cos\theta \cos\phi \end{bmatrix}$$

The corresponding Euler angle rate vector is defined as

$$\begin{bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} = \frac{1}{\cos\theta} \begin{bmatrix} \cos\psi & -\sin\psi & 0 \\ \sin\psi \cos\theta & \cos\psi \cos\theta & 0 \\ -\sin\theta \cos\psi & \sin\theta \sin\psi & \cos\theta \end{bmatrix} [\underline{\omega} - A \cdot \underline{\omega}_0]$$

where $\underline{\omega}_0 = \begin{bmatrix} 0 \\ -\omega_0 \\ 0 \end{bmatrix}$

and $\phi = \arctan(-\frac{A_{32}}{A_{33}})$

$\theta = \arcsin(A_{31})$

$\psi = \arctan(-\frac{A_{21}}{A_{11}})$

The disturbance torque profiles for the $\underline{T}_d$ vector are shown in Figures 2.3-1 thru 2.3-3. The gravity gradient torque vector is given by

$$\underline{T}_{gg} = 3 \omega_0^2 \cdot \underline{r} \times (I\underline{r})$$

where $\underline{r} = A(\underline{q}) \cdot \begin{bmatrix} 0 \\ 0 \\ -1 \end{bmatrix}$ or $A(\psi, \theta, \phi) \cdot \begin{bmatrix} 0 \\ 0 \\ -1 \end{bmatrix}$

The control law vector $\underline{T}_c$ is developed for each of the control scenarios as follows.

3.2.1 Gravity Gradient Stabilized Mode

This mode has been analyzed for a circular orbit with an altitude of 350 kilometers and inclination of 33.83 degrees. Because the minimum inertia value is about the yaw (Z) axis, the spacecraft is gravity gradient stabilized but librates for a long period of time before attaining the equilibrium point of zero attitude error. In order to reduce this acquisition time, an active magnetic control scenario has been proposed, which utilizes three TORQRODs, two two-axis magnetometers, and an attitude error estimating scheme which calculates the attitude errors and their derivatives. These estimates can be provided by processing the three-axis magnetometer data via an on-board Kalman Filter, i.e., via the Magnetic Attitude Determination Subsystem (MADS).

The total momentum of the spacecraft is given by

$$\underline{H} = I\underline{\omega} + \underline{h}$$

where $\underline{h}$ represents a bias momentum vector which is zero for the GG stabilized control mode, but may have non-zero values for the other modes of operation. Thus for GG control,

$$\underline{h} = \begin{bmatrix} 0 \\ 0 \\ 0 \end{bmatrix}$$

The active control torque generated by the TORQRODs is given by

$$\underline{T}_c = \underline{M} \times \underline{B}$$

where $\underline{B}$ is the Earth's magnetic field vector and $\underline{M}$ is the dipole moment vector generated by the TORQRODs. The $\underline{M}$ vector is given by

$$\underline{M} = \frac{\underline{m} \times \underline{B}}{|\underline{B}|}$$

where $\underline{m} = K_p \cdot \underline{a} + K_d \cdot \underline{\dot{a}}$

and $\underline{a} = $ attitude error vector $= \begin{bmatrix} \phi \\ \theta \\ \psi \end{bmatrix}$

$\underline{\dot{a}} = $ attitude rate vector $= \begin{bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix}$

$K_p = 3 \times 3$ diagonal position gain matrix
$K_d = 3 \times 3$ diagonal rate gain matrix

In this study the $\underline{a}$ and $\underline{\dot{a}}$ vectors are assumed to be perfect, i.e., no noise or estimation errors in the measurements are included.

3.2.2 Gravity Gradient Stabilized With Yaw Rotation (Rotisserie) Mode

This mode has been analyzed for the circular polar orbit with an inclination of 98 degrees and an altitude of 897 kilometers. This operational mode is identical to the GG stabilized mode discussed in the previous section, with the addition of a constant rotation of 0.2 rpm about the spacecraft yaw (Z) axis in order to provide the required thermal balance conditions. To provide the attitude control about the roll (X) and pitch (Y) axes, and to maintain the rotation about the yaw axis, the previous GG control laws are applied with the proper modifications.

Again three TORQRODs and two two-axis magnetometers are required, and the Kalman filter algorithms are needed to estimate the attitude errors and rates from the magnetometer data. A yaw wheel with a constant momentum of 10 N-m-s has been used to approximately cancel the yaw axis spacecraft momentum produced by the yaw rotation. This technique provides a zero momentum yaw axis which is much more easily maneuvered once per orbit than a yaw axis with a nonzero (bias) momentum.

The total momentum of the spacecraft is given by

$$\underline{H} = I\underline{\omega} + \underline{h} = I\underline{\omega} + \begin{bmatrix} 0 \\ 0 \\ 10 \end{bmatrix} \text{N-m-s}$$

Again the active control torque generated by the TORQRODs is given by

$$\underline{T}_c = \underline{M} \times \underline{B}$$

where $\underline{B}$ is the instantaneous Earth's magnetic field vector along the spacecraft body axes and $\underline{M}$ is the dipole moment vector generated by the TORQRODs.

In order to translate between the rotating body axes and the fixed orbital frame, define a rotation matrix, $A_3(\psi)$, which translates the orbital frame to the body frame via a rotation about the yaw axis by an angle ψ. The magnetic field components are measured in the spacecraft body frame, which are thus defined in the orbital frame as

$$\underline{B}_c = A_3^T (\psi) \cdot \underline{B} = \begin{bmatrix} \cos\psi & -\sin\psi & 0 \\ \sin\psi & \cos\psi & 0 \\ 0 & 0 & 1 \end{bmatrix} \cdot \underline{B}$$

The control dipole moment vector required in the orbital frame is calculated as

$$\underline{D} = \frac{\underline{m} \times \underline{B}_c}{|\underline{B}_c|}$$

where $\underline{m} = K_p \cdot \underline{a} + K_d \cdot \dot{\underline{a}} - K_p \cdot \underline{a}_r - K_d \cdot \dot{\underline{a}}_r$

3.2.2 Gravity Gradient Stabilized With Yaw Rotation (Rotisserie) Mode (Cont'd)

K_p, K_d, $\underline{a}$, and $\underline{\dot{a}}$ are the same terms as previously defined for the GG stabilized mode of operation. The quantities $\underline{a}_r$ and $\underline{\dot{a}}_r$ are defined as follows:

$$\underline{a}_r = \begin{bmatrix} 0 \\ 0 \\ \psi_r \end{bmatrix}$$

$$\underline{\dot{a}}_r = \begin{bmatrix} 0 \\ 0 \\ \dot{\psi}_r \end{bmatrix}$$

where

$\dot{\psi}_r$ = reference yaw rate = 0.02 rad/sec = 0.2 rpm

ψ_r = reference yaw position signal = $\dot{\psi}_r \cdot$ time

Thus the $\underline{M}$ vector in the body axes is given by

$$\underline{M} = A_3\,(\psi) \cdot \underline{D} = \begin{bmatrix} \cos\psi & \sin\psi & 0 \\ -\sin\psi & \cos\psi & 0 \\ 0 & 0 & 1 \end{bmatrix} \cdot \underline{D}$$

3.2.3 Spin Stabilized Mode

This mode has been evaluated using a circular orbit at an altitude of 350 kilometers with an inclination 33.83 degrees. The first event in this sequence is to spin up the RRS about the pitch (Y) axis to the predetermined rate between 0.6 and 9.6 rpm, in order to provide the corresponding artificial gravity level between 0.1 and 1.5 g. Due to the very large pitch axis inertia in the deployed configuration, the RRS needs a correspondingly large spin up torque. The magnetic TORQRODs cannot provide sufficient torque to accomplish this spin up maneuver in the required time, thus propulsion is recommended. The spin up and subsequent despin maneuvers are performed with the use of a pair of thrusters, each with a thrust level of 2.2 Newtons (0.5 pounds force) and with a moment arm length of 1750 centimeters (689 inches).

Because the preliminary RRS inertia values for both the roll and the pitch axes are identical, a 20 N-m-s bias momentum wheel has been introduced to make the effective pitch axis inertia larger than that for the roll axis. This provides stability and insures that the RRS is spun up only about the pitch (Y) axis. The total momentum of the spacecraft is given by

$$\underline{H} = I\underline{\omega} + \underline{h} = I\underline{\omega} + \begin{bmatrix} 0 \\ -20 \\ 0 \end{bmatrix} \text{N-m-s}$$

and the thruster spin up torque can be identified as

$$\underline{T}_c = \begin{bmatrix} 0 \\ -78 \\ 0 \end{bmatrix} \text{N-m}$$

Once the appropriate spacecraft rotation rate is achieved, the spin up thruster logic is terminated.

4.0 SIMULATION ANALYSES

This section presents the computer simulation results for the different control modes described in the previous section. In order to evaluate the performance of each ACS mode, the following curves depict the pertinent parameters (such as attitude errors, dipole magnitudes, spin rates, etc) with appropriate units, against time (number of orbits). For the first two modes (GG stabilized and the rotisserie modes), the initial conditions for all attitude errors are assumed to be 0.1 radian (5.7 degrees). These starting conditions facilitate the evaluation of the steady state performance of the corresponding control algorithms.

Figures 4.0-1 thru 4.0-3 show the attitude errors in degrees for the three-axis GG stabilized RRS (with no bias momentum wheel). Figures 4.0-4 thru 4.0-6 depict the same data, but with expanded scales to show the steady-state errors. These curves indicate that the maximum steady-state roll error is less than 0.07 degree, the pitch error has an offset of around -0.35 degree, while the yaw error has an initial offset of around -1.8 degrees which decreases to around -1.2 degrees after 16 orbits. These simulations are based on constant aerodynamic disturbance torques of -0.0068 and -0.000105 N-m about the pitch and yaw axes, respectively, which introduce the corresponding attitude offsets. Figures 4.0-7 thru 4.0-9 show the magnetic control dipole components necessary to achieve the above steady state errors. These plots indicate that the control dipole magnitude requirements are well below the maximum value 350 Am^2, except for the initial acquisition and capture sequence.

Figures 4.0-10 and 4.0-11 show the roll and pitch error histories for the rotisserie (yaw rotation) mode. This operational control mode was also simulated with maximum dipoles of 350 Am^2 along each axis. This data indicates that the steady state roll and pitch attitude errors are well controlled, with values around zero. Figure 4.0-12 depicts the yaw rotation rate history from 0 to the steady state value of 0.2 rpm (0.02 rad/sec). Figures 4.0-13 thru 4.0-15 show the magnetic control dipole requirements along each axis.

For the spin mode of operation, the initial roll and yaw attitude conditions were taken as the steady state conditions obtained by the three-axis GG stabilized control mode. A basic assumption is that the spin up sequence will be activated only after gravity gradient stabilization. Thus the roll and yaw initial attitude errors were chosen as 0.2 and 1.6 degrees, respectively. Two thrusters, each with a thrust level of 2.2 Newton and a moment arm of 1750 centimeters, were then fired continuously until attaining the required spin. Figure 4.0-16 shows the pitch axis spin rate build up and indicates that the final rate of 1.0 rad/sec (9.6 rpm) is achieved in less than one orbit. During the spin up, no roll or yaw control was provided for the RRS. Thus Figures 4.0-17 and 4.0-18 show that the roll and yaw attitude errors are slowly increasing and that thrusters will be required for auxiliary control.

4.0 SIMULATION ANALYSES (Cont'd)

The summary of the performance of each of the three modes is presented in Table 4.0-1.

TABLE 4.0-1
PERFORMANCE SUMMARIES FOR THE RRS

| MODE | MAX ATTITUDE CONTROL ERRORS (DEGREES)* | | | SPIN RATE (AXIS/RPM) |
	ROLL	PITCH	YAW	
GG	0.1 (0.0)**	0.1 (-0.3)**	0.4 (-1.2)**	N/A
Rotisserie	0.1	0.1	N/A	Yaw / 0.2
Spin	TBD	N/A	TBD	Pitch / 9.6

*   Simulation did not include sensor model or sensor noise.
** Attitude offset
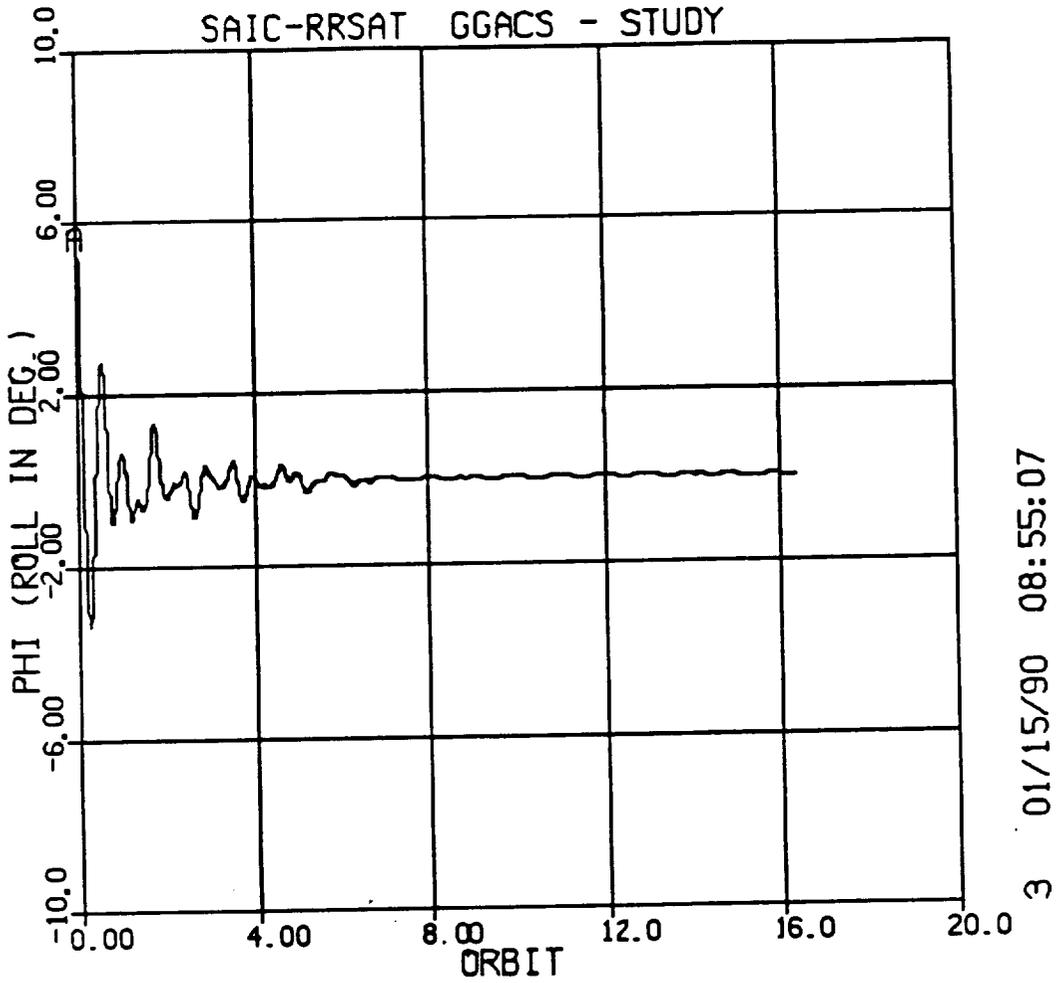
FIGURE 4.0-1
ROLL AXIS ATTITUDE ERROR FOR GG STABILIZED MODE
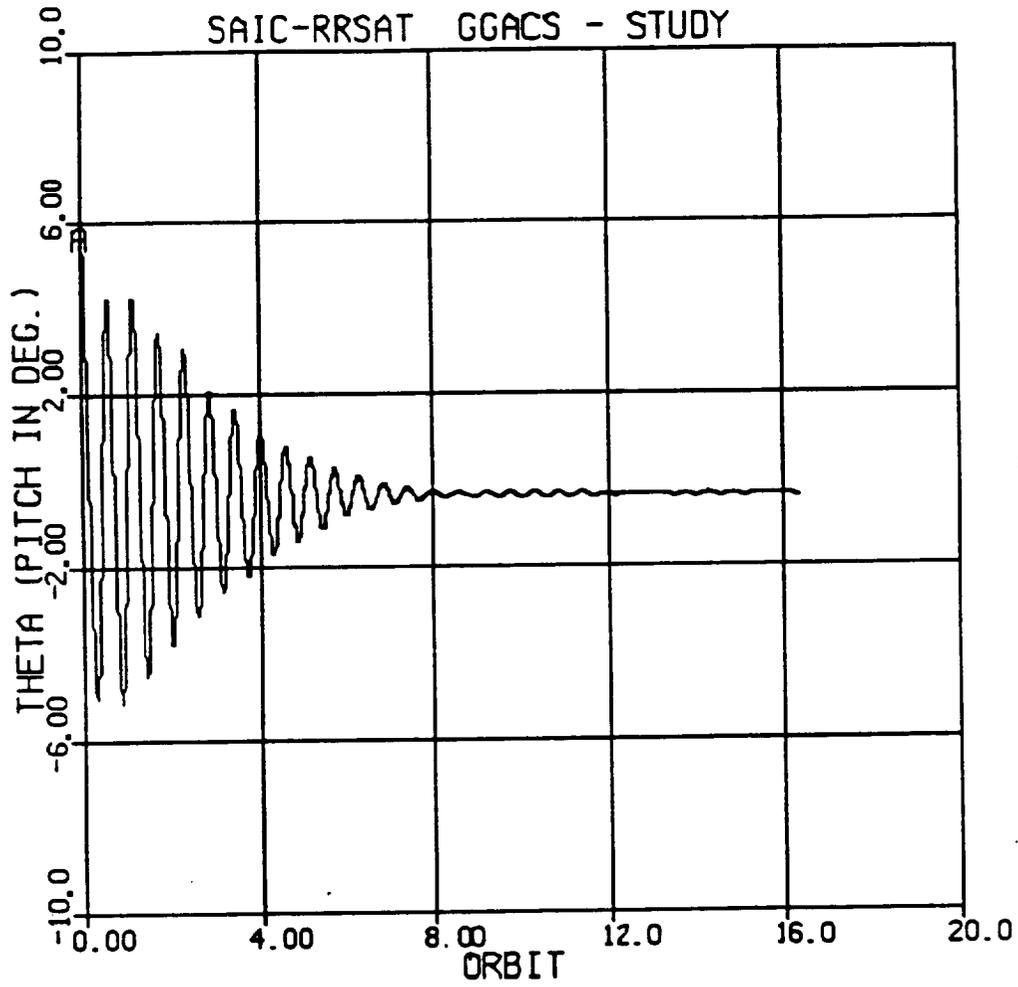
FIGURE 4.0-2
PITCH AXIS ATTITUDE ERROR FOR GG STABILIZED MODE
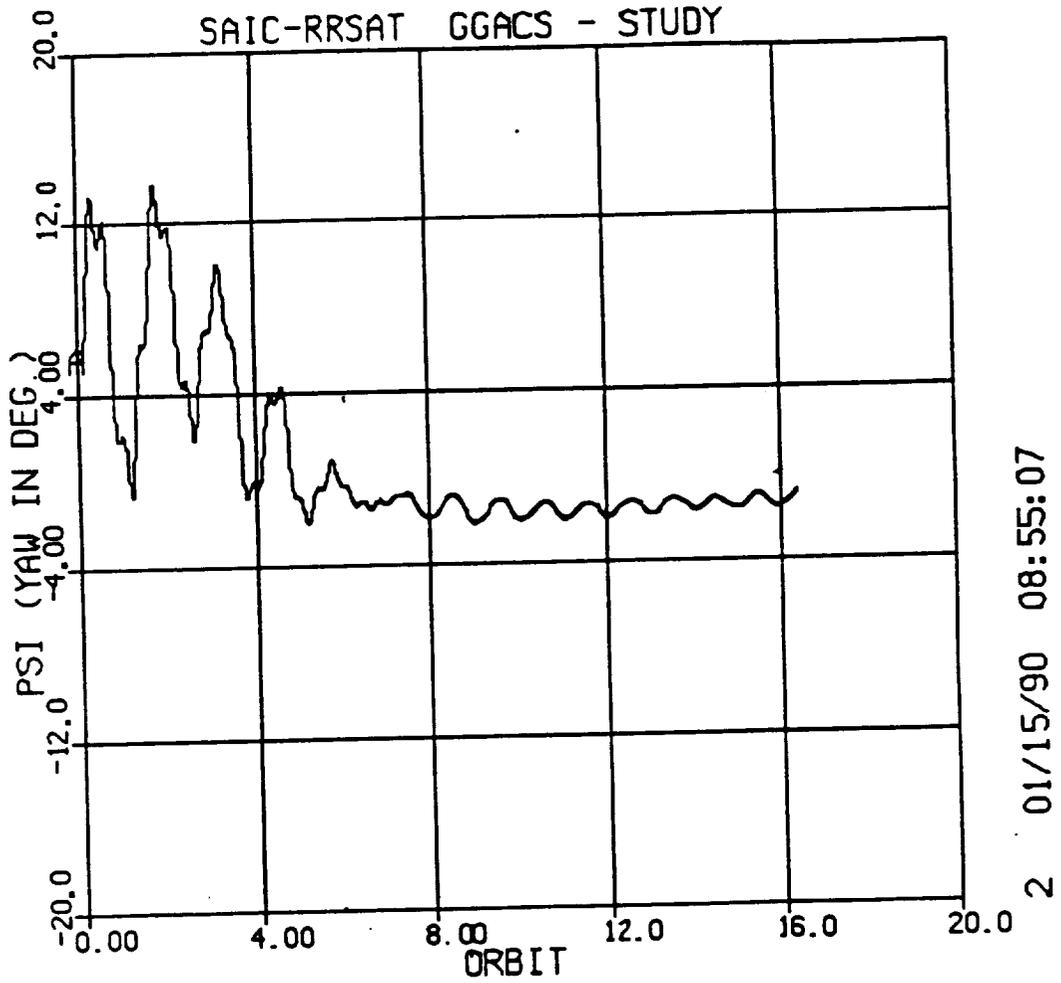
FIGURE 4.0-3
YAW AXIS ATTITUDE ERROR FOR GG STABILIZED MODE
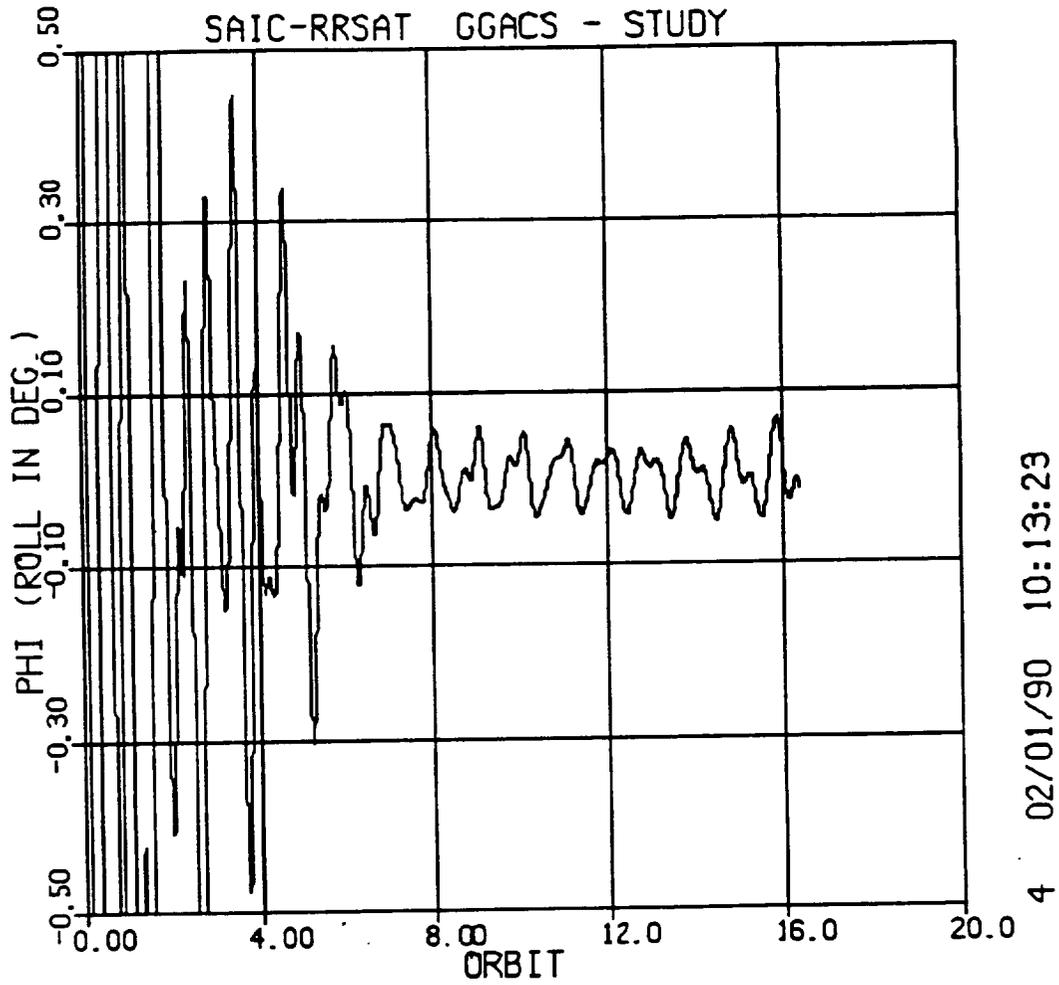
FIGURE 4.0-4
ROLL AXIS ATTITUDE ERROR FOR GG STABILIZED MODE (EXPANDED)
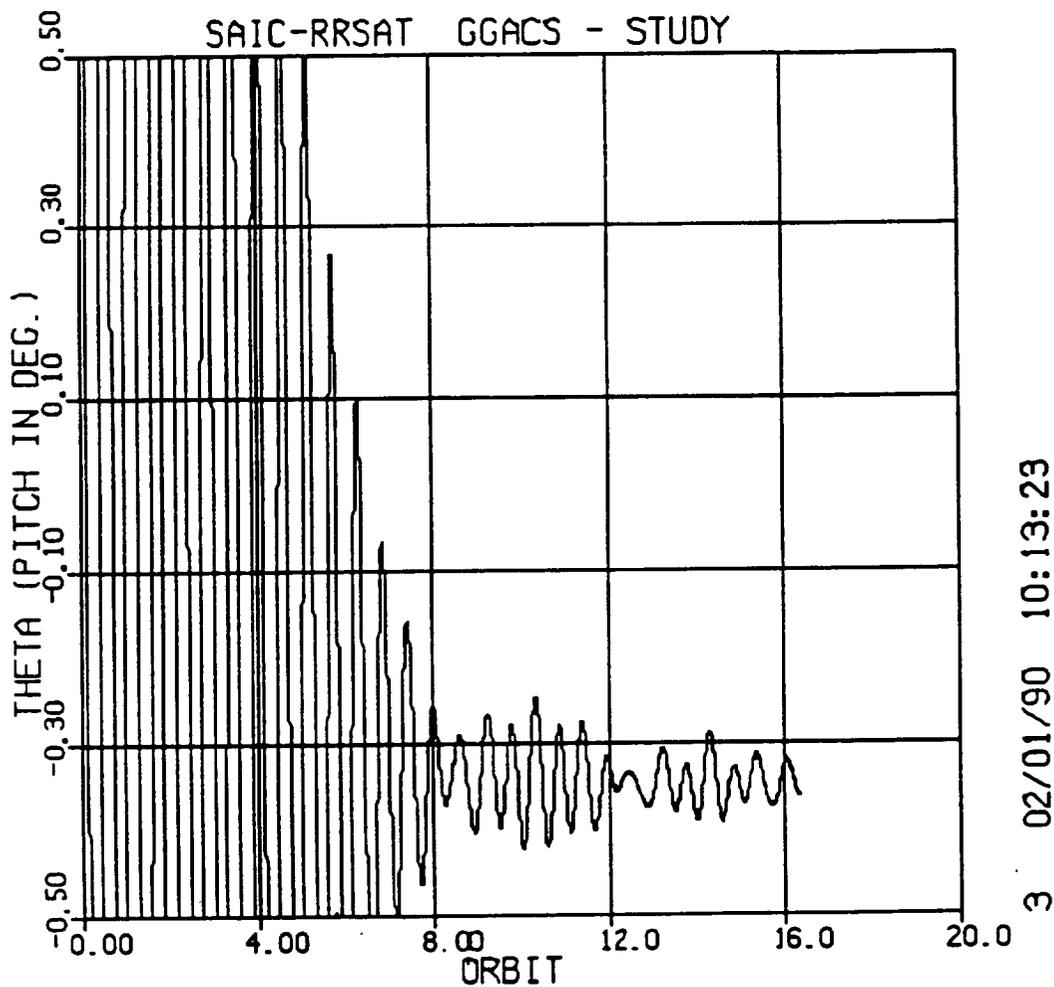
FIGURE 4.0-5
PITCH AXIS ATTITUDE ERROR FOR GG STABILIZED MODE (EXPANDED)
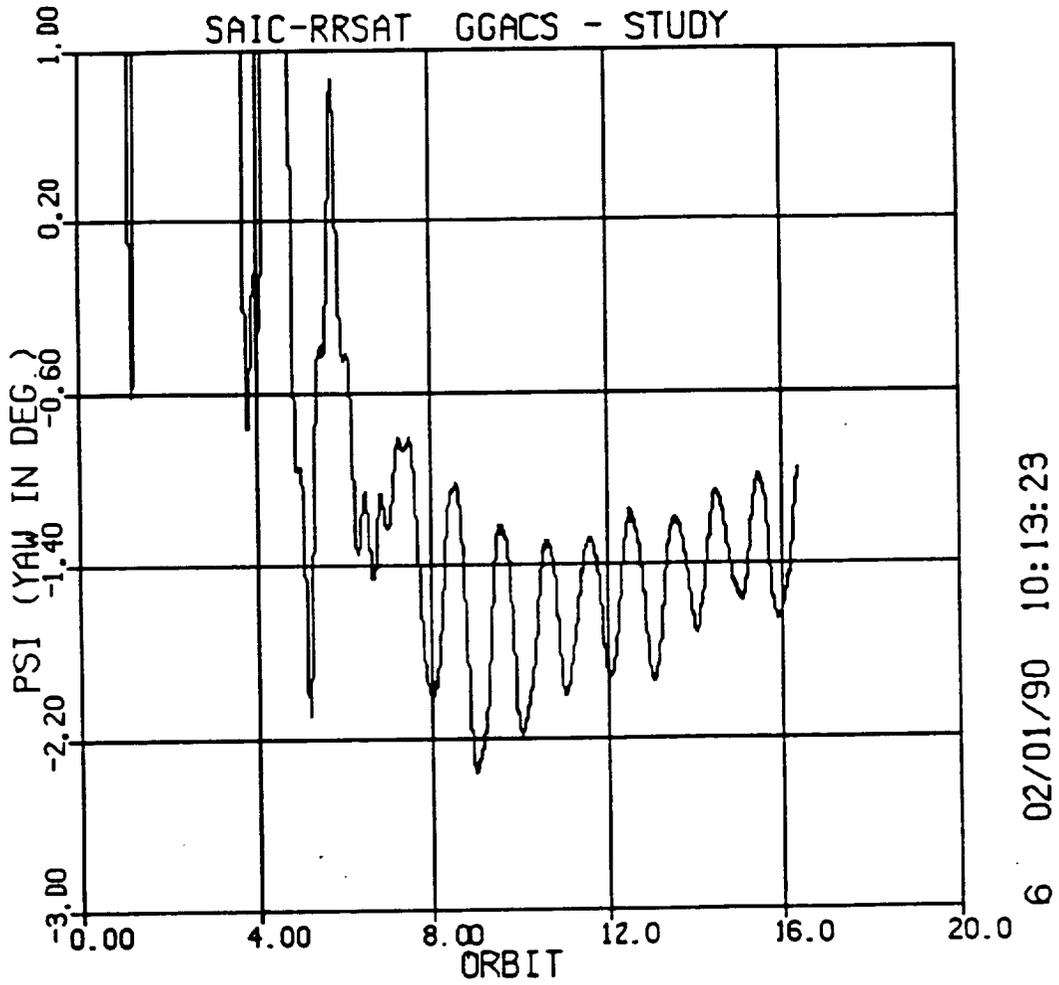
FIGURE 4.0-6
YAW AXIS ATTITUDE ERROR FOR GG STABILIZED MODE (EXPANDED)
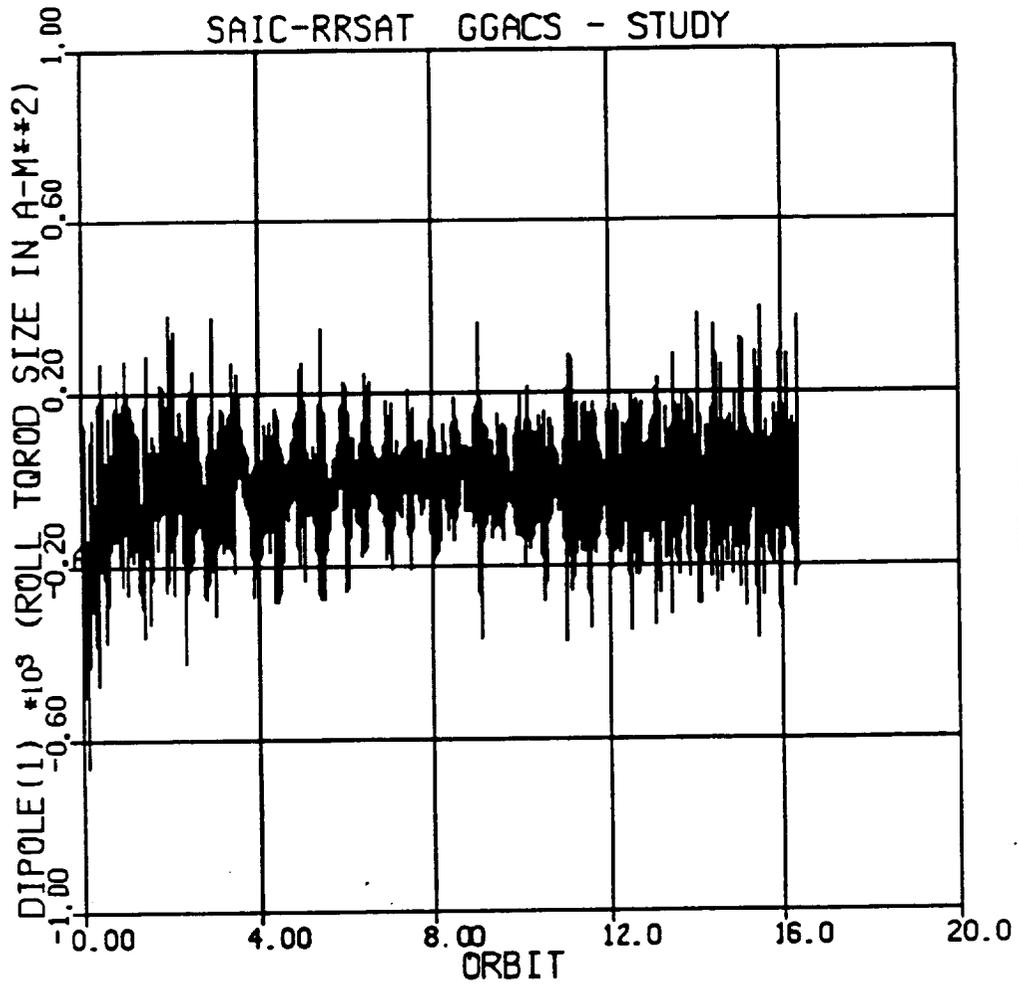
FIGURE 4.0-7
ROLL AXIS DIPOLE REQUIREMENTS FOR GG STABILIZED MODE

SAIC-RRSAT   GGACS - STUDY

DIPOLE(2) *10³ (PITCH TQROD SIZE IN A-M**2)

ORBIT

7   01/15/90   08:55:07
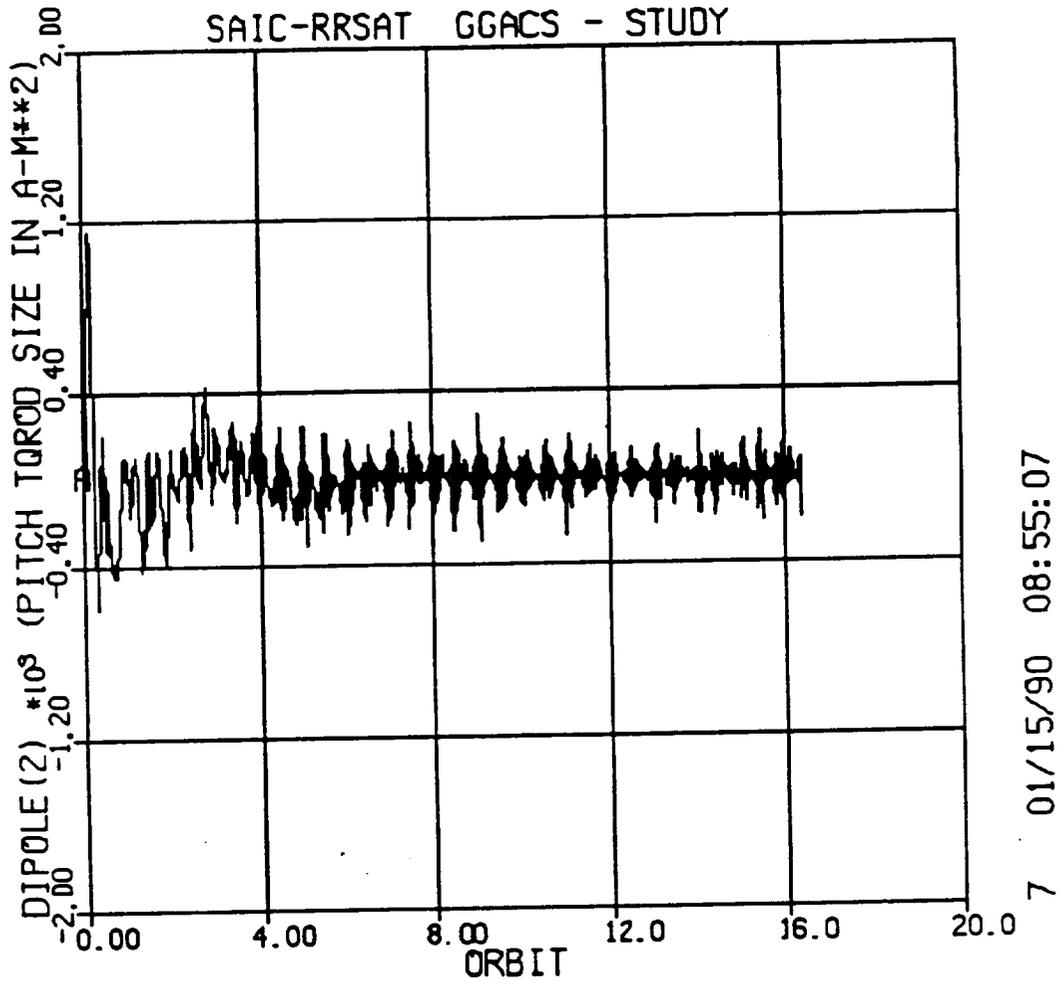
FIGURE 4.0-8
PITCH AXIS DIPOLE REQUIREMENTS FOR GG STABILIZED MODE
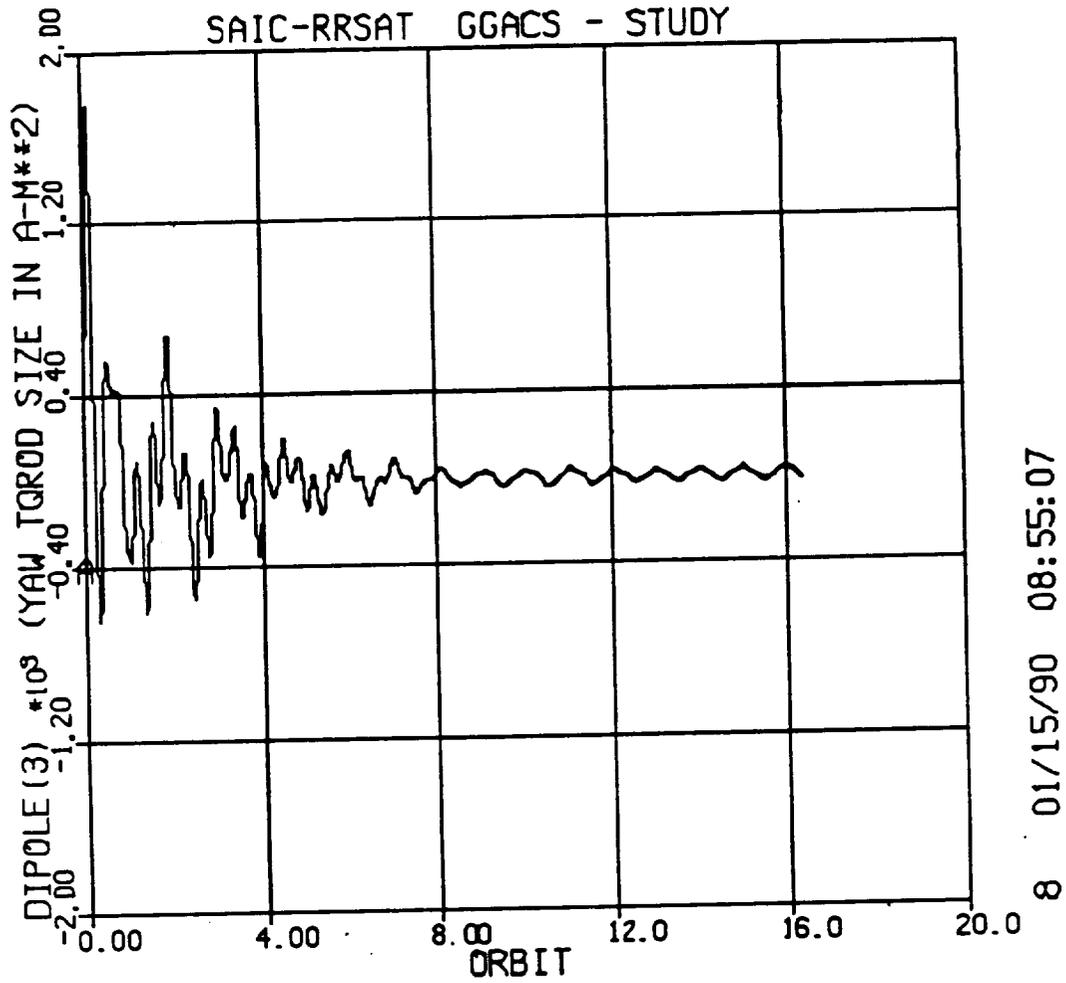
FIGURE 4.0-9
YAW AXIS DIPOLE REQUIREMENTS FOR GG STABILIZED MODE
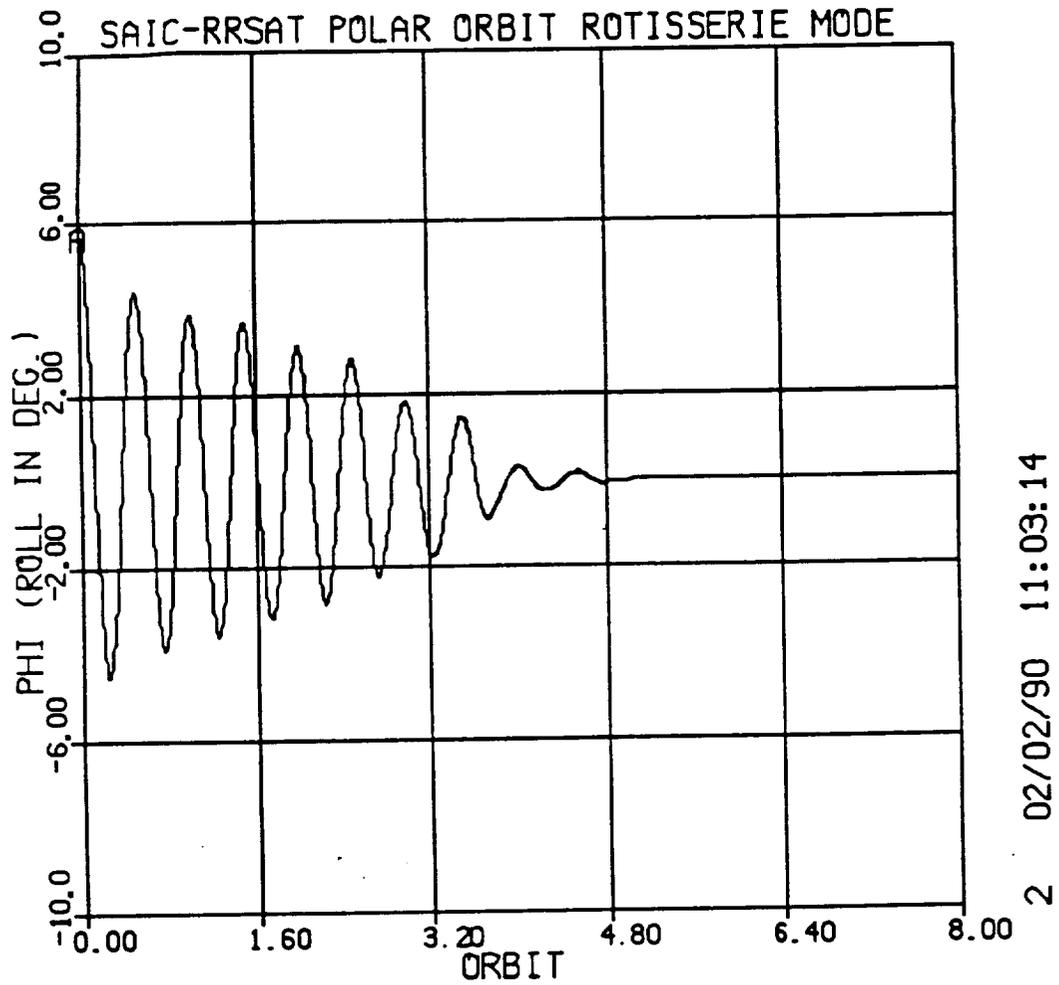
FIGURE 4.0-10
ROLL AXIS ATTITUDE ERROR FOR ROTISSERIE MODE
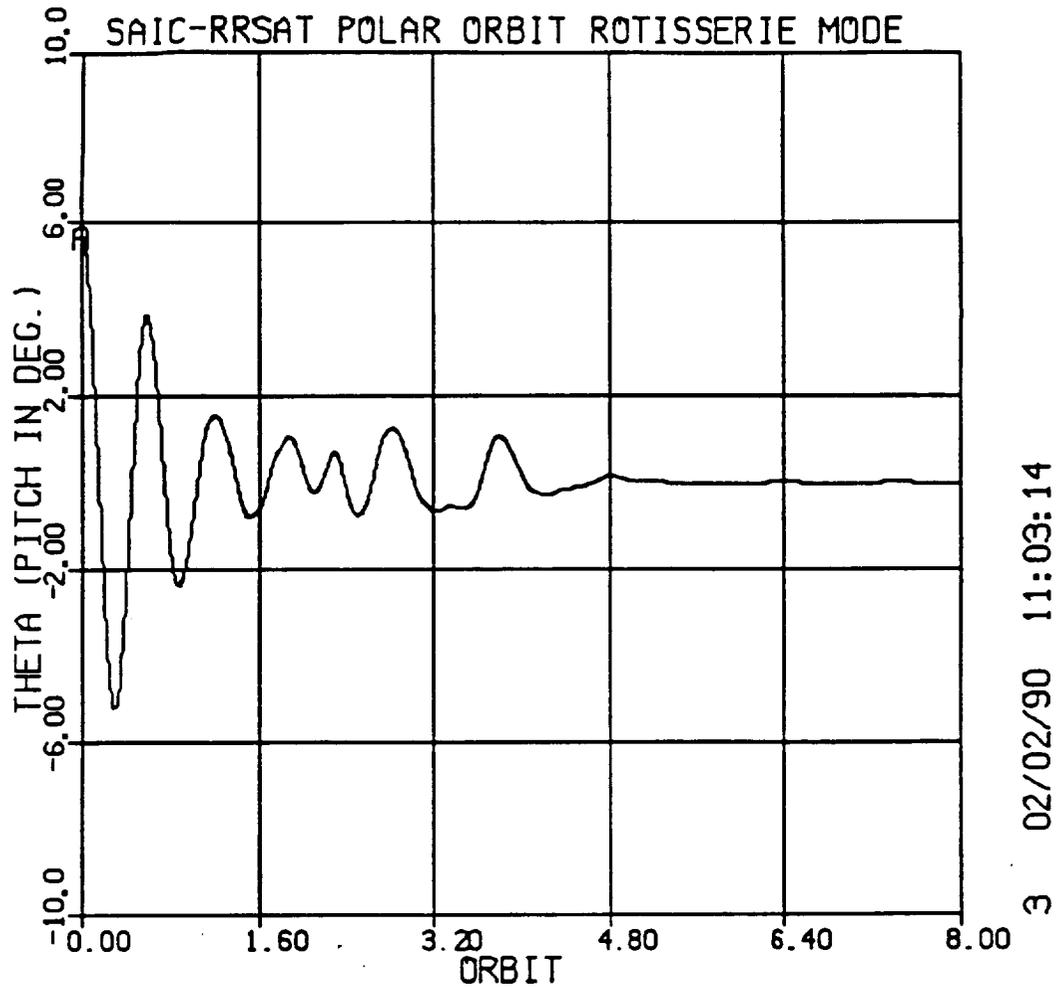
FIGURE 4.0-11
PITCH AXIS ATTITUDE ERROR FOR ROTISSERIE MODE
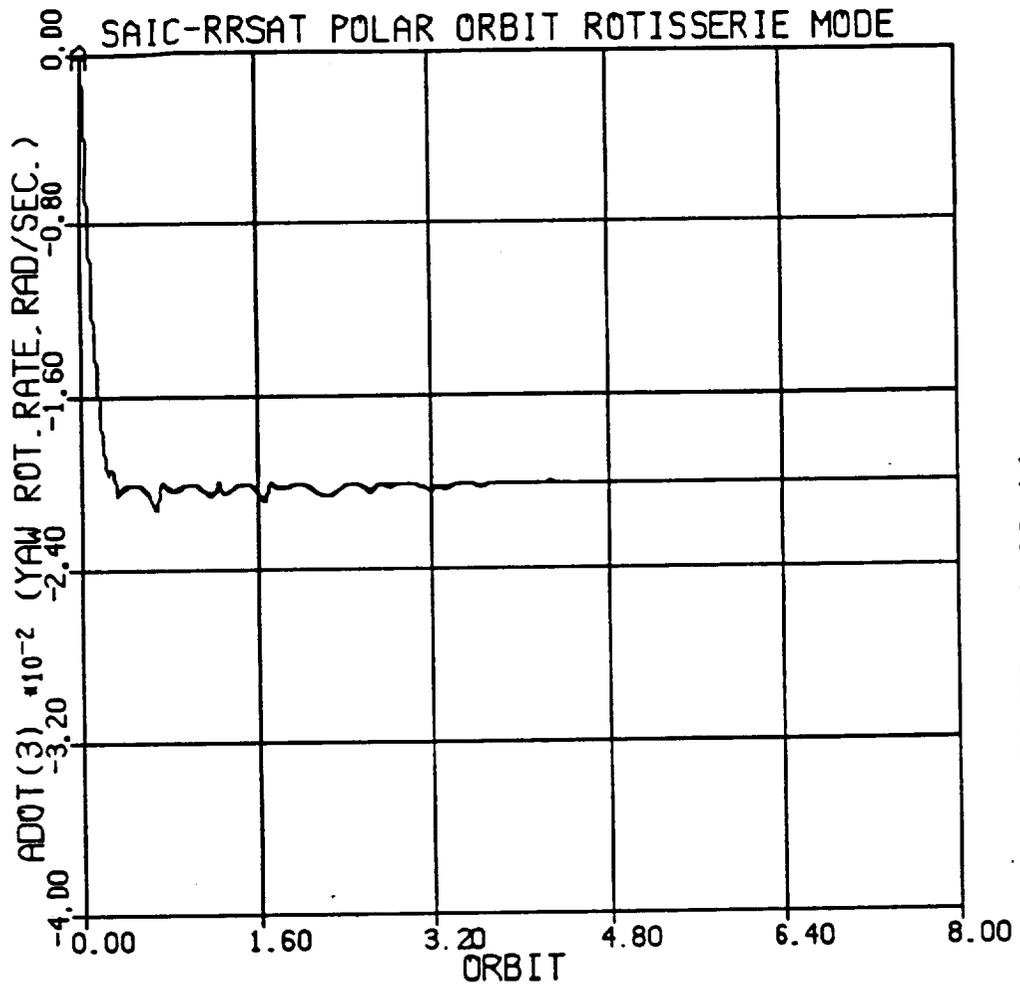
FIGURE 4.0-12
YAW AXIS ROTATION RATE FOR ROTISSERIE MODE
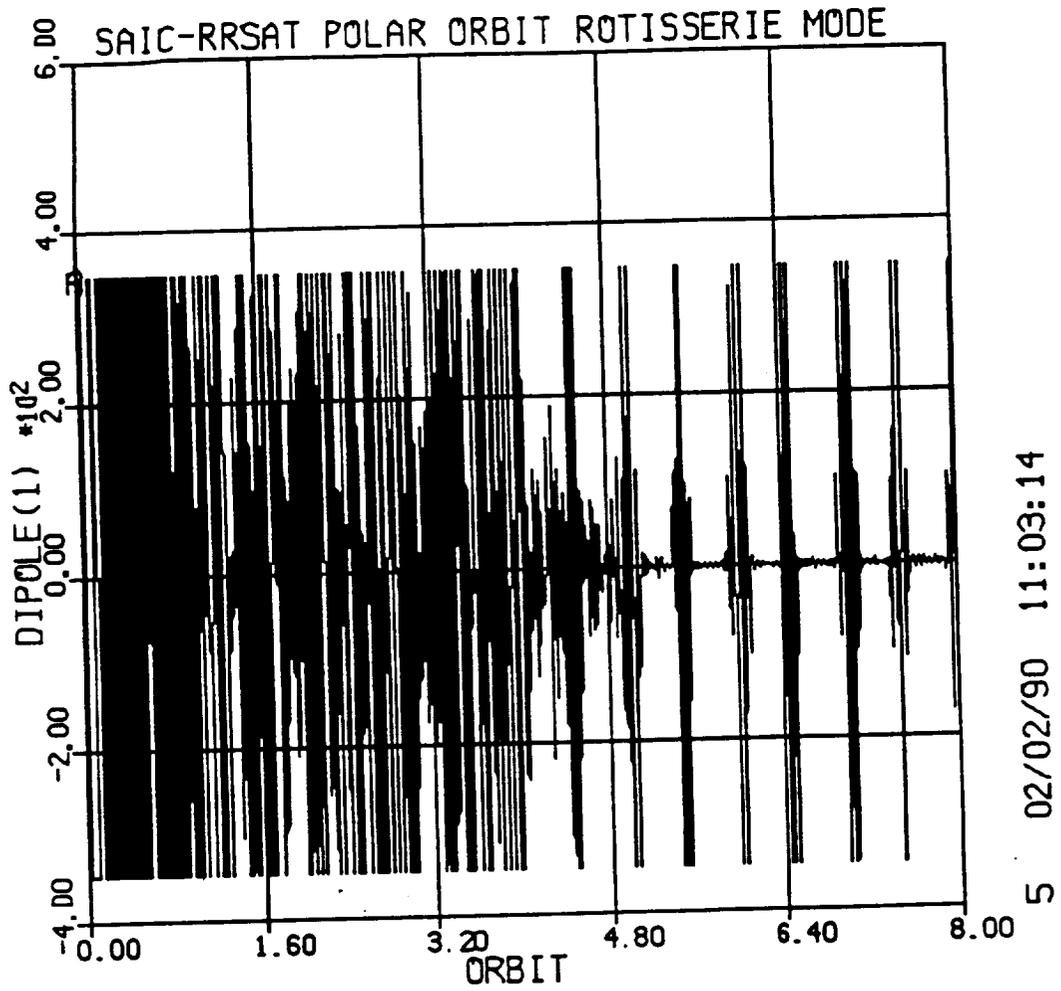
FIGURE 4.0-13
ROLL AXIS DIPOLE FOR ROTISSERIE MODE
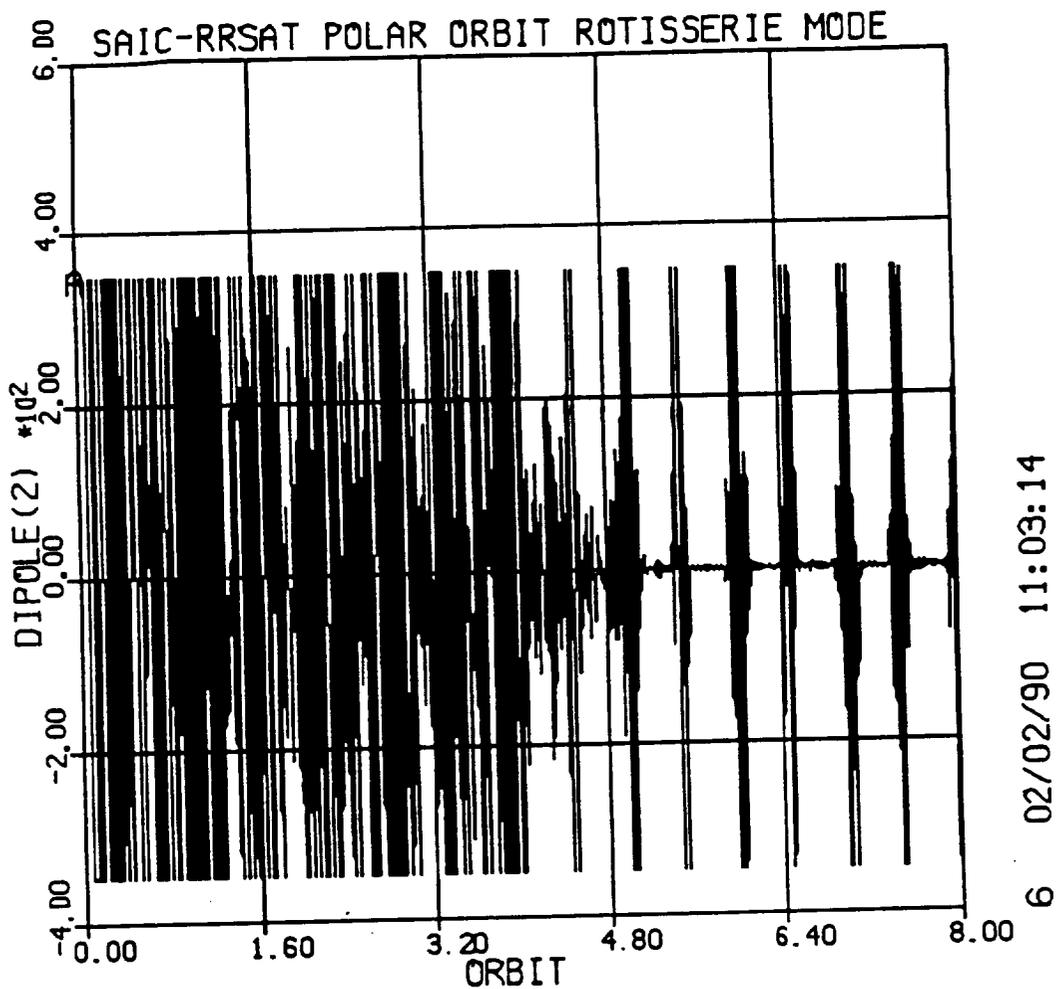
FIGURE 4.0-14
PITCH AXIS DIPOLE FOR ROTISSERIE MODE
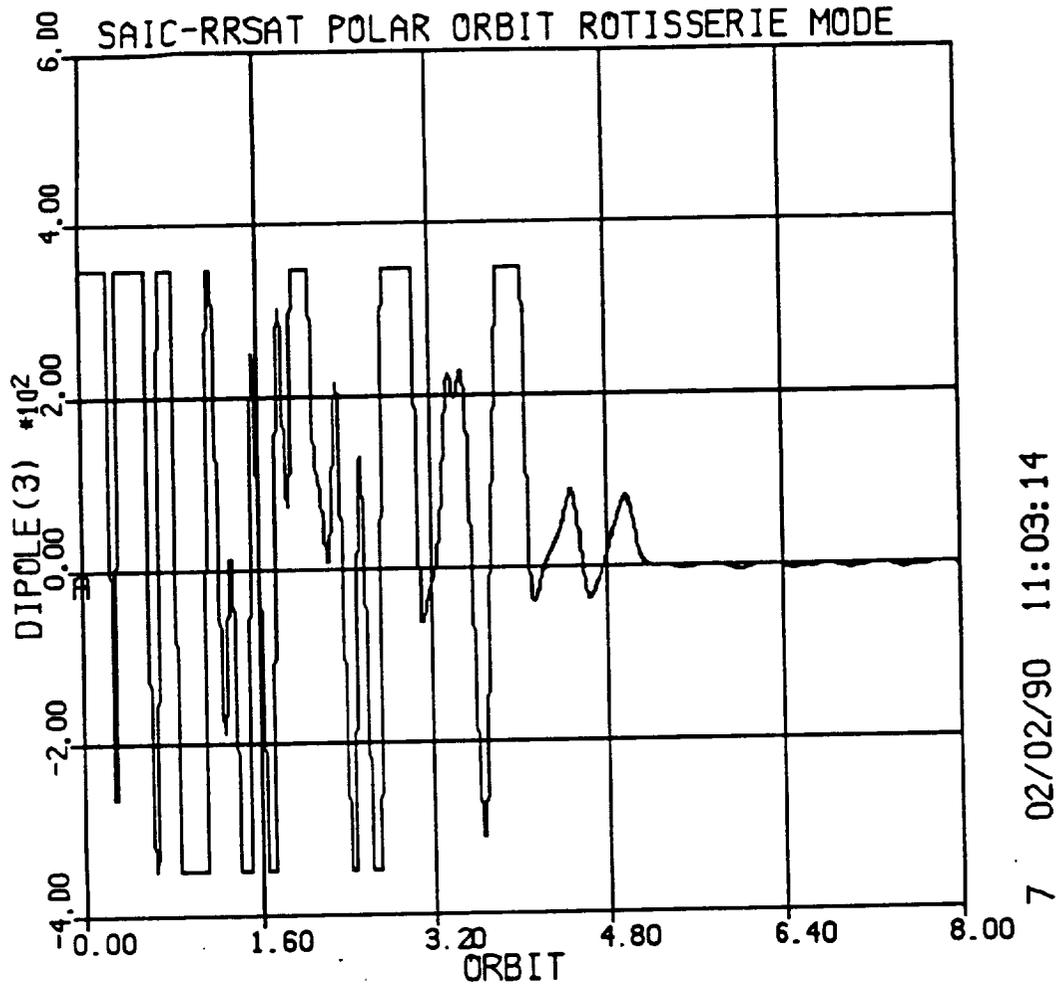
FIGURE 4.0-15
YAW AXIS DIPOLE FOR ROTISSERIE MODE
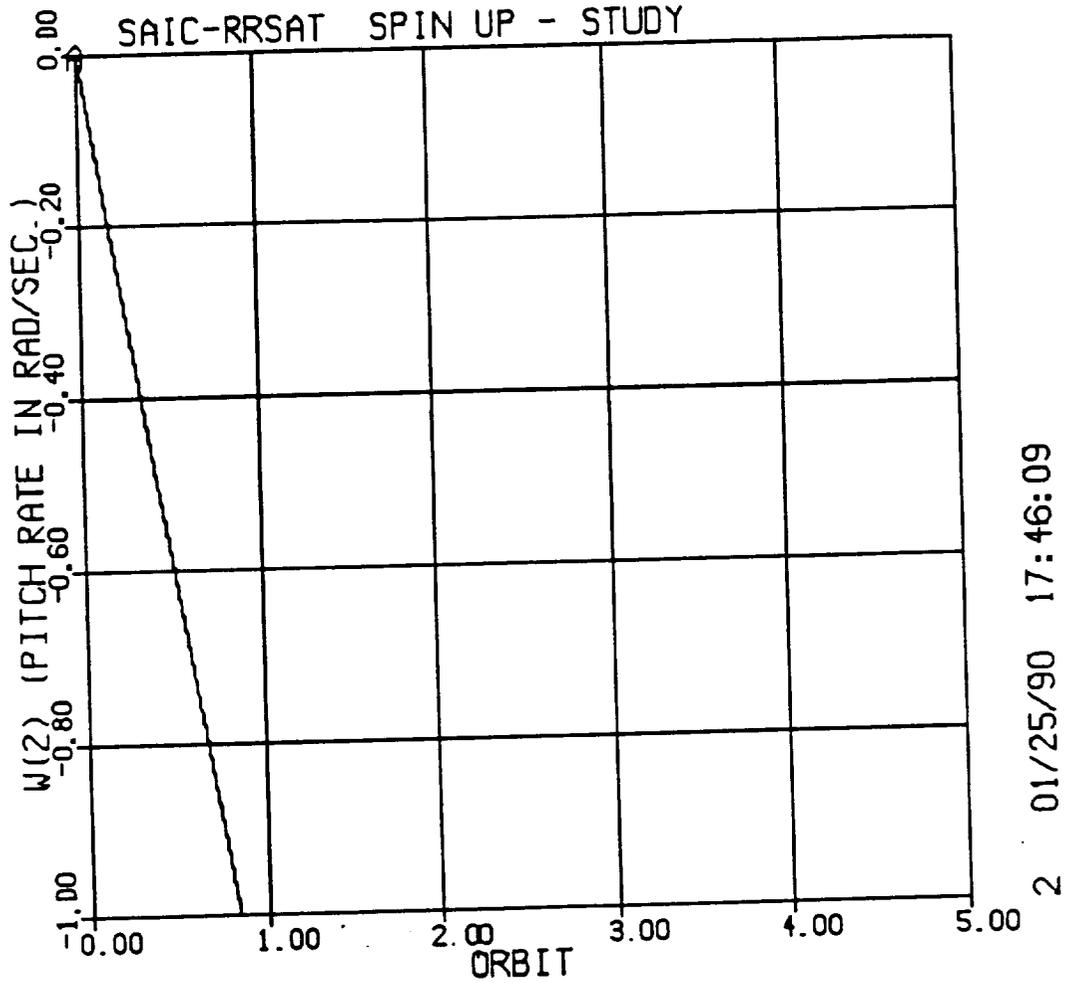
FIGURE 4.0-16
PITCH AXIS SPIN RATE HISTORY FOR SPIN MODE
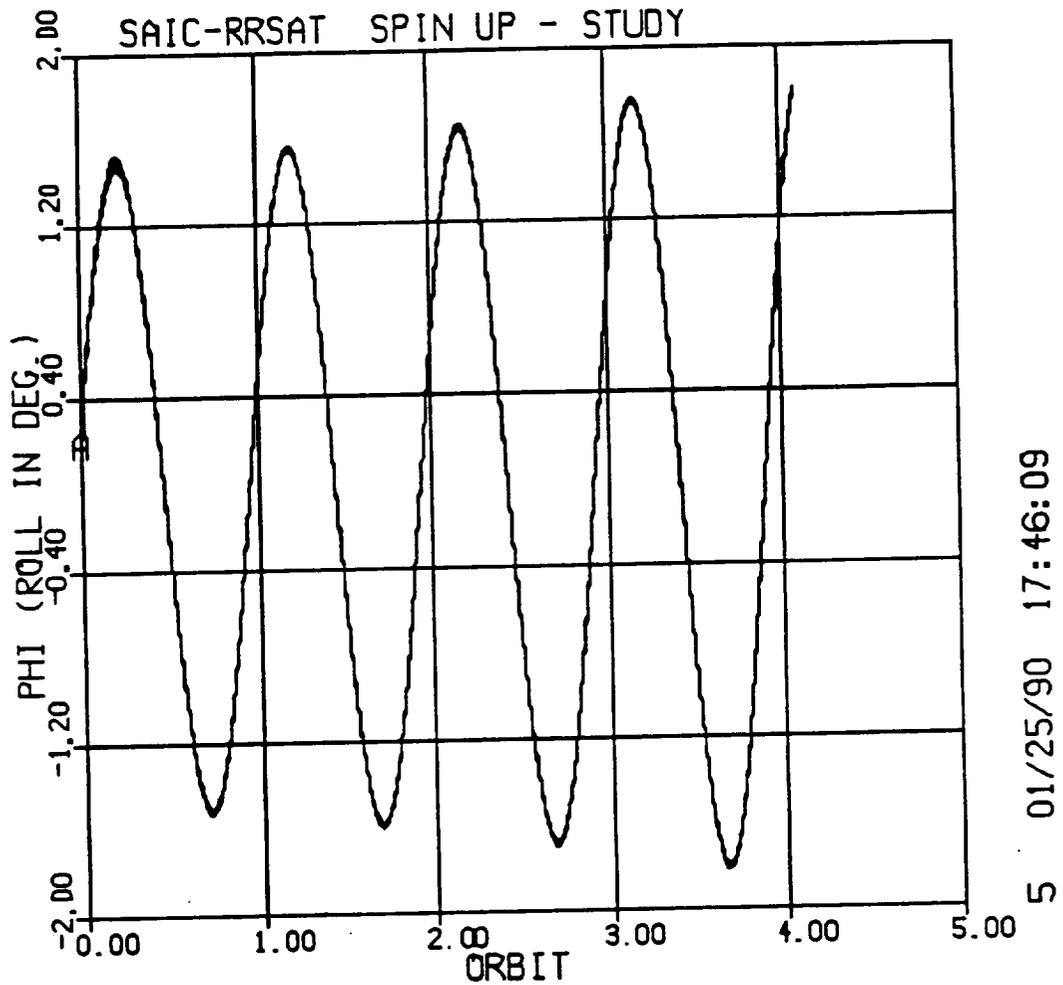
FIGURE 4.0-17
ROLL AXIS ATTITUDE ERROR FOR SPIN MODE
(NO ACTIVE ROLL ERROR CONTROL)
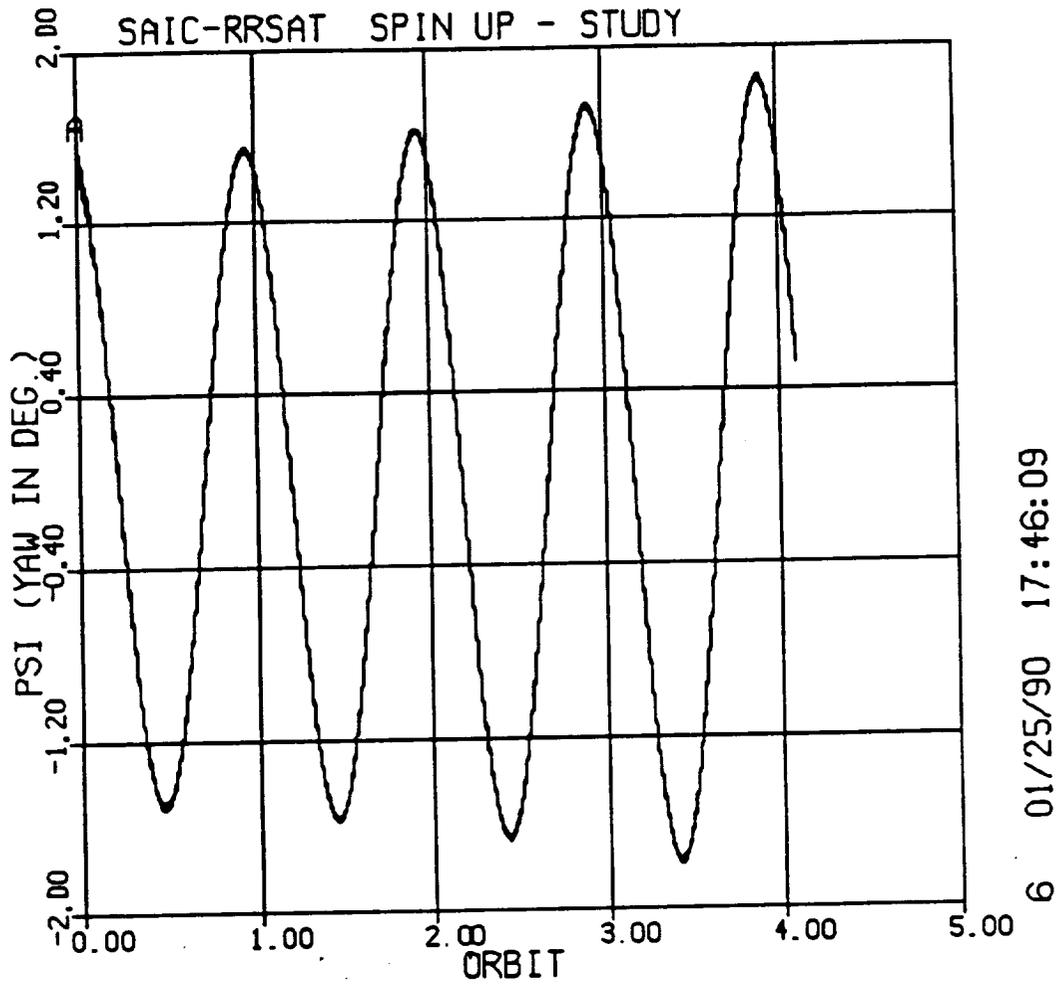
FIGURE 4.0-18
YAW AXIS ATTITUDE ERROR FOR SPIN MODE
(NO ACTIVE YAW ERROR CONTROL)

5.0 HARDWARE COMPONENT SELECTION

In order to meet the ACS design goals of low mass and minimum power, various combinations of equipment and components have been investigated and reviewed. ITHACO has designed and manufactured magnetometers, TORQRODs, momentum wheels, Earth horizon sensors, and the associated electronics for numerous spacecraft.

5.1   Hardware Description

Only components developed and built by ITHACO have been examined in order to define the overall ACS mass and power requirements. ITHACO has become the leader in the area of attitude determination and control systems for small spacecraft, where mass and power are at a premium. The proposed ITHACO components have been developed for the specific purpose of satisfying low mass and minimum power requirements.

5.1.1 Magnetometer

The magnetometer is a two-axis flux gate instrument, which provides magnetic field measurements over the range of ±1 gauss. The unit dimensions are 7.6(l) x 4.1(w) x 3.0(h) centimeters, has a mass of 132 grams, and consumes less 25 milliwatts at 5 volts. The ITHACO magnetometer was designed and develop specifically for spacecraft with low power. The first flights of these magnetometers will be on DARPA's Special Communications Satellite Clusters (SCSC) spacecraft.

5.1.2 TORQROD

ITHACO TORQRODs have been used on over 60 satellites during the past 25 years with no known failures. Among the programs for which we have supplied TORQRODs are SATCOM, GSTAR, SPACENET, ANIK-E, and LANDSAT.

The TORQROD proposed for the various ACS modes is a single design which provides a maximum dipole level of 350 Am^2. Each TORQROD consists of a magnetic core with two sets of coils, each of which will provide the required dipole. Current flowing through the coil generates a dipole moment capable of interacting with an external magnetic field to produce a torque. All three TORQRODs are identical in design, with a length of 91.4 centimeters (36 inches) and a diameter of 2.8 centimeters (1.1 inches). Power is supplied to the TORQRODs directly from the batteries.

5.1.3 T-SCANWHEEL

The momentum wheel proposed for the RRS is a growth version of the recently developed T-SCANWHEEL, with a mass of 6 kg (13.2 lb) and an average power consumption of 1 watt at a speed of 1000 rpm (excluding the associated electronics). The wheel has a diameter of 25 centimeters (10 inches) and a height of 10 centimeters (4 inches). In addition, an Earth horizon sensor can be integrated with the wheel to provide spacecraft attitude data. The first production model of the momentum wheel will be flown on the German Center for Applied Space Technology and Microgravity (ZARM) BREMSAT spacecraft, scheduled to be launched in 1991.

Attitude information is provided by the optical portion of the T-SCANWHEEL. The rotation of the T-SCANWHEEL causes a mirror in the optical system to scan out a cone in space looking for infrared energy from the Earth. The optics directs infrared light reflected from the scanning mirror onto a detector. The detector signal is amplified and fed to the signal processor which provides data for roll and pitch attitude determination and error correction. The T-SCANWHEEL also provides tachometer and rotor position information.

5.1.3 T-SCANWHEEL (Cont'd)

The rotation axis of the T-SCANWHEEL is directed away from the spacecraft to afford the scanning mirror a clear FOV. For maximum attitude coverage the scan cone should be free of obstructions, however portions of the scan can be electronically blanked.

5.1.4 Earth Horizon Sensor

The infrared horizon sensor detects the thermal discontinuity between Earth and space as the line of sight crosses the horizon. As mounted on the T-SCANWHEEL, the sensor FOV is scanned by the rotating mirror into a ±45 degree cone, with an optical passband of 14 to 16 microns. As the cone intercepts the Earth, measurements of the chord and phase are accomplished, which can be converted into attitude errors. Pitch information is derived by comparing the timing of the detector signal with the rotor position information from the T-SCANWHEEL. Roll information is derived from measurements of the percentage of the scan that intercepts the Earth. The sensor has dimensions of 25(l) x 7.5(w) x 12.5(h) centimeters, has a mass of 1 kilogram, and consumes less than 100 milliwatts.

5.1.5 ACS Electronics

The ACS electronics box is a multiple module assembly, compiled for each mission. Based upon ITHACO's modular 15.2 x 15.2 centimeter frame, the electronics box receives power and commands from the spacecraft. It operates the T-SCANWHEELs and TORQRODs, closes the control loops, and provides the necessary telemetry for operation and monitoring of the spacecraft.

The signal processor modules (or cards) in the electronics box develop the amplified detector signals into logic compatible pulses that can be used by the attitude computer cards. Motor driver cards operate the wheel motors in response to signals from the interface card. A minimum speed loop insures that the wheels will always rotate at an appropriate rate so that attitude data will be available to operate the ACS. Tachometer and position information is provided. The attitude computers derive pitch and roll error signals from data provided by the signal processors and the motor drivers.

A magnetometer interface card receives the magnetometer data and prepares it for use in the rest of the ACS. A magnetic trim capability allows cross talk effects between the TORQRODs and the magnetometer to be removed. The vector components are processed into field and field rate signals and provided to the rest of the ACS. The TORQROD driver card drives each TORQROD coil current in response to control signals developed either on that card or on other cards in the ACS. Analog bias input ports allow ground control of each TORQROD for magnetic balancing purposes.

Finally, mission specific cards may be added to the electronics box, as necessary. Special functions such as thruster drivers, additional reaction or momentum wheels, experiment interfaces, etc, can be accommodated by the addition of special function cards to the basic ACS.

If a spacecraft host microprocessor is available, many of the functions of the ACS electronics package can be incorporated as digital logic, allowing some of the electronics cards to be removed or simplified. The exact modifications depend upon the specific microprocessor employed.

5.2   Hardware Requirements

Individual components have been selected to satisfy the requirements defined by each attitude control mode. A single ACS configuration is also defined which incorporates the equipment required to accommodate all the operational modes.

5.2.1   Gravity Gradient Stabilized Mode

For this ACS, two dual-axis magnetometers are used as the control sensors. Attitude information is estimated from measurements of the Earth's magnetic field via Kalman filter algorithms incorporated into a host microprocessor. This attitude data is evaluated via microprocessor based control algorithms, which provide the necessary activation signals to the appropriate combination of TORQRODs, one of which is aligned along each of the three spacecraft axes. The size, mass, and power summaries for this ACS are shown in Table 5.2.1-1.

TABLE 5.2.1-1
Size, Mass, and Power Summary For
Three-Axis Gravity Gradient Stabilized Mode

Component	Dimensions (cms)	Mass (Kg(lb))	Power (watts) Ave	Peak
Magnetometer A	7.6 x 4.1 x 3.0	0.13 (0.3)	0.03	0.03
Magnetometer B	7.6 x 4.1 x 3.0	0.13 (0.3)	0.03	0.03
350 Am^2 TQR A	2.8 d x 91.4	4.1 (9.1)	0.9	11.4
350 Am^2 TQR B	2.8 d x 91.4	4.1 (9.1)	0.9	11.4
350 Am^2 TQR C	2.8 d x 91.4	4.1 (9.1)	0.9	11.4
ACS Electronics				
- Magnetometer I/F*		0.64 (1.4)	1.5	1.5
- Power Supply**		0.64 (1.4)	1.14	1.14
- TQR Controller		0.64 (1.4)	1.5	1.5
- Subsystem I/F		0.64 (1.4)	1.5	1.5
Total	16.8 x 17.0 x 12.4	2.56 (5.6)	5.64	5.64
Subsystem Total		15.12 (33.5)	8.40	39.90

Notes: Microprocessor Required For MADS
     *   Could be replaced by direct interface to microprocessor
    ** Could be deleted if external power supply available

5.2.2 Gravity Gradient Stabilized With Yaw Rotation (Rotisserie) Mode

This control technique is identical to that described in Section 5.2.1, but with the addition of a two momentum wheels, only one of which is required. The spin axis of each wheel is aligned along the yaw (Z) axis of the spacecraft and the wheel speed is constant. The momentum generated by the rotation of the spacecraft about its yaw axis is nullified by the momentum produced by one wheel. Thus, this concept maintains a zero-momentum RRS.

With the incorporation of the two wheels, the ACS electronics is increased by three cards - one for each motor driver and one to provide the control law interface between the wheels and the spacecraft. It should be noted that the peak wheel power of 16 watts is required only during the initial spin up of the wheel when the maximum acceleration torque is demanded.
The size, mass, and power summaries for this ACS are shown in Table 5.2.2-1.

TABLE 5.2.2-1
Size, Mass, and Power Summary For
Gravity Gradient Stabilized Mode With Yaw Rotation

Component	Dimensions (cms)	Mass (Kg(lb))		Power (watts) Ave	Peak
Magnetometer A	7.6 x 4.1 x 3.0	0.13	(0.3)	0.03	0.03
Magnetometer B	7.6 x 4.1 x 3.0	0.13	(0.3)	0.03	0.03
350 Am^2 TQR A	2.8 d x 91.4	4.1	(9.1)	0.9	11.4
350 Am^2 TQR B	2.8 d x 91.4	4.1	(9.1)	0.9	11.4
350 Am^2 TQR C	2.8 d x 91.4	4.1	(9.1)	0.9	11.4
Momentum Wheel D	25.0 d x 10.0	6.0	(13.2)	1.0+	16.0++
Momentum Wheel C	25.0 d x 10.0	6.0	(13.2)	1.0+	16.0++
ACS Electronics					
- Magnetometer I/F*		0.64	(1.4)	1.5	1.5
- Power Supply**		0.64	(1.4)	2.39	2.39
- TQR Controller		0.64	(1.4)	1.5	1.5
- Motor Driver C		0.64	(1.4)	2.0	2.0
- Motor Driver D		0.64	(1.4)	2.0	2.0
- Control Law I/F*		0.64	(1.4)	1.0	1.0
- Subsystem I/F		0.64	(1.4)	1.5	1.5
Total	16.8 x 17.0 x 21.7	4.48	(9.8)	11.89	11.89
Subsystem Total		29.04	(64.10)	16.65	78.15

Notes: Microprocessor Required For MADS
    +   Average Wheel Power @ 1000 rpm
    ++ Peak Wheel Power (including motor driver) = 18.0 watts (full torque of 0.02 N-m
          (2.8 oz-in) @ 2000 rpm)
    *   Could be replaced by direct interface to microprocessor
    ** Could be deleted if external power supply available

5.2.3 Spin Stabilized Mode

For this mode of operation, the spin up, despin, and attitude control actuators are the 2.2 Newton (0.5 pound force) thrusters. Two dual-axis magnetometers are used as the initial attitude determination and control sensors. Since the roll (X) and pitch (Y) inertias are expected to be identical, two additional momentum wheels (for redundancy) are incorporated, with the spin axis of each wheel aligned along the pitch axis of the spacecraft. Prior to the initiation of the spin up maneuver, either or both wheels are run to a constant speed to provide additional pitch momentum and to insure that the pitch axis is the major inertia axis.

Once a wheel has been spun up, improved attitude sensing is provided by an Earth horizon sensor incorporated with the wheel. This attitude data is evaluated via either analog or digital control algorithms, which provide the necessary activation signals to the appropriate combination of thrusters. As indicated in Table 5.2.3-1, additional mass and power reductions could be achieved for the ACS if appropriate options are available via other spacecraft subsystems.

With respect to the peak power consumption of the wheel, it should be emphasized that this is only required when the maximum torque of 0.02 N-m (2.8 oz-in) is applied with a wheel speed initially at 2000 rpm. Power requirements of this magnitude are only expected during the initial wheel spin up. The average power consumption of 1 watt at 1000 rpm is expected to be typical during the mission life.

TABLE 5.2.3-1
Size, Mass, and Power Summary For Spin Stabilized Mode

Component	Dimensions (cms)	Mass (Kg(lb))		Power (watts) Ave	Peak
Magnetometer A	7.6 x 4.1 x 3.0	0.13	(0.3)	0.03	0.03
Magnetometer B	7.6 x 4.1 x 3.0	0.13	(0.3)	0.03	0.03
Momentum Wheel A	25.0 d x 10.0	6.0	(13.2)	1.0+	16.0++
Sensor Module A	25.0 x 7.5 x 12.5	1.0	(2.2)	0.1	0.1
Momentum Wheel B	25.0 d x 10.0	6.0	(13.2)	1.0+	16.0++
Sensor Module B	25.0 x 7.5 x 12.5	1.0	(2.2)	0.1	0.1
ACS Electronics					
- Magnetometer I/F*		0.64	(1.4)	1.5	1.5
- Power Supply**		0.64	(1.4)	3.19	3.19
- Thruster I/F*		0.64	(1.4)	1.5	1.5
- Motor Driver A		0.64	(1.4)	2.0	2.0
- Signal Processor A		0.64	(1.4)	1.0	1.0
- Attitude Computer A*		0.64	(1.4)	0.5	0.5
- Motor Driver B		0.64	(1.4)	2.0	2.0
- Signal Processor B		0.64	(1.4)	1.0	1.0
- Attitude Computer B*		0.64	(1.4)	0.5	0.5
- Control Law I/F*		0.64	(1.4)	1.0	1.0
- Subsystem I/F		0.64	(1.4)	1.5	1.5
Total	16.8 x 17.0 x 34.1	7.04	(15.4)	15.69	15.69
Subsystem Total		21.30	(46.80)	17.95	47.95

Notes: Microprocessor Required For MADS
+ Average Wheel Power @ 1000 rpm
++ Peak Wheel Power (including motor driver) = 18.0 watts (full torque of 0.02 N-m (2.8 oz-in) @ 2000 rpm)
* Could be replaced by direct interface to microprocessor
** Could be deleted if external power supply available

5.2.4 Multi-Mode Mission

The baseline RRS is to be capable of accommodating any and all ACS modes of operation during any one mission sequence. Consequently all necessary attitude determination and control components must be available for each mission. The size and mass requirements for a complete complement of ACS components are summarized in Table 5.2.4-1. The power consumption requirements are dependent upon the specific operational ACS mode, and are defined in Tables 5.2.1-1 thru 5.2.3-1.

TABLE 5.2.4-1
Size and Mass Summary For
Multi-Mode Mission

Component	Dimensions (cms)	Mass (Kg(lb))	
Magnetometer A	7.6 x 4.1 x 3.0	0.13	(0.3)
Magnetometer B	7.6 x 4.1 x 3.0	0.13	(0.3)
350 Am^2 TQR A	2.8 d x 91.4	4.1	(9.1)
350 Am^2 TQR B	2.8 d x 91.4	4.1	(9.1)
350 Am^2 TQR C	2.8 d x 91.4	4.1	(9.1)
Momentum Wheel A	25.0 d x 10.0	6.0	(13.2)
Sensor Module A	25.0 x 7.5 x 12.5	1.0	(2.2)
Momentum Wheel B	25.0 d x 10.0	6.0	(13.2)
Sensor Module B	25.0 x 7.5 x 12.5	1.0	(2.2)
Momentum Wheel C	25.0 d x 10.0	6.0	(13.2)
Momentum Wheel D	25.0 d x 10.0	6.0	(13.2)
ACS Electronics			
- Magnetometer I/F*		0.64	(1.4)
- Power Supply**		0.64	(1.4)
- TQR Controller		0.64	(1.4)
- Motor Driver A		0.64	(1.4)
- Signal Processor A		0.64	(1.4)
- Attitude Computer A*		0.64	(1.4)
- Motor Driver B		0.64	(1.4)
- Signal Processor B		0.64	(1.4)
- Attitude Computer B*		0.64	(1.4)
- Motor Driver C		0.64	(1.4)
- Motor Driver D		0.64	(1.4)
- Control Law I/F*		0.64	(1.4)
- Subsystem I/F		0.64	(1.4)
- Thruster I/F		0.64	(1.4)
Total	16.8 x 17.0 x 43.4	8.96	(19.60)
Subsystem Total		47.52	(104.70)

Notes: Microprocessor Required For MADS
  * Could be replaced by direct interface to microprocessor
  ** Could be deleted if external power supply available

6.0 CONCLUSIONS

Preliminary control laws and estimates of the mass and power requirements for an ACS to be used on the Re-Usable Re-entry Satellite have been determined. Three basic operational scenarios have been defined: a three-axis GG stabilized mode, a GG mode with a yaw rotation, and a spin stabilized mode. The proposed ACS also accomplishes the initial acquisition and orientation of the RRS after separation from the launch vehicle. In addition, the ACS properly orients the RRS and provides initial attitude data for the re-entry sequence.

Table 6.0-1 provides a complete summary of the attitude control and knowledge performance capability associated with each of the ACS scenarios examined. The components which are identified for the baseline power and mass trade-offs in this study, are standard ITHACO products which can be provided with little or no non-recurring costs. ITHACO has the ability and the experience to work with the customer in providing an ACS which optimally satisfies all the mass, power, and cost constraints.

As stated in Section 4.0, the analysis of the spin stabilized mode of operation has not been finalized. The roll and yaw attitude errors depicted in Figure 4.0-17 and 4.0-18 are slowly increasing and thus, thrusters are required for auxiliary control. It is anticipated that the following scenario will be satisfactory:

Roll Control - Pulse appropriate thruster if $| \varphi | > \varepsilon_\varphi$

Yaw Control - Quarter-orbit roll/yaw interchange

Spin Control - Pulse appropriate thruster if $| \Delta\omega | \geq \varepsilon_\omega$

where ε_φ and ε_ω are the roll error and spin rate thresholds, respectively. As noted in Table 6.0-1, the attitude control performance for the spin stabilized mode is what is expected, based upon previous experience with this control technique. Additional analyses are required to confirm these expectations and to establish the fuel requirements necessary for the mission.

TABLE 6.0-1
RRS ADACS Performance Summary

Operational Mode	Attitude Control**			Attitude Knowledge		
	Roll (deg)	Pitch (deg)	Yaw (deg)	Roll (deg)	Pitch (deg)	Yaw (deg)
GG Stable	0.1 (0.0)+	0.1 (0.3)+	0.4 (1.2)+	0.4	0.9	1.0*
Yaw Spin	0.1	0.1	---	0.4	0.9	---
Spin Stable	0.1++	---	0.3++	0.2	---	0.5

Note: *   Improved MADS 3-sigma accuracy
        **   Assumes perfect attitude knowledge
        +   Attitude offset
        ++   Expected