N91-17095

On-board Attitude Determination and Control Algorithms for SAMPEX

Thomas W. Flatley
Josephine K. Forden
Debra A. Henretty
E. Glenn Lightsey
F. Landis Markley

Guidance and Control Branch
NASA/Goddard Space Flight Center

Algorithms for onboard attitude determination and control of the Solar, Anomalous, and Magnetospheric Particle Explorer (SAMPEX) have been developed. The algorithms include: spacecraft ephemeris and geomagnetic field models, attitude determination with 2 degree accuracy, control of pitch axis pointing to the sun and yaw axis pointing away from the earth to achieve control of pitch axis within 5 degrees of sunline, momentum unloading, and nutation damping. The closed loop simulations were performed on a VAX 8830 using a prototype version of the on-board software.

I. INTRODUCTION

The Solar, Anomalous, and Magnetospheric Particle Explorer (SAMPEX), scheduled for launch in June 1992, is the first in the Small Explorer (SMEX) series of spacecraft. The SAMPEX software algorithm design philosophy is to develop the common aspects of the attitude determination and control (ADC), such as sensor data processing, attitude determination, ephemeris propagation, and command and telemetry processing, in as generic a form as possible.

SAMPEX, a 169 kg payload, will be launched into a 580 km circular orbit with 82 degrees inclination by a Scout launch vehicle. The spacecraft attitude must simultaneously satisfy a variety of on-orbit pointing requirements. The scientific instruments are located on the +z side of the spacecraft (Figure 1), and detect the impingement of magnetically aligned solar and cosmic particles. The spacecraft z-axis (instrument boresight) should therefore point within 15 degrees of local vertical (zenith) in the polar regions. Due to damaging space dust and orbital debris, the experiment boresight is required to point at least 45 degrees away from the spacecraft ram velocity vector. The spacecraft has fixed solar arrays with all cells on the +y side. Pointing of the +y axis to within 5 degrees of the sun line is desired during the entire mission. Attitude knowledge to 2 degrees or better is also required. [1]

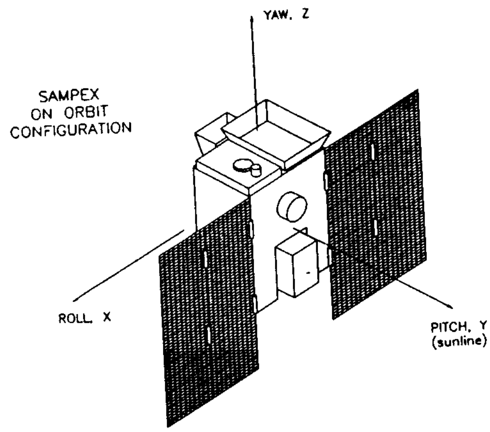

Figure 1. Spacecraft Mechanical Configuration

The ADC sensors include a two-axis fine digital sun sensor, five coarse sun sensors , and a triaxial search coil magnetometer. All sensor data are acquired and processed at the ADC sampling frequency, 2 Hz. Three magnetic torquer bars and a pitch axis momentum wheel are used as control actuators. The sensor data processing algorithms are presented in Section II.

There are two science pointing modes - vertical pointing and orbit rate rotation. The computation of the pitch error angle is different for the two modes. The on-board attitude determination algorithm is based on the sun and magnetic field data only. The rate information will be generated without gyros and the nadir vector will be obtained without earth sensors. Therefore accurate spacecraft ephemeris and magnetic field vector knowledge are essential. The detailed description of the attitude determination is given in Section III.

The vertical pointing mode minimizes the angle between the spacecraft z-axis and the zenith vector within the sun pointing constraint. However, this mode has the undesirable property of pointing the experiment boresight directly into the ram vector twice per orbit when the orbit plane is parallel with the sun vector. The orbit rate rotation mode does not maintain the instrument boresight as close to the zenith vector as the vertical pointing mode, but it satisfies the spacecraft pointing requirement as well as the velocity vector avoidance criterion. The control scheme for each of these modes is described in Section IV. The digital sun pointing only mode, which does not perform pitch axis control, is used for reacquisition and is also described in this section.

This paper also discusses the attitude dynamic simulator and results used for analysis in Section V, and presents a summary in Section VI.

II. SENSOR DATA PROCESSING

Sun Sensor Data Processing

The fine sun sensor (FSS) is used for on-board attitude determination and sun pointing. It is a two-axis sun sensor with a field of view of ± 64 degrees. It outputs 8 bits of data in gray code for each axis with a resolution of 0.5 degrees. After the gray coded data is converted to binary, the two binary counts (N_a and N_b) are converted into coordinates of the sun's image within the sensor, i.e.

$$x = 0.002754\ N_a - 0.350625$$
$$z = 0.002754\ N_b - 0.350625$$

A sun vector (s) in the spacecraft body frame is computed using the following equations [2]:

$$s = \frac{1}{\sqrt{x^2 + z^2 + t^2}} \begin{bmatrix} -n\,x \\ \sqrt{t^2 - (n^2 - 1)\ (x^2 + z^2)} \\ -n\,z \end{bmatrix}$$

Where: n - the refraction index of the FSS glass
t - the glass thickness (cm)
x, z - measurements in the sensor frame (cm)

The above equations are derived based on the assumptions : 1) the sensor frame x, y, z is aligned with the body frame x, y, z axes respectively, 2) the misalignment between sensor frame and the body frame is within tolerance, 3) pitch axis is normal to FSS. However, the FSS algorithm has means to correct 90 degree and 180 degree misalignment of the x, z axes.

The coarse sun sensor (CSS) is used for initial acquisition, reacquisition, and on-board attitude determination when the sun is out of the field of view of the FSS. The CSS system consists of five eyes: one pitch eye mounted on the center of the spacecraft facing the negative pitch(y) axis, two roll(x) eyes and two yaw(z) eyes. Each pair is mounted on the solar array panels facing 180 degrees apart along the axis. The roll and yaw eye outputs are differenced. All outputs are converted to 12 bit words. When divided by the maximum expected output these signals represent the x, y, and z components of the sun vector in body frame. When the negative pitch eye is illuminated, the normalized coarse sun vector (s_c) in body frame is given by the following equation:

$$s_c = \frac{1}{\sqrt{x^2 + y^2 + z^2}} \begin{bmatrix} x \\ -y \\ z \end{bmatrix}$$

and when the negative pitch eye is not illuminated, s_c is given by the following equation:

$$s_c = \begin{bmatrix} x \\ \sqrt{1 - x^2 - z^2} \\ z \end{bmatrix}$$

when $x^2+z^2 >= 1$, the above equation will become:

$$s_c = \frac{1}{\sqrt{x^2 + z^2}} \begin{bmatrix} x \\ 0 \\ z \end{bmatrix}$$

Magnetometer Data Processing

The triaxial search coil magnetometer is used for on-board attitude determination and momentum management (sun pointing and momentum unloading). It outputs 12 bit words for each axis. Its resolution is 0.3125×10^{-7} tesla and its range is $\pm\ 640 \times 10^{-7}$ tesla. The earth's magnetic field in the body frame is computed using the following equation [2]:

$$\mathbf{B} = [\mathbf{Scale}]\mathbf{V} + \mathbf{Bias} - [\mathbf{C}]\mathbf{dm}$$

where: **B** - magnetic field vector (tesla)

[Scale] - scale factor matrix with all off-diagonal elements being zero (tesla/volts)

V - magnetometer reading (volts)

Bias - constant magnetic field bias (tesla)

[C] - torquer bar & magnetometer coupling matrix (tesla/A-m^2)

dm - dipole moment of torque rods computed from the torque magnetic assembly current feedback (A-m^2).

The SAMPEX algorithm contains a procedure to calibrate the torque-rod/magnetometer coupling matrix ([C]). This procedure is excited periodically via ground command. It turns off all torquer bars first and obtains the uncontaminated magnetometer readings ($\mathbf{B_o}$). Then it sequentially turns on each torquer bar with a 10 A-m^2 excitation and obtains the contaminated magnetometer reading ($\mathbf{B_c}$). Finally [C] is computed using the following equation:

$$\mathbf{B_c} = \mathbf{B_o} + [\mathbf{C}]\mathbf{dm}.$$

This technique enables attitude determination to be performed in the presence of significant contamination of the magnetometer signal during magnetic actuation.

Momentum Wheel Speed Processing

The momentum wheel is used for pitch error angle control. The momentum wheel speed is used to generate the system momentum vector for momentum management. The processing of momentum wheel speed simply multiplies the raw tachometer signal by a scale factor and corrects it with a constant bias if necessary.

III. ATTITUDE DETERMINATION

The objective of on-board attitude determination is to produce an estimate of the inertial-to-body transformation matrix for the system, a representation of its three-axis attitude. The attitude determination process includes: generation of the spacecraft position vector, sun vector and earth's magnetic field vector in the inertial frame, construction of the transformation matrix, and computation of the pitch error angle, error rate and body angular momentum for control.

The attitude determination algorithm is disabled when the sun vector aligns with the earth's magnetic field vector.

During eclipse, when the sun information is not available, the body pitch axis is assumed to remain inertially fixed at the value given by the second row of the last inertial-to-body transformation matrix computed before entering eclipse. This vector **J** and **j**, a y-axis unit vector in the body frame, are substituted for the inertial and body sun vectors in the construction of the transformation matrix.

Figure 2 illustrates the scheme of construction of the inertial-to-body transformation matrix.

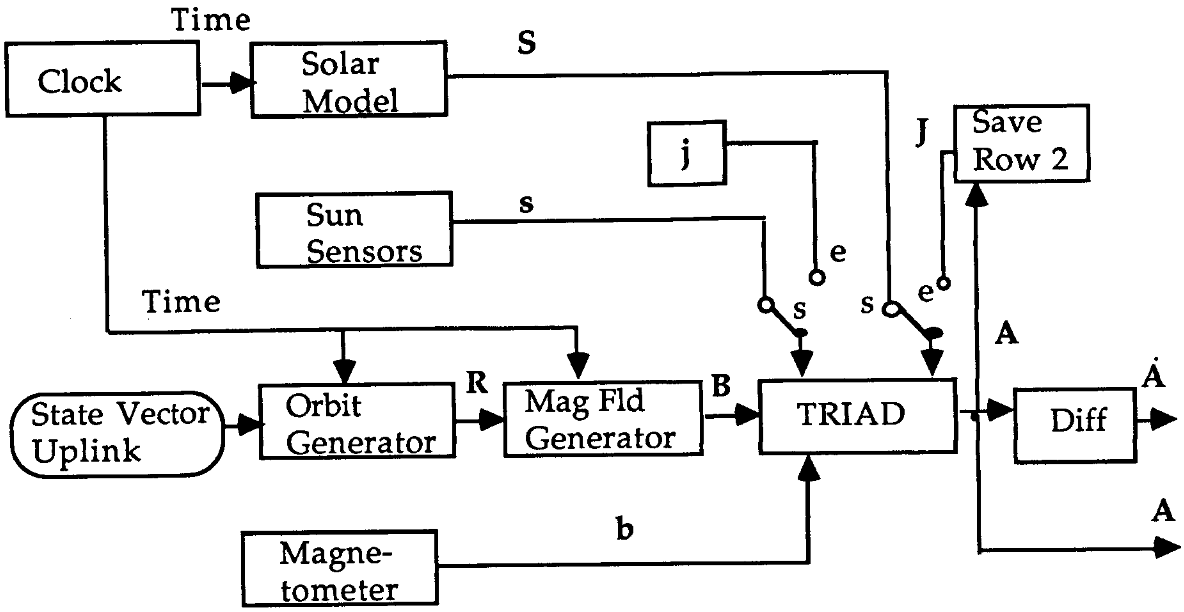

Figure 2. Construction of Inertial-To-Body Transformation Matrix

Reference Vector Generation

Although pointing requirements for SAMPEX are modest, accurate on-board spacecraft ephemeris and magnetic field vectors are required to meet the design goal of 2 degrees accuracy for on-board attitude determination.

The spacecraft ephemeris is propagated employing the fourth order Runge-Kutta integration method with integration step size equal to the ADC sampling frequency. The dynamic equation for generating the spacecraft ephemeris includes the perturbations due to the zonal gravity harmonics and the atmospheric drag [3]:

$$\ddot{\mathbf{R}} = -\frac{\mu}{|\mathbf{R}|^3}\mathbf{R} + \mathbf{a}_g + \mathbf{a}_d$$

where: $\mathbf{R}$ - spacecraft position vector (km)

μ - earth's gravitational constant, 3.9860064×10^5 (km^3/sec^2)

$|\mathbf{R}|$ - magnitude of $\mathbf{R}$ (km)

$\mathbf{a}_g$ - acceleration due to zonal gravity harmonics (km/sec^2)

$\mathbf{a}_d$ - acceleration due to atmospheric drag (km/sec^2)

The nonspherical contribution to the gravity acceleration is computed with the spherical harmonic model using zonal harmonic coefficients up to J_4.

$$\mathbf{a}_g = \frac{\mu R_e^2}{2.0\,|\mathbf{R}|^5}\begin{bmatrix} R(1)\,(f_{xy2}+f_{xy3}+f_{xy4}) \\ R(2)\,(f_{xy2}+f_{xy3}+f_{xy4}) \\ R(3)(f_{z2}+f_{z4})+f_{z3} \end{bmatrix}$$

where

$$\begin{aligned}
f_{xy2} &= 3.0\, j_2\,(5.0\, Z_r^2 - 1.0) \\
f_{xy3} &= 5.0\, j_3\,(Re/\,|\mathbf{R}|\,)\,(7.0\, Z_r^3 - 3.0 Z_r) \\
f_{xy4} &= 3.75\, j_4\,(R_e/\,|\mathbf{R}|\,)^2(21.0\, Z_r^4 - 14.0\, Z_r^2 + 1.0) \\
f_{z2} &= 3.0\, j_2\,(5.0\, Z_r^2 - 3.0) \\
f_{z3} &= j_3\,(R_e/\,|\mathbf{R}|\,)\,(Z_r\,(35.0\, Z_r^3 - 30.0\, Z_r) + 3.0\,|\mathbf{R}|\,) \\
f_{z4} &= 1.25\, j_4\,(R_e/\,|\mathbf{R}|\,)^2\,(63.0\, Z_r^4 - 70.0\, Z_r^2 + 15.0) \\
j_2 &= 1.0826271 \times 10^{-3} \\
j_3 &= -2.5358868 \times 10^{-6} \\
j_4 &= -1.6246180 \times 10^{-6} \\
R_e &= 6.37814 \times 10^3, \text{ earth radius at equator (km)} \\
Z_r &= \frac{R(3)}{|\mathbf{R}|}
\end{aligned}$$

The drag model is

$$\mathbf{a}_d = [-\rho d(1+f)A_c/2m)\;|\dot{\mathbf{R}}_r|\;\dot{\mathbf{R}}_r$$

where: ρ - atmospheric density at the spacecraft's altitude computed using a Jacchia-Roberts density model (kg/km^3), [4]

d - drag coefficient (nominally 2.2)

f - drag scale factor

A_c - cross sectional area of spacecraft perpendicular to the direction of motion (km^2)

m - mass of the spacecraft (kg)

$|\dot{\mathbf{R}}_r|$ - magnitude of $\dot{\mathbf{R}}_r$ (km/sec)

$\dot{\mathbf{R}}_r$ - velocity of the spacecraft relative to the rotating atmosphere (km/sec)

$$\dot{\mathbf{R}}_r = \begin{bmatrix} \dot{x} + \dot{\theta}\, y \\ \dot{y} - \dot{\theta}\, x \\ \dot{z} \end{bmatrix}$$

where $[\ \dot{x}\ \ \dot{y}\ \ \dot{z}\ ]^T$ is the inertial velocity of the spacecraft and $\dot{\theta}$ is the rate of rotation of the earth ($7.2921151467 \times 10^{-5}$ rad/sec).

The state vector, drag scale factor and parameters to specify the Jacchia-Roberts model are reinitialized every 24 hours via ground command. They are generated by Flight Dynamic Support System.

The earth's magnetic field is generated using the spherical harmonic model with order eight [2]. The Gaussian coefficients are updated to epoch 1985. The position of the sun is computed using a rapid analytical technique [2]. The parameters are updated to epoch 1991. Both the earth's magnetic field vector and sun vector are used to construct the transformation matrix.

Transformation Matrix Construction

The transformation matrix [A] is constructed by employing the standard TRIAD method [5]. This method only requires two body frame vectors and two reference frame vectors. The sun vector is more heavily weighted than the magnetic field vector except when the two vectors are perpendicular. The TRIAD method is given by:

$$[A] = [P]\,[Q]^T$$

where [P] and [Q] are orthogonal matrices. [P] is a matrix constructed using the sun (s) and earth's magnetic field (b) vectors in the body frame, and [Q] is a matrix constructed using the sun (S) and earth's magnetic field (B) vectors in the inertial frame:

$$\mathbf{P} = [\ \mathbf{s}\ , \frac{\mathbf{s}\times\mathbf{b}}{|\mathbf{s}\times\mathbf{b}|}, \frac{\mathbf{s}\times(\mathbf{s}\times\mathbf{b})}{|\mathbf{s}\times\mathbf{b}|}\ ]$$

$$\mathbf{Q} = [\ \mathbf{S}\ , \frac{\mathbf{S}\times\mathbf{B}}{|\mathbf{S}\times\mathbf{B}|}, \frac{\mathbf{S}\times(\mathbf{S}\times\mathbf{B})}{|\mathbf{S}\times\mathbf{B}|}\ ]$$

Computation of Angular Velocity

The angular velocity vector along with the momentum wheel speed is used to compute the system angular momentum for momentum management.

The angular velocity matrix is determined from the matrix identity:

$$\begin{bmatrix} 0 & \omega_z & -\omega_y \\ -\omega_z & 0 & \omega_x \\ \omega_y & -\omega_x & 0 \end{bmatrix} = [\dot{A}]\,[A]^T$$

where $[\dot{A}]$ is the derivative of the inertial-to-body transformation matrix [A], and is estimated by simple differencing of sequential values of [A] as decribed in the following equation :

$$[\dot{A}] = \frac{\Delta[A]}{\Delta t} = \frac{A_i - A_{i-1}}{\Delta t}$$

Where Δt is the difference between the current time and the time when the previous sample is taken. The angular velocity vector is formed using the average of the two off-diagonal elements for each of the rates ω_x, ω_y, and ω_z.

Computation of Pitch Error Angle, and Error Rate

Both the pitch error angle and error rate are used in the computation of the momentum wheel control torque.

In vertical pointing control mode, the pitch error angle is computed using the following equation:

$$e = \tan^{-1}[-r(1)/r(3)]$$

where r(1) and r(3) are x and z components of the spacecraft position vector in the body frame, respectively. This vector is obtained by transforming the spacecraft position vector in the inertial frame to the body frame by using the transformation matrix **[A]**.

In the orbit rate rotation mode, the spacecraft y-axis is desired to point at the sun while the z-axis rotates at one revolution per orbit in a plane perpendicular to the sun vector. At the same time, the z-axis is desired to point as close to north as possible at the northernmost point in the orbit, south at the southernmost point, and parallel to the equator at the equatorial crossings.

Let θ be the spacecraft orbit angle (measured from the northernmost point of the orbit), Figures 3 and 4 show the desired pointing direction for the z-axis for the two cases $\mathbf{S} \bullet \mathbf{N} > 0$ and $\mathbf{S} \bullet \mathbf{N} < 0$ respectively, where **N** is the orbit normal in the direction of $\mathbf{R} \times \dot{\mathbf{R}}$ and **S** is the inertial sun vector.

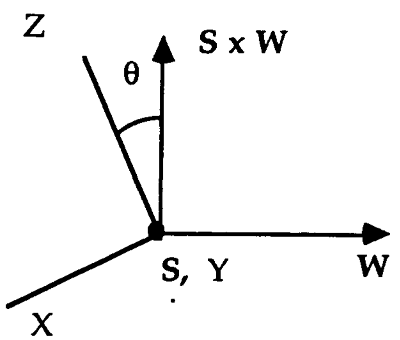

Figure 3. Desired Spacecraft Orientation for $\mathbf{S} \bullet \mathbf{N} > 0$

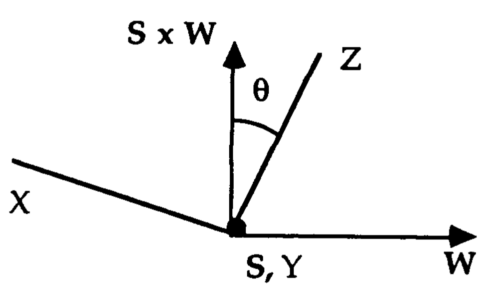

Figure 4. Desired Spacecraft Orientation for $\mathbf{S} \bullet \mathbf{N} < 0$

In these figures, **W** is a unit vector in the direction of $\mathbf{NP} \times \mathbf{S}$, where **NP** is a north pole unit vector, $[0, 0, 1]^T$. An inertial target (U) for z-axis pointing will be given by :

$$\mathbf{U} = \cos\theta\, \mathbf{S} \times \mathbf{W} + \sin\theta\, \mathbf{W} \qquad \text{for } \mathbf{S} \bullet \mathbf{N} < 0$$

$$\mathbf{U} = \cos\theta\, \mathbf{S} \times \mathbf{W} - \sin\theta\, \mathbf{W} \qquad \text{for } \mathbf{S} \bullet \mathbf{N} > 0$$

where $$\mathbf{W} = \frac{\mathbf{NP} \times \mathbf{S}}{|\mathbf{NP} \times \mathbf{S}|}$$

The sine and cosine of the orbit angle are determined from the spacecraft position (**R**) and velocity ($\dot{\mathbf{R}}$) vectors. If **AN** is a unit vector in the direction of **NP** $\times$ **N** and locates the orbit ascending node, and **NMP** is a unit vector in the direction of **N** $\times$ **AN** and locates the northernmost point, then $\sin\theta$ and $\cos\theta$ are defined by:

$$\sin\theta = -\frac{\mathbf{R} \bullet \mathbf{AN}}{|\mathbf{R}|}$$

$$\cos\theta = \frac{\mathbf{R} \bullet \mathbf{NMP}}{|\mathbf{R}|}$$

The test for the sign of the dot product of the sun and the orbit normal vectors will automatically change the spin direction when the sun passes through the orbit plane. The pitch error angle for control can be computed using:

$$e = \tan^{-1}(-u(1)/u(3))$$

where **u** is obtained by transforming the target vector **U** in the inertial frame to the body frame by using the transformation matrix **[A]**.

The pitch error rate is estimated by differencing of sequential values of pitch error angle.

IV. CONTROL LAWS

In the two science control modes, z-axis pointing is accomplished by controlling the momentum wheel speed, and sun pointing is accomplished by magnetic torquing. In addition to pointing control, the magnetic torquer bars provide nutation damping.

When the sun vector is parallel to the earth's magnetic field vector (at singularity), the momentum wheel speed is held constant at the speed before singularity occurs, and all torquer bars are turned off.

During eclipse, all magnetic torques are turned off, and momentum wheel is controlled using the normal control law except in the singular case described above.

In the sun pointing only control mode, the momentum wheel speed is held at a commandable rate to provide a momentum bias, and the magnetic torquer bars are activated for sun pointing as well as nutation damping.

Momentum Wheel Control Law for Science Pointing

The goal of controlling the momentum wheel speed is to control the motion of the spacecraft about pitch axis (y-axis), i.e. z-axis pointing control. The torque to the momentum wheel is driven by rate and position error:

$$T_y = I_y (\omega_n^2 e + 2\xi\omega_n \dot{e})$$

where I_y is inertia (N-m-sec^2), ω_n is the control frequency and is set to 0.01 (rad/sec), ξ is a damping constant and is set to 0.707, e is the pitch error angle (rad), and $\dot{e}$ is the pitch error rate (rad/sec). Both e and $\dot{e}$ are products from the attitude determination algorithm.

Magnetic Control for Science Pointing

SAMPEX attitude control involves a momentum bias along the spacecraft y-axis and the pointing of this axis at the sun. Let H_0 be a desired level of angular momentum, with **j** a y-axis unit vector and **s** the sun unit vector in the body frame. Ideally, we would like to have the system angular momentum vector (H) equal to both H_0 **j** and H_0 **s**. This will be true only if the y-axis is pointed at the sun, momentum is at the desired level and there is no nutation.

We now consider two momentum error vectors, $\mathbf{H} - H_0\,\mathbf{j}$ and $\mathbf{H} - H_0\,\mathbf{s}$ and add them together to form an" excess" momentum vector to be "unloaded" by the magnetic torquing system. Let $\Delta\mathbf{H} = 2\mathbf{H} - H_0(\mathbf{j} + \mathbf{s})$. One common method of momentum unloading is to let $\mathbf{dm} = \Delta\mathbf{H} \times \mathbf{b}$ where **dm** is an applied magnetic moment, **ΔH** is the undesired system momentum and **b** is the magnetic field vector in the body frame. In our case, we let

$$\mathbf{dm} = \text{Mag\_gain}\,(\Delta\mathbf{H} \times \mathbf{b})$$

where Mag_gain - an appropriate control gain (A-m^2/N-m-sec-tesla)
ΔH - undesired system momentum (N-m-sec)
b - Measured earth's magnetic field (tesla)

In more detail, this becomes

$$\mathbf{dm} = \text{Mag\_gain} \begin{vmatrix} \mathbf{i} & \mathbf{j} & \mathbf{k} \\ 2H_x - H_o\, s_x & 2H_y - H_o\, (1+s_y) & 2H_z - H_o\, s_z \\ b_x & b_y & b_z \end{vmatrix}$$

Sun Point Only Mode (Software Safe Hold)

The sun pointing only mode is a non-science control mode. It is executed at computer power on, or when the spacecraft pitch axis drifts away from the sun line by more than 15 degrees.

The goal of this control mode is to remove any excess spacecraft body spin rates, and to precess the spacecraft pitch axis to within 15 degrees of the sun line. In this mode, there is no pitch axis control and the momentum wheel speed is held constant. The torque signal in N-m is given by:

$$T_y = G_{mw}(\text{fixed\_rate} - v)$$

where v is the momentum wheel speed (rad/sec), and G_{mw} is momentum wheel gain in N-m-sec/rad.

The roll and yaw torquer bars are used for spin rate control (spin control) and momentum unloading (Bdot control). In addition to these two goals, the pitch axis torquer bar is used to precess the pitch axis toward the sun line (y-axis precession control). The following equation describes these three controls:

$$\mathbf{dm} = -G_{Bdot}(\mathbf{b}-\mathbf{b_p})/\Delta t + SC_y G_s \begin{bmatrix} b(3) \\ 0 \\ b(1) \end{bmatrix} + SC_z G_s \begin{bmatrix} b(2) \\ b(1) \\ 0 \end{bmatrix} + \text{SIGN}(bs) G_p F_e F_p \begin{bmatrix} 0 \\ 1 \\ 0 \end{bmatrix}$$

where G_{Bdot} - Bdot control gain (A-m^2-sec/tesla)
G_s - spin control gain (A-m^2/tesla)
G_p - y-axis precession gain (A-m^2)
bs - the second element of $\mathbf{b}\times\mathbf{s}$, $\mathbf{s}$ is the body sun vector and $\mathbf{b}$ is the measured earth's magnetic field vector (tesla)
$\mathbf{b_p}$ - previous measured earth's magnetic field vector (tesla)
SC_y - y-axis spin control flag, values can be -1, 0, 1 corresponding to despin, off, spin
SC_z - z-axis spin control flag, values can be -1, 0, 1 corresponding to despin, off, spin
F_e - eclipse flag, 1 = no eclipse, 0 = eclipse
F_p - y-axis precession flag, 1 = y-axis precession enable

0 = y-axis precession disable

Δt - the difference between the current time and the time when the previous sample is taken (sec)

The first term of the above equation describes the Bdot control for the momentum unloading. The dipole moment for each torquer bar is proportional to the rate of change of the measured earth's magnetic field along each axis. The second and third terms describe the spin control with the second term for y-axis spin and the third term for z-axis spin. These two spin controls will not be on simultaneously. The control options spin, despin, or off are selected by ground command. The last term describes y-axis precession control. The dipole moment for this control is driven by a gain with a polarity determined by the cross product of the measured sun and magnetic field vectors. The precession dipole moment is not computed when the spacecraft is in eclipse and can be disabled by ground command. All gains are modifiable via ground command.

V. SIMULATION

The dynamic simulator has two parts: 1) system state model (SSM) and 2) on-board attitude determination and control. The on-board attitude control and determination has been discussed in sections II, III, and IV. The SSM includes the kinematics (including the quaternion and direction cosine matrix), the equations of motion, and the sensor models. Figure 5 below describes the dynamic simulator.

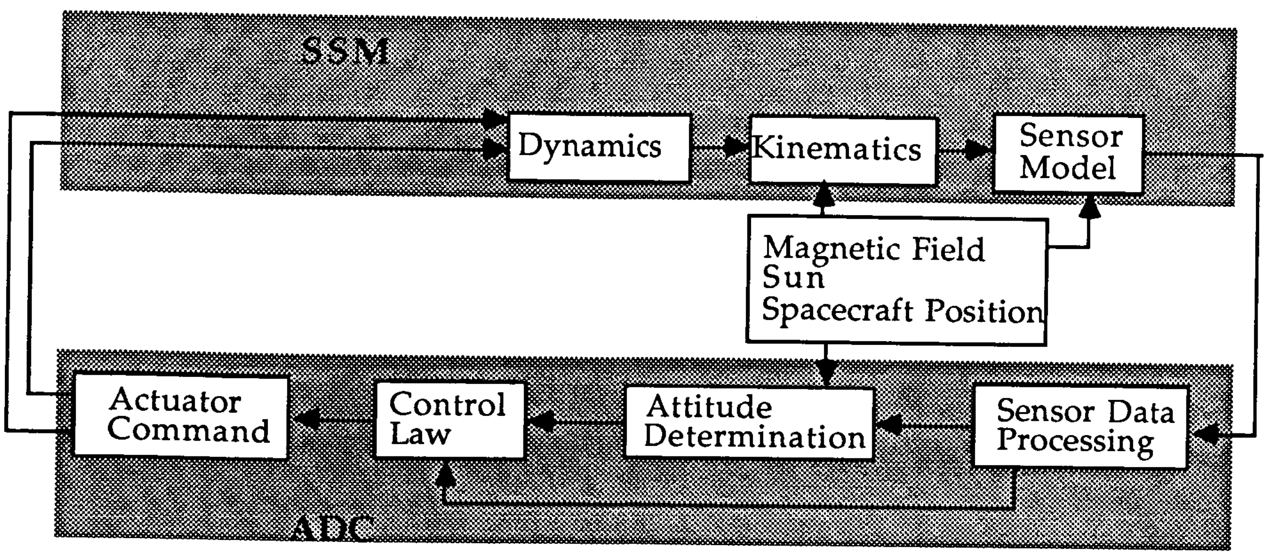

Figure 5. The Dynamic Simulator

Runge-Kutta integration on the derivatives of the system momentum and wheel momentum, the spacecraft position and velocity vectors, and the quaternion.

Momentum

The system momentum is computed by integrating Euler's equation [2]:

$$\dot{\mathbf{H}} = \mathbf{T}_{aero} + \mathbf{T}_{gg} + \mathbf{T}_{mu} - \boldsymbol{\omega} \times \mathbf{H}$$

where:
- H - system angular momentum (N-m-sec)
- T_{aero} - aerodynamic torque (N-m)
- T_{gg} - gravity gradient torque (N-m)
- T_{mu} - magnetic unloading torque (N-m)
- ω - spacecraft rate (rad/sec)

The Pictorial Solar Pressure (PSP) program is used to compute the aerodynamic torques on SAMPEX. PSP takes into account the effects of shadowing on the spacecraft. The density is selected for a worst case solar cycle.

$$T_{aero} = \mathrm{conv}[(1/2)\rho V^2 C_d T_{norm}]$$

where:
- conv - conversion factor $\left(16387.3 \dfrac{N-m-sec^2}{(gm/cm^3)km^2\, in^3}\right)$
- ρ - atmospheric density (gm/cm^3)
- V - orbit velocity (km/sec)
- C_d - coefficient of drag
- T_{norm} - normalized torque output from PSP (in^3)

The gravity gradient torque is computed by [2]:

$$T_{gg} = 3(\mu / |\mathbf{R}|^3)[\mathbf{r}_b \times ([I] \cdot \mathbf{r}_b)]$$

where:
- μ - earth's gravitational constant ($3.98601 \times 10^5\ km^3/sec^2$)
- $\mathbf{r}_b$ - zenith vector in body frame
- $|\mathbf{R}|$ - magnitude of radius vector (km)
- I - inertia tensor (N-m-sec^2)

The magnetic unloading torque is determined from:

$$T_{mu} = k_1 M_{lim} \times \mathbf{B}$$

where $\mathbf{B}$ is the earth's magnetic field strength in tesla computed using a tenth order spherical harmonic magnetic field model [2] and $\mathbf{M}_{lim}$ is the limited magnetic dipole moment with a torquer bar limit of 20 A-m^2. The magnetic dipole moment is determined from the sun pointing and nutation damping error vectors discussed earlier in Section IV. The constant k_1 is a conversion factor (1.0 N-m/A-m^2-tesla).

The $\boldsymbol{\omega} \times \mathbf{H}$ term is the gyroscopic reaction torque. It is the cross product of the body rates and the system angular momentum vector.

The system angular momentum is determined from the following integral

$$\mathbf{H} = \int \dot{\mathbf{H}}\, dt = \int (\mathbf{T}_{ext} - \boldsymbol{\omega} \times \mathbf{H}) dt$$

where $\mathbf{T}_{ext}$ is the sum of the external torques on the spacecraft.

Similarly, the reaction wheel angular momentum is the integral of the applied wheel torque

$$h_w = \int (T_w + T_{friction}) dt$$

where T_w is the commanded wheel torque and $T_{friction}$ is the torque due to friction.

Quaternion

The quaternion obeys the kinematic equation of motion [2]:

$$\dot{q} = (1/2)\Omega q$$

where $$\Omega = \begin{bmatrix} 0 & \omega_z & -\omega_y & \omega_x \\ -\omega_z & 0 & \omega_x & \omega_y \\ \omega_y & -\omega_x & 0 & \omega_z \\ -\omega_x & -\omega_y & -\omega_z & 0 \end{bmatrix}$$ is the skew-symmetric matrix

$$\omega = [I]^{-1}(H - h_w j)$$

h_w - wheel momentum (N-m-sec)

j - unit vector along the y-axis.

Using this quaternion, a direction cosine matrix can be generated:

$$\text{AMATRIX} = \begin{bmatrix} q_1^2 - q_2^2 - q_3^2 + q_4^2 & 2(q_1q_2 + q_3q_4) & 2(q_1q_3 - q_2q_4) \\ 2(q_1q_2 - q_3q_4) & -q_1^2 + q_2^2 - q_3^2 + q_4^2 & 2(q_2q_3 + q_1q_4) \\ 2(q_1q_3 + q_2q_4) & 2(q_2q_3 - q_1q_4) & -q_1^2 - q_2^2 + q_3^2 + q_4^2 \end{bmatrix}$$

This matrix and its transpose are used to transform vectors from the inertial frame to the body frame and vice versa. This process is used in the sensor models in the simulation.

Sensors

The spacecraft ephemeris is generated by integrating the equations of motion including the perturbations due to nonspherical earth gravity effects and aerodynamic forces.

The sun vector in inertial frame is used to generate the sun vector in body frame and for attitude determination. It is computed using 1985 coefficients.

The magnetometer reading includes the earth's magnetic field, computed using a 10th order spherical harmonic magnetic field model [2], and the magnetic field produced by the torquer bars. The contamination due to the torquer bars is approximated by modelling the torquer bar as a dipole. For thin cylindrical bar magnets, the distance between the two poles of the dipole is approximately five-sixths the length of the magnet [6] (the torquer bar in our case). A contamination matrix is formed based on the contamination due to each bar in each of the axes of the three-axis magnetometer.

Results

The dynamic simulator has been used to simulate both the vertical pointing mode and the orbit rate rotation mode for various orbit configurations. These configurations include 6 PM, 9 PM, and midnight orbits, where the time is the local time at the ascending node. The simulator also examines how the relative motion of the sun and the earth throughout the year effects such things as the sun pointing error and pitch loop position error.

Plots of the sun pointing error and the pitch loop position error for 6 PM, 9 PM, and midnight orbits are shown in Figures 6, 7, and 8, respectively, for the vertical pointing mode and in Figures 9, 10, and 11 for the orbit rate rotation mode. These simulation runs are for the winter solstice time of year. The results of the simulation show that the five degree sun pointing requirement is met for all cases investigated for both control modes.

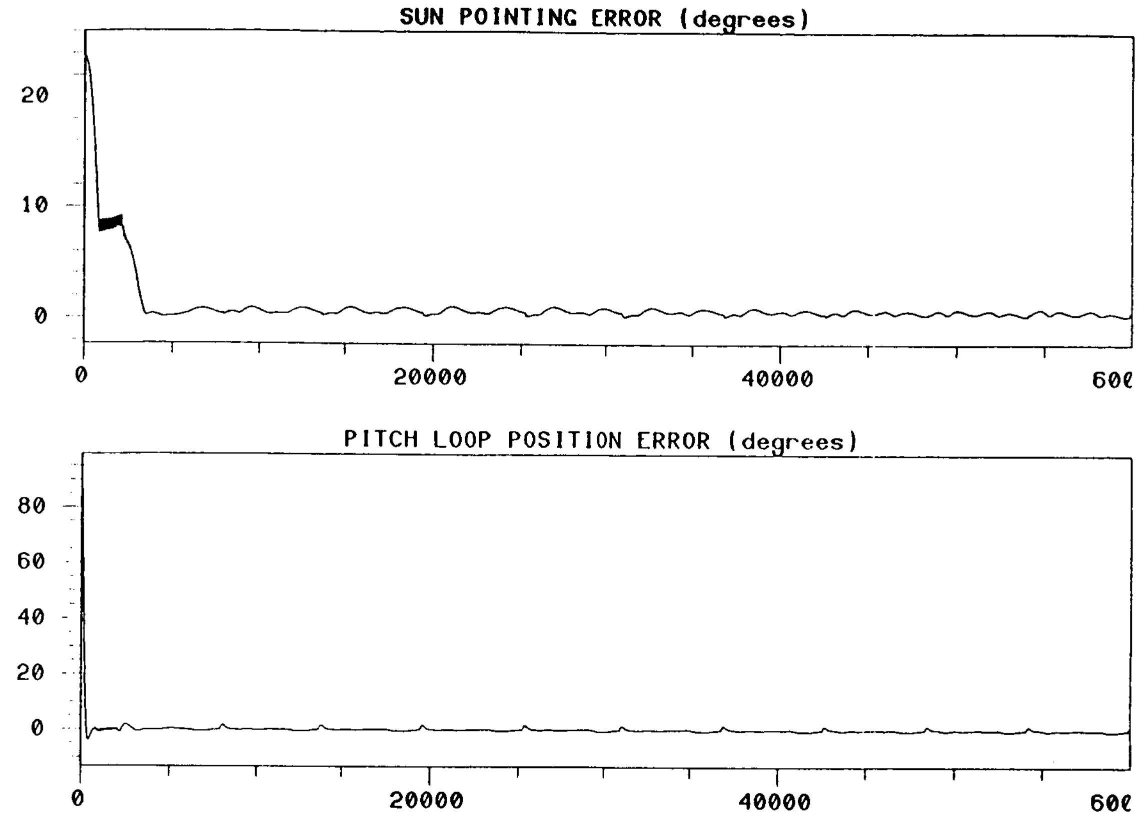

Figure 6. 6 PM Orbit - Vertical Pointing Mode

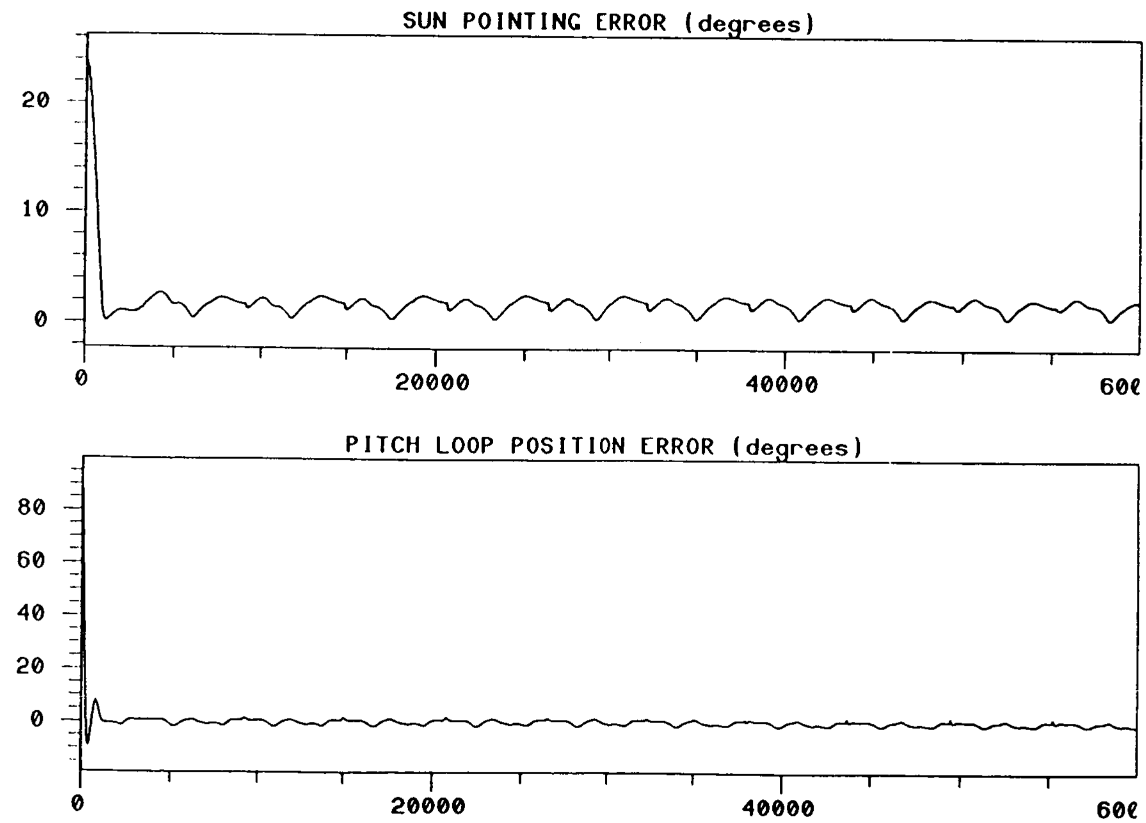

Figure 7. 9 PM Orbit - Vertical Pointing Mode

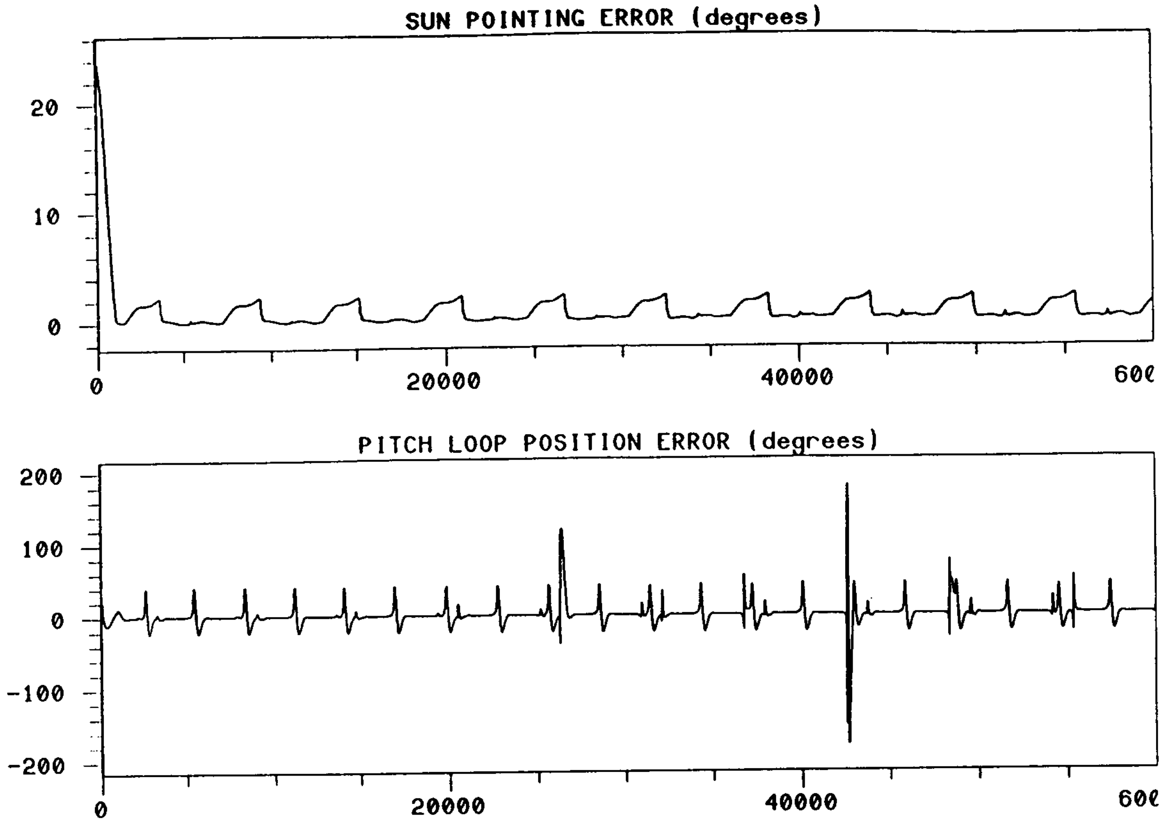

Figure 8. Midnight Orbit - Vertical Pointing Mode

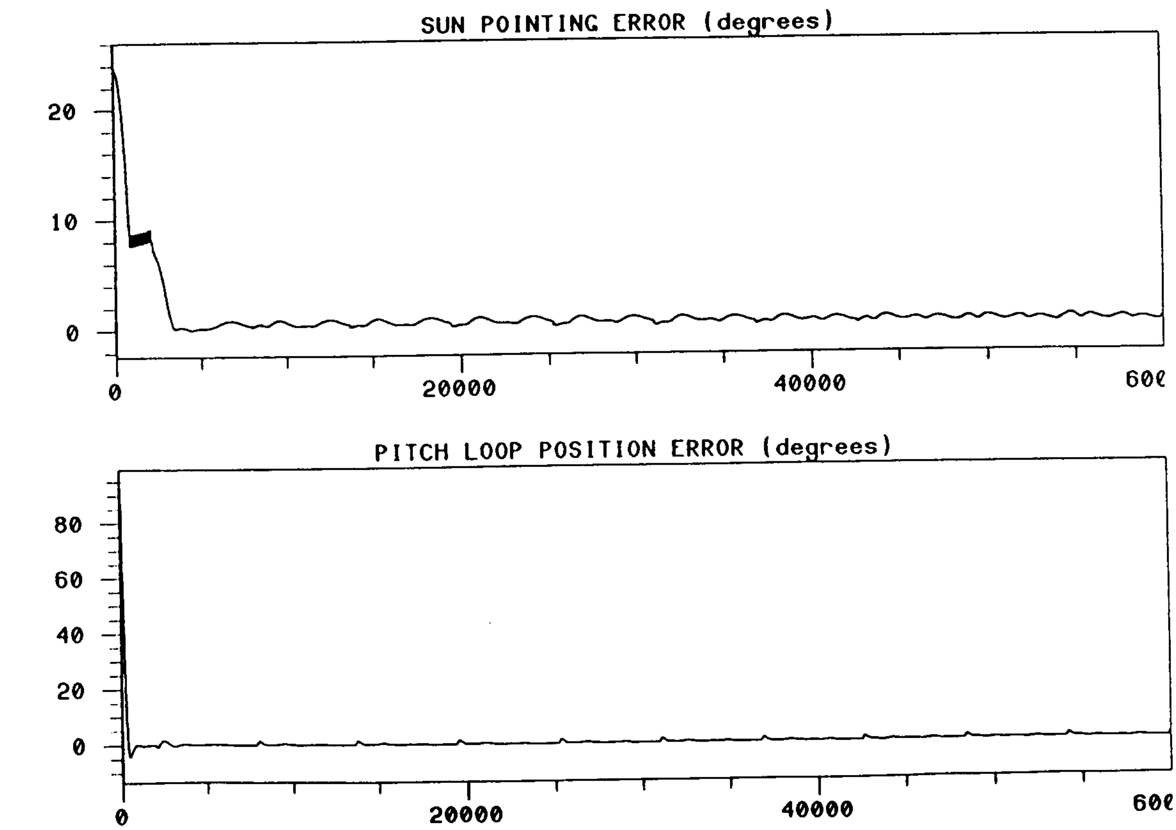

Figure 9. 6 PM Orbit - Orbit Rate Rotation Mode

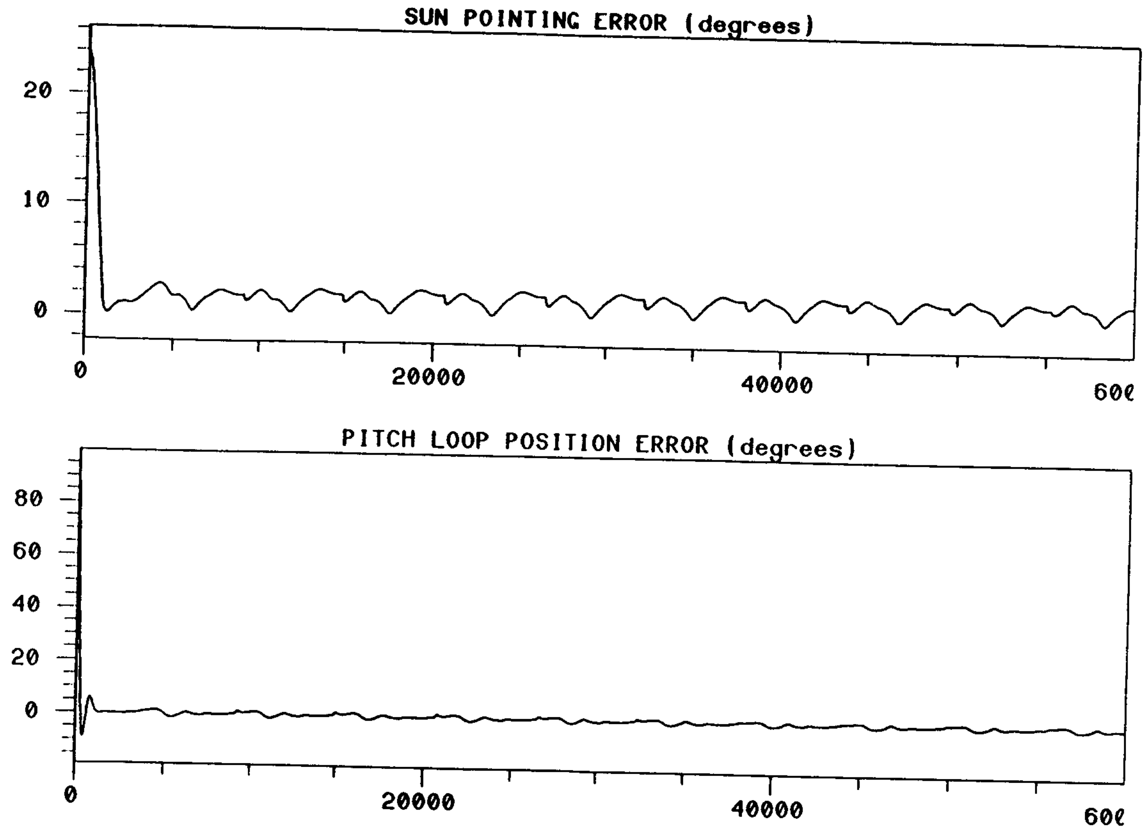

Figure 10. 9 PM Orbit - Orbit Rate Rotation Mode

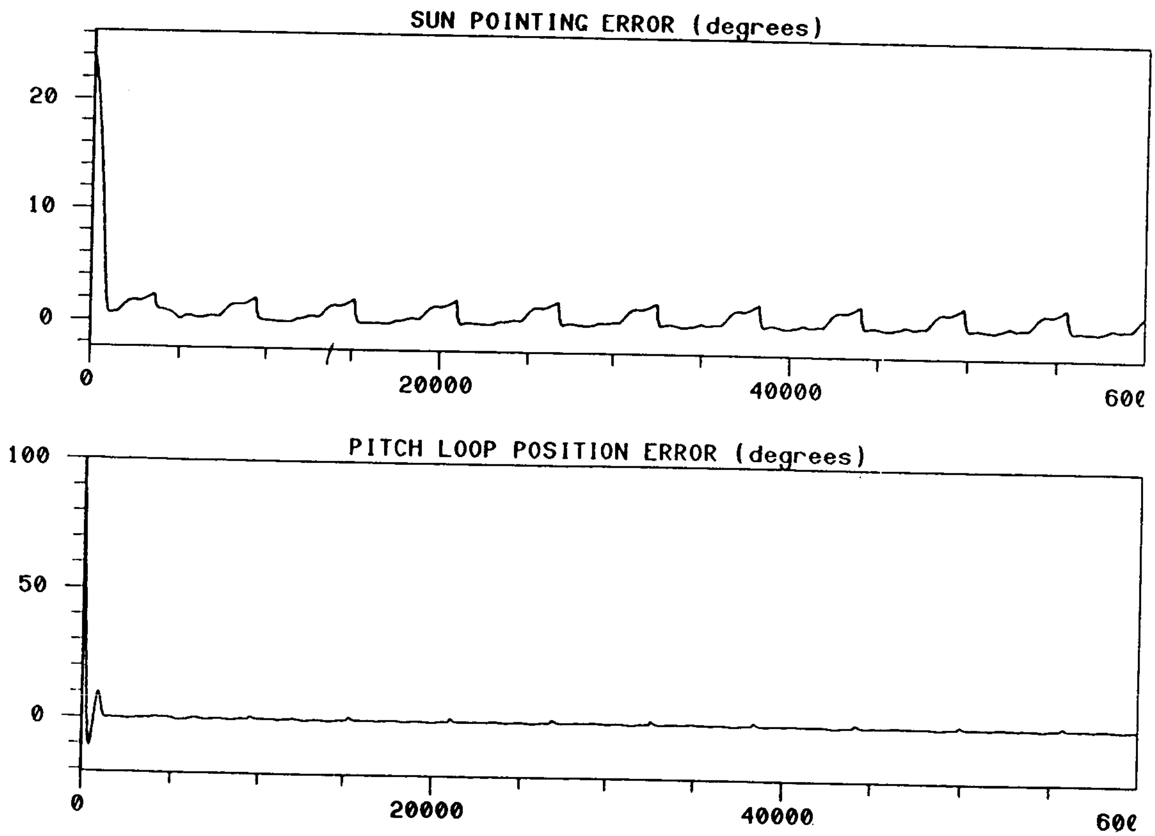

Figure 11. Midnight Orbit - Orbit Rate Rotation Mode

VI. SUMMARY

The simulation results show that the on-board attitude determination and control algorithm satisfies the attitude knowledge and control requirements. Since the orbit rate rotation mode satisfies the spacecraft pointing requirement as well as the velocity vector avoidance criterion, this mode is recommended as the base control mode for normal operation.

ACKNOWLEDGEMENT

We would like to acknowledge the assistance of John Azzolini, Jim Donohue, Henry Hoffman, the Guidance and Control Branch, and the SMEX project at Goddard Space Flight Center for their contributions to the SAMPEX project. Their input aided us in the development of the software algorithm and the design andanalysis of the dynamic simulator.

REFERENCES

[1] *Proposal for a Solar, Anomalous, and Magnetospheric Particle Explorer* (SAMPEX) , submitted to NASA in response to A. O. No. OSSA 2-88 for the Small-Class Explorer Mission, September 1988

[2] Wertz, James R. (editor), *Spacecraft Attitude Determination and Control,* 1986

[3] Report pending under NAS5-39412, Task 18

[4] GSFC, Flight Dynamics Division Document No. FDD/552-89/001

[5] Shuster, M. D. and Oh, S. D., *Three-axis Attitude Determination from Vector Observations,* Journal of Guidance and Control, 1981 Jan-Feb, P70-77

[6] Lemon, Harvey Brace and Ference, Michael, Jr, *Analytical Experimental Physics*, P244