Onboard Spin Axis Controller for a Geostationary Spin-Stabilized Satellite

Carl Leei Chen,* Loren I. Slafer,† and W. F. Hummel Jr.‡
Hughes Aircraft Company, El Segundo, California

Design and performance of an onboard system for precision control of the spin axis attitude of either spin- or dual-spin-stabilized geostationary satellites is presented in this paper. The spin axis controller uses the output of a spinning Earth horizon sensor to derive onboard real-time estimates of the spin axis roll and yaw attitude errors and Earth sensor bias. The Earth sensor directly provides the spacecraft roll angle measurement, while the Earth orbit kinematic coupling is used to derive the spacecraft yaw angle of sensor bias information. Maneuver logic uses these attitude error data to implement a simple control scheme that spin synchronously pulse fires a single axial thruster, providing for complete roll and yaw control of the spacecraft attitude. The proposed system provides substantially increased spacecraft autonomy and significantly reduces operational loads and the costs of the ground control station. The system also provides high spin axis pointing accuracy in the presence of severe attitude precession rates. The spin axis controller system is currently being implemented on the Intelsat VI communication satellite.

I. Introduction

FOR all spin- or dual-spin-stabilized spacecraft, basic attitude stabilization is achieved through gyroscopic stiffness developed by the vehicle spin, with periodic attitude adjustments to correct for errors induced by external environmental disturbances.[1] For dual-spin spacecraft like Intelsat VI (Fig. 1), the fundamental spacecraft design is based on a spinning rotor, which contains most of the utility subsystems (i.e., power, propulsion, attitude determination and control, and a portion of the telemetry and command subsystem), and a large, mechanically despun, Earth-oriented platform containing the communications payload and the remaining portion of the telemetry and command subsystem. This spacecraft configuration has been used successfully in many military, scientific, and commercial applications, including Tacsatcom I; DSCS II; Intelsats III, IV, IVA, VI; Leasat; the Hughes HS376 series; Galileo; the Orbiting Solar Observatory; and geostationary meteorological satellites.

A primary characteristic of this class of spacecraft is the extremely high system angular momentum that minimizes the vehicle attitude precession rate in the presence of environmental disturbances such as gravity gradient, magnetic, and solar pressure-generated torques. Angular momentum for the Intelsat IV and IVA and HS376 series of geosynchronous communications satellites is approximately 1200 ft-lb-s, while newer vehicles such as Leasat and Intelsat VI will have momentum levels of 6000–7000 ft-lb-s on-orbit with up to 15,000 ft-lb-s during geosynchronous orbit transfer operations.

For these types of spacecraft, attitude control functions can be divided into three basic categories: spin axis attitude determination and control, payload despin (rate and pointing) control, and vehicle nutation stabilization. Orientation of the spacecraft spin axis attitude is achieved with phased pulse firings of a spinning axial thruster, based upon inertial attitude data derived from the outputs of spinning Earth and sun sensors. For the Intelsat VI type of spacecraft, primary rate and pointing control of the platform about the spacecraft spin axis is achieved by the rotor-based servo illustrated in Fig. 2. Rotor-mounted inertial attitude sensors, together with a platform-to-rotor relative angle sensor, are used to estimate and control the inertial spin rate and pointing direction of the platform.

For geosynchronous satellites, the dominant source of disturbance to the spin axis attitude is solar pressure. For well-balanced spacecraft (small center of mass to center of pressure separation) such as Leasat or the Hughes HS376 series, the high level of angular momentum results in attitude drift rates of less than 0.01 deg/day. For more complex spacecraft such as Comstar and Intelsat IVA, drift rates on the order of 0.03 deg/day are common. For a spacecraft such as Intelsat VI, drift rates between 0.1 and 0.2 deg/day are anticipated over the 14 year satellite lifetime.

Thus, for earlier spacecraft, which had more symmetric configurations (i.e., low drift rates) and less severe payload pointing requirements, attitude determination was carried out by direct ground control using long-term (daily) monitoring of telemetered attitude sensor data. Typically, a least-square estimation algorithm is used to compute the attitude with periodic (every few days) real-time ground-commanded thruster pulse firings to correct the attitude error.

However, for a spacecraft of the Intelsat VI class, which requires precise antenna beam pointing (i.e., the spin axis attitude must be controlled to within 0.04 deg of orbit normal) and which will experience high-attitude precession rates, attitude corrections two to five times per day would place an excessive burden on both workload and cost on the ground station.

This paper describes the onboard spin axis controller (SAC) developed for the Intelsat VI spacecraft currently under development by Hughes Aircraft Company for the International Telecommunications Satellite organization (INTELSAT). The controller design uses the existing attitude sensors and microprocessor-based attitude determination and control subsystem (ADCS) to implement an onboard spin axis controller that significantly increases the spacecraft autonomy and dramatically reduces operational loads on the ground station. Section II of the paper provides an overview of the design and mechanization of the SAC. Section III defines the orbit dynamics and sensor kinematics used in the design. Design and performance of the attitude estimator

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*Senior Staff Scientist. AIAA Member.
†Senior Scientist. AIAA Member.
‡Consultant.

algorithm are presented in Sec. IV and the attitude control logic in Sec. V. SAC performance simulations are given in Sec. VI.

II. Spin Axis Controller Design Overview

Figure 3 presents a functional block diagram of the spin axis controller. The SAC uses the real-time output of a spinning Earth horizon sensor to derive onboard estimates of spin axis roll ϕ and yaw ψ attitude angles as well as the Earth sensor chord bias b. The Earth sensor (described in Sec. III) directly provides a measurement of the spacecraft roll angle, while orbital kinematics are used to estimate spacecraft yaw attitude and sensor bias. Earth sensor bias variations will result primarily from long-term electronic and thermal effects on the sensor as well as effects of nonuniform Earth radiance due to atmospheric variability. Static biases result from residual sensor misalignment and electronic processing bases. Estimating sensor bias eliminates static pointing errors that would otherwise develop.

A simplified model of the spacecraft dynamics is shown where the dominant forcing function of the system is the angular drift rate about the sunline V due to solar torque. For this model, vehicle nutation dynamics have been neglected. This assumption is valid because of the high system angular momentum for this class of spacecraft. Vehicle inertial nutation frequencies are typically in the range of 0.10–1.0 Hz. Nutation damping time constants due to control system action will be significantly shorter than the SAC estimator time constant discussed in the following sections. The nutation control time constant for Intelsat VI ranges between 40 and 90 s. Thus, the vehicle nutation dynamics can be considered uncoupled from the SAC estimation dynamics and have been neglected in the following analysis. The state estimate propagation algorithm is executed every 10 min (corresponding to 2.5 deg of orbital motion). This calculation uses the previous estimates along with a specified average drift rate to predict the present value of the states. The Earth chord processor nominally uses 256 consecutive Earth chord measurements to determine a heavily averaged (to reduce sensor noise) chord width Z_{avg}, which is then used in conjunction with the state predictions to determine a prediction error. This prediction error is multiplied by the estimator gains K_ϕ, K_ψ, and K_b and added to the respective predictions to determine the new state estimates. A coordinate transformation of these state estimates from the orbit-fixed frame (ϕ, ψ) to a sun-fixed frame (U, V) is then performed before the maneuver logic is evaluated. If attitude corrections are necessary, the magnitudes of the desired corrections in the sun-fixed frame (U_c, V_c) are sent to the thruster command processor. Meanwhile, the corresponding corrections to the estimates of roll ϕ and yaw ψ are made and the results fed back to the state estimate propagation algorithm for use in the next cycle.

The automatic attitude maneuver control logic (described in Sec. V) comprises two operational modes: 1) a fixed-time firing (FTF) mode for the normal steady-state operation and 2) an error threshold firing (ETF) mode for the normal transient-state operation. In the FTF mode, attitude corrections are constrained to occur at specific orbit intervals (as marked by a sidereal period clock) relative to the nominal orbit node. The design goal is to have all FTF-generated attitude correction maneuvers take place at orbital locations as close to the orbit node as possible, resulting in the maximum amount of fuel applicable to north-south (inclination) orbit control. On the other hand, the ETF mode employs an error threshold detection logic such that attitude corrections take place whenever either of the sun-referenced coordinate frame estimated attitude components exceeds ground programmable thresholds.

To minimize the spin axis attitude error prior to or following orbit correction maneuvers, the SAC incorporates two automatic sequences for nulling the SAC estimated attitude error: 1) immediate attitude trim (IAT) mode and 2) quick roll estimate and attitude trim (QAT) mode. Either sequence can be executed via onboard stored command or direct ground command. In IAT mode, the SAC uses the latest onboard attitude estimate to execute an immediate attitude trim. In QAT mode, the SAC initiates a quick roll attitude estimate that is then followed by an immediate attitude trim maneuver.

III. Orbit/Spacecraft Dynamics and Sensor Kinematics

Figure 4 shows the two coordinate frames used in the design and analysis of the SAC. The orbit reference frame (X_o, Y_o, Z_o) is oriented as follows: the orbit X_o axis is in the direction of flight, the orbit Z_o axis is normal to the orbital plane, and the orbit Y_o axis is pointed toward the Earth along the radius vector from the satellite to the center of the Earth. The sun fixed reference frame (X_s, Y_s, Z_s) is defined as follows: the Z_s axis coincides with the orbit Z_o axis, the sun vector is in the $Y_s - Z_s$ plane, and the X_s axis completes a right-handed orthogonal triad. Figure 4 also defines the coordinate transformation between these two coordinate systems where the roll ϕ and yaw ψ angles are defined as the rotation angles about the X_o and Y_o axes, respectively, U is the rotation angle about the X_s axis, and V is that about the Y_s axis. The two coordinate systems coincide at local midnight. The orbit coordinate frame rotates about the sun frame, Z_s axis, with an orbit rate of ω_o.

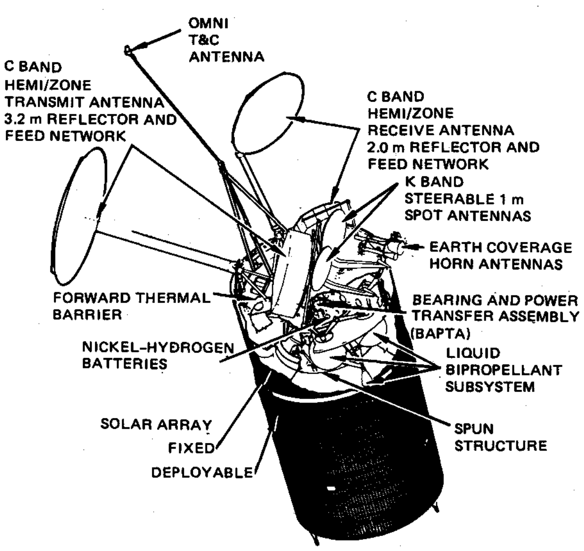

Fig. 1 Intelsat VI spacecraft—on-orbit configuration.

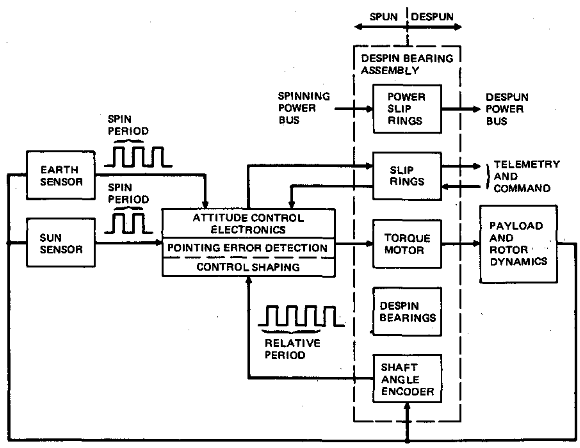

Fig. 2 Attitude determination and control system for dual-spin spacecraft.

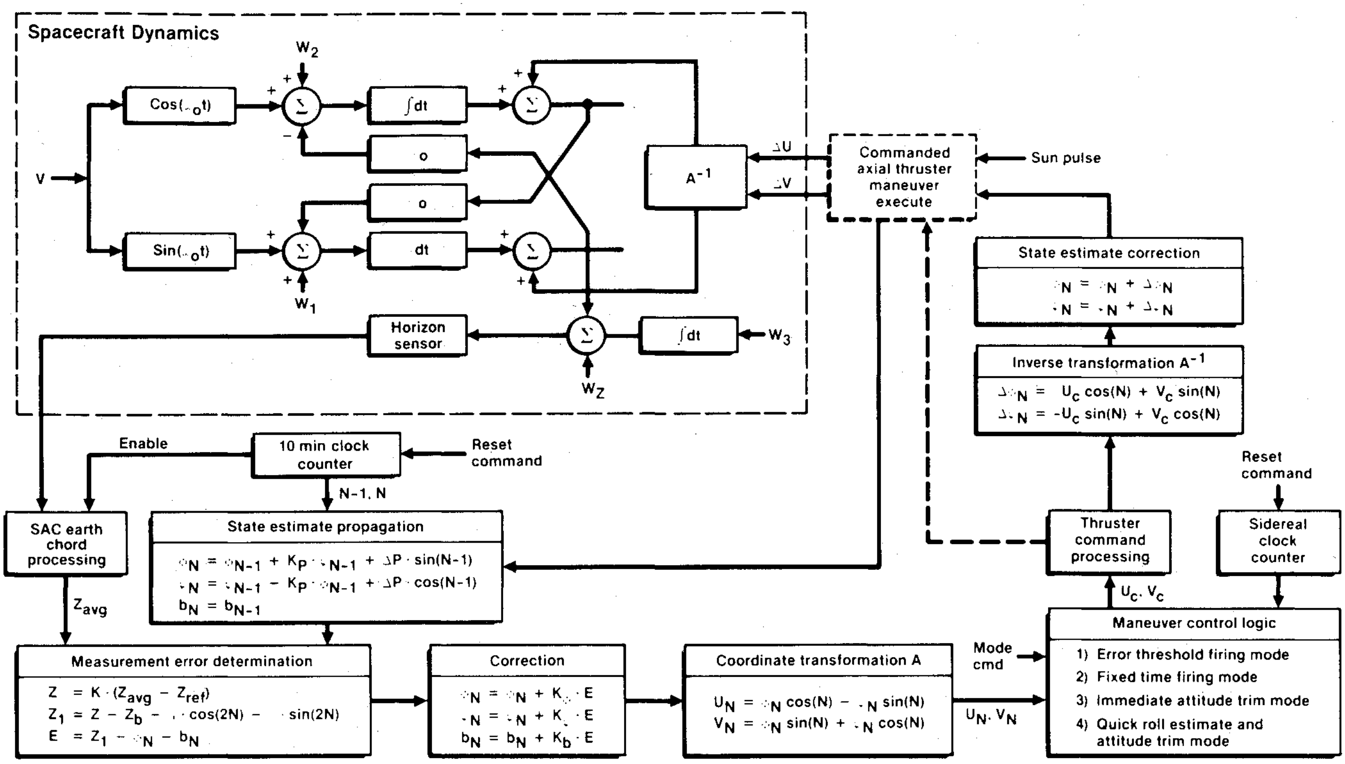

Fig. 3 Spin axis controller functional block diagram.

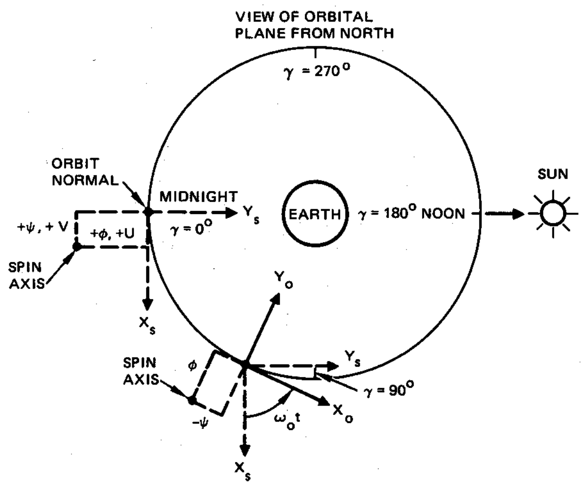

Fig. 4 Spin axis controller coordinate frames.

In geosynchronous orbit, external disturbances to the spacecraft attitude are dominated by the solar wind. The predicted solar torque for Intelsat VI is given in Fig. 5. Using standard modeling techniques, the detailed solar wind generated disturbing torques, which act to precess the spacecraft spin axis, can be predicted and expressed in terms of a Fourier series. Using the dominant secular term in the solar torque Fourier expansion T_o and the spacecraft angular momentum H_o, the spacecraft precession rate due to solar torque can be modeled as

$$\dot{U}=0 \tag{1}$$

$$\dot{V}=A_o \tag{2}$$

where $A_o = T_o/H_o$ (rad/s).

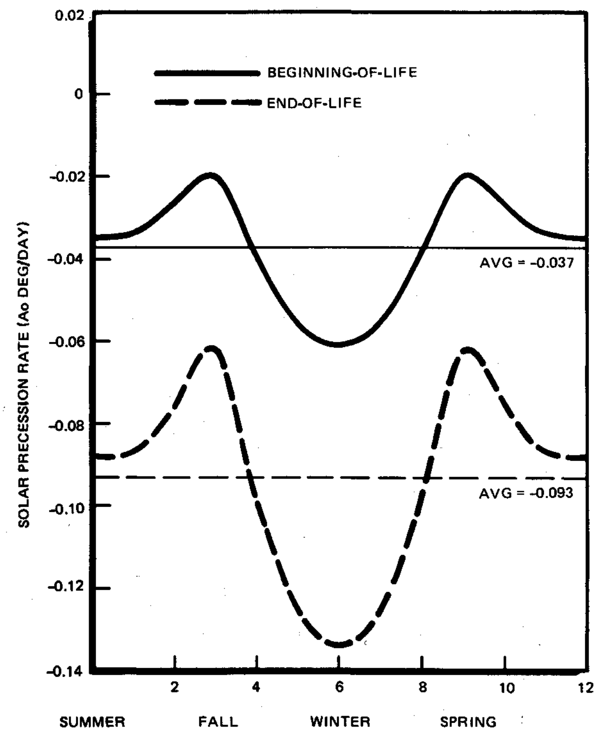

Fig. 5 Intelsat VI solar pressure profile.

The linearized kinematic rotation equation of the spacecraft in roll ϕ and yaw ψ in synchronous orbit are

$$\dot{\phi}=\omega_o\psi+A_o \sin\omega_o t+W_1 \tag{3}$$

$$\dot{\psi}=-\omega_o\phi+A_o \cos\omega_o t+W_2 \tag{4}$$

where W_1 and W_2 are white Gaussian noises intended to model the uncertainty (due to spacecraft modeling errors) in the higher-order terms of the solar torque Fourier expansion and ω_o represents the orbit rate.

The Earth sensor bias variation has been modeled as a random walk with input noise sufficient to cause its 1σ value to grow at a rate determined from on-orbit measurements. Although the mean square value of a random walk process grows linearly with time and seems to be suitable only for short-term operation, it can be justified because the control loop time constant (in the order of 1 h) is much shorter than the characteristic time of the random walk process.

The spinning Earth horizon sensor measures the Earth chord length Z which can be related to the spacecraft roll angle ϕ by the following approximation[2]:

$$Z \cong \left(\frac{2\sin\delta}{\sqrt{\cos^2\delta-\cos^2\rho}}\right)\phi+b=\frac{1}{K}\phi+b \tag{5}$$

where b represents the chord bias.

The sensor geometry and definition of the parameters in the above equation are shown in Figs. 6 and 7. The roll attitude measurement Z_N can be obtained at iteration period N from the following equation:

$$Z_N=K(Z_{\text{avg}}-Z_{\text{ref}})+Z_b-\alpha\cos(2\gamma_N)-\beta\sin(2\gamma_N)+W_z \tag{6}$$

where

Z_{avg} = average Earth chord width
Z_{ref} = referenced Earth chord width, a fixed number based on nominal sensor alignment
Z_b = commandable bias for transient free sensor switching
a,β = commandable parameters to compensate for potential sensor cyclic bias
γ_N = orbital angle at iteration period N
W_z = Earth sensor high-frequency noise modeled as Gaussian white noise with zero mean and with standard deviation of σ_z (typically 0.02 deg)

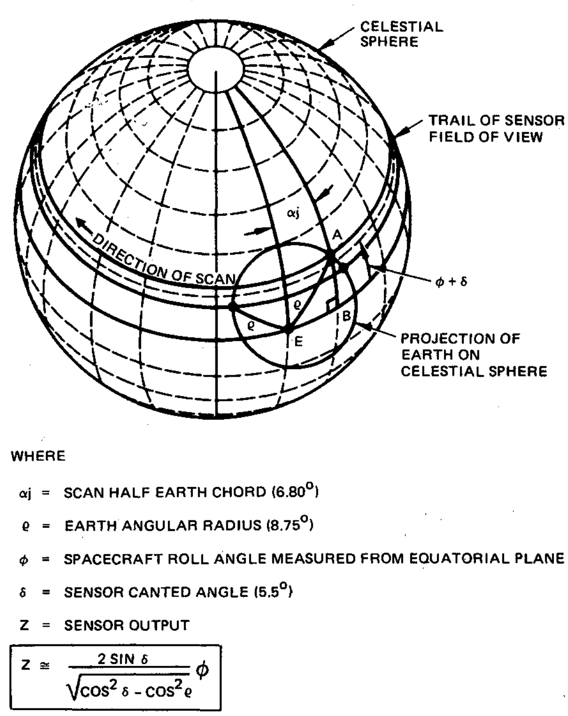

Fig. 6 Earth horizon geometry for geosynchronous satellite.

The random sensor noise will be averaged out after M measurements taken in accordance with

$$w_z=(\sigma_z/\sqrt{M})*N(0,1) \tag{7}$$

No direct yaw angle measurement is available with the spinning Earth sensors; the yaw information can be obtained through the orbit kinematic coupling as described in Eqs. (3) and (4) and Fig. 7.

IV. Attitude Estimator Design and Performance

A three-state estimator, using Earth sensor roll measurements for input data, generates the attitude state estimates that drive the thruster control logic. The three estimated states are roll angle ϕ, yaw angle ψ, and sensor bias b. Other variables (e.g., the parameter A_o in the precession rate model), which vary slowly compared to ϕ, ψ, or b, are treated as constants and may be updated periodically via ground command. The state estimator has the following simplified two step structure:

1) Propagation

$$\bar{\phi}_N=\phi_{N-1}+(\omega_o t)\psi_{N-1}+\hat{A}_o t\sin\omega_o(N-1)t \tag{8}$$

$$\bar{\psi}_N=\psi_{N-1}-(\omega_o t)\phi_{N-1}+\hat{A}_o t\cos\omega_o(N-1)t \tag{9}$$

$$\bar{b}_N=b_{N-1} \tag{10}$$

2) Correction

$$\phi_N=\bar{\phi}_N+K_\phi(Z_N-\bar{\phi}_N-\bar{b}_N)+\Delta\phi_N \tag{11}$$

$$\psi_N=\bar{\psi}_N+K_\psi(Z_N-\bar{\phi}_N-\bar{b}_N)+\Delta\psi_N \tag{12}$$

$$b_N=\bar{b}_N+K_b(Z_N-\bar{\phi}_N-\bar{b}_N) \tag{13}$$

where

$\hat{A}_o$ = ground-estimated average precession rate (commandable constant)
t = estimator update time (10 min)
$\bar{\phi}_N, \bar{\psi}_N, \bar{b}_N$ = predicted values of ϕ, ψ, and b at sample N
$\hat{\phi}_N, \hat{\psi}_N, \hat{b}_N$ = updated values of ϕ, ψ, and b at sample N
K_ϕ, K_ψ, K_b = estimator gains in ϕ, ψ, and b, respectively
$Z_N-\bar{\phi}_N-\bar{b}_N$ = measurement error at sample N
$\Delta\phi_N, \Delta\psi_N$ = control inputs in delta angles resulting from axial thruster firing

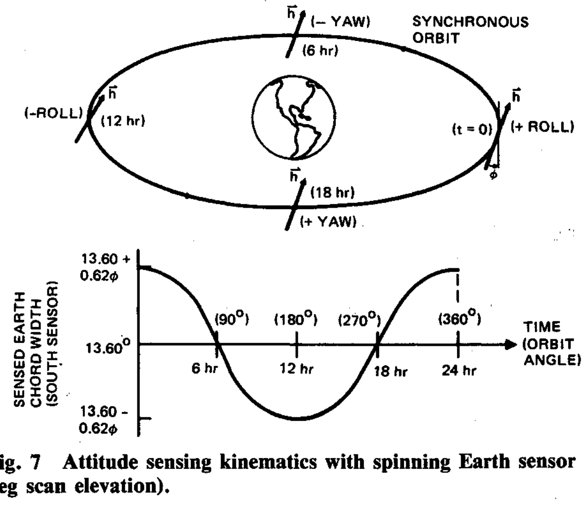

Fig. 7 Attitude sensing kinematics with spinning Earth sensor (5.5 deg scan elevation).

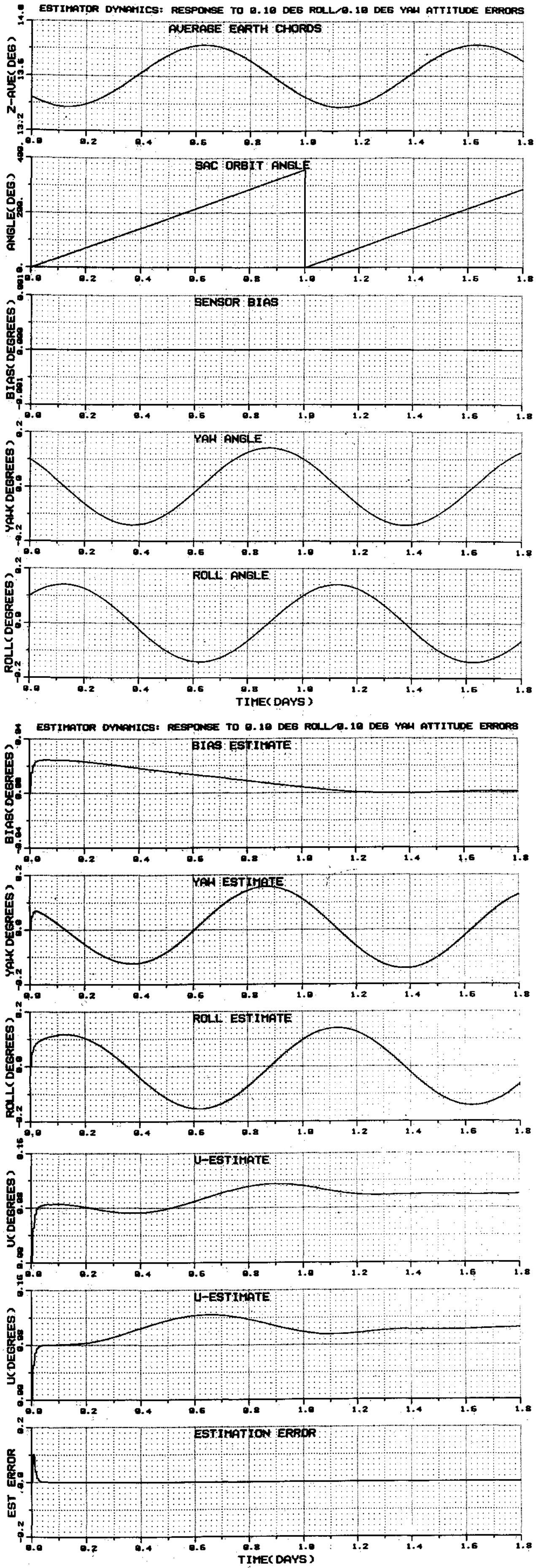

Fig. 8 SAC estimator capture dynamics (open-loop response to 0.10 deg roll and yaw attitude errors).

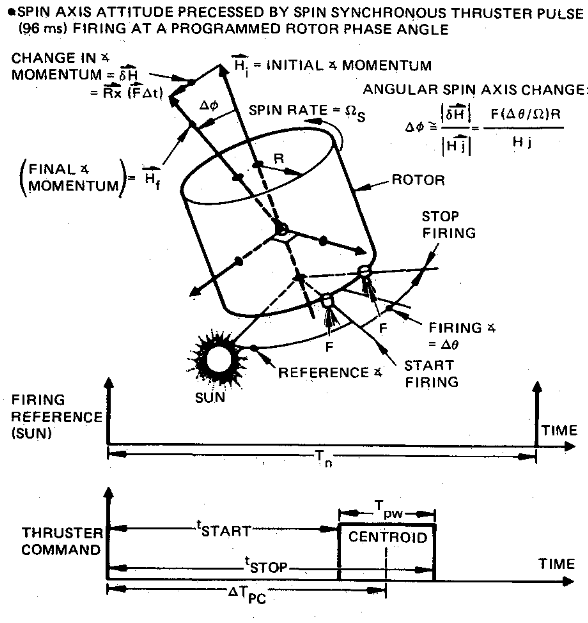

Fig. 9 Attitude control for spin-stabilized spacecraft.

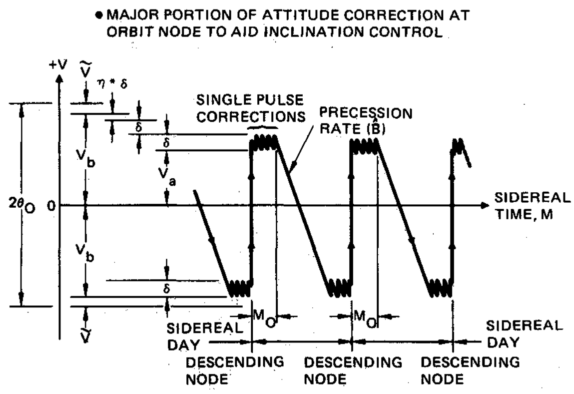

Fig. 10 Fixed-time firing mode correction profile.

Because the maneuver durations are considerably shorter than the SAC update interval (typically less than 30 s), control torques are modeled as impulsive inputs to the estimator. This set of simplified estimator equations is applicable when the estimator update time t is much shorter than the orbital period of 24 h.

The nominal estimator gains chosen for the Intelsat VI SAC algorithm are

$$K_\phi = 0.50,\ K_\psi = 0.50,\ K_b = 0.148$$

The resulting Z-plane poles of the estimator are

$$Z_1 = 0.365360 \text{ and } Z_{2,3} = 0.98241 \pm j0.01160324$$

Mapping these poles into their equivalent S-plane roots gives one real pole (the roll estimate) at $\omega_o = 0.001678$ rad/s ($\tau = 0.165$ h) and a complex pole pair (yaw and bias estimates) with natural frequency $\omega_n = 3.54298 \times 10^{-5}$ rad/s and a damping ratio of 0.831. This gives a yaw/bias time constant of 9.43 h.

The open-loop, transient response characteristics of the attitude estimator demonstrating the system acquisition dynamics in response to individual static errors in the spacecraft states are illustrated in Fig. 8. The figure shows the estimator time histories in response to step changes (0.10 deg) in vehicle roll and yaw attitude. The fast roll response ($\tau = 0.168$ h) is seen with the short initial transient in estima-

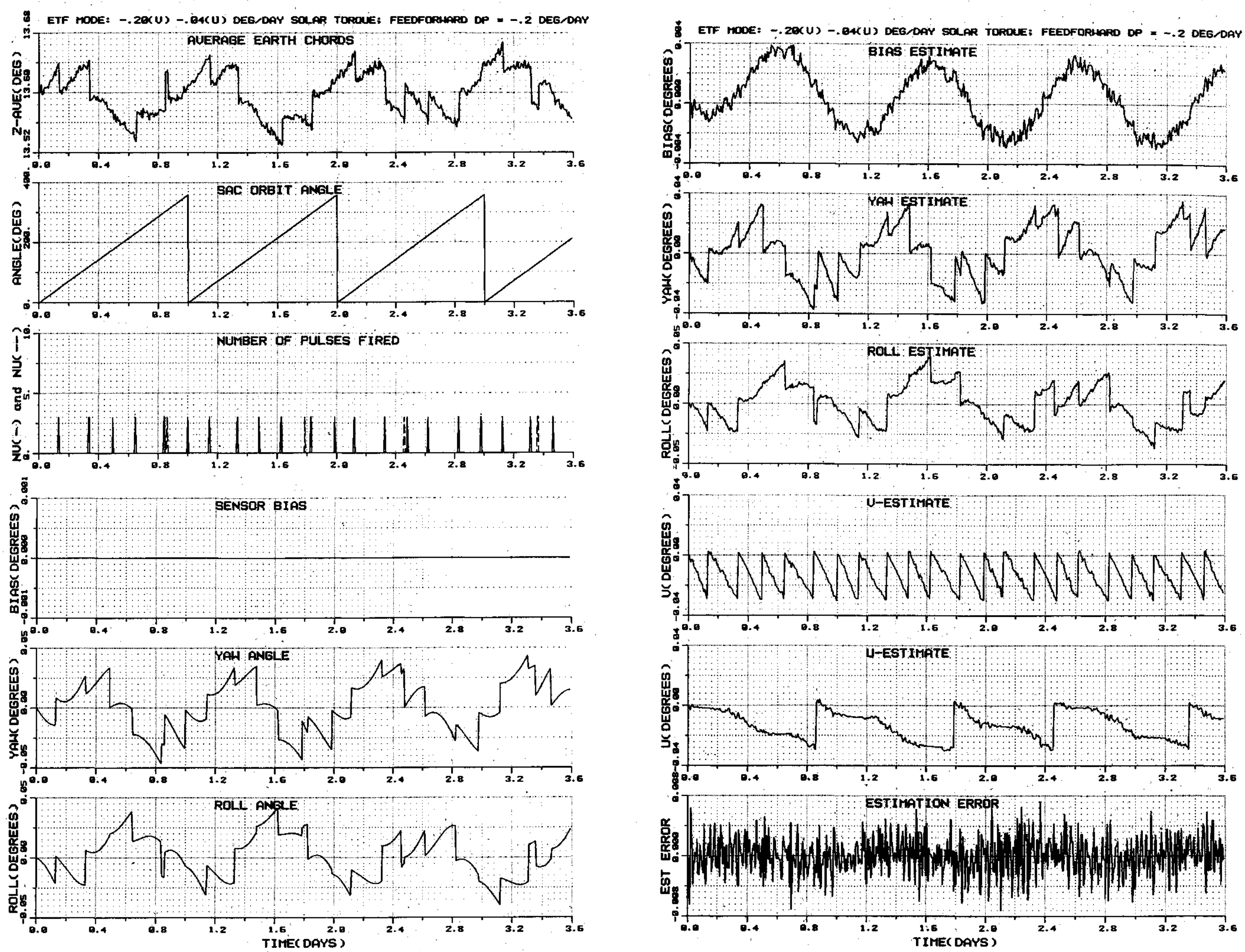

Fig. 11 Spin axis controller closed-loop response to worst-case solar torque (ETF mode).

tion error. The lower-frequency second-order response shown represents the slower yaw/bias estimator dynamics, which rely on the longer-term orbit coupling.

V. Thruster Firing Control Logic

Attitude corrections for a spin-stabilized spacecraft are carried out by spin synchronously firing an axially oriented thruster over a short portion of the spin cycle. This technique is illustrated in Fig. 9. By properly controlling the phase of the thruster pulse centroid, the spin axis can be precessed in any desired direction. For a spacecraft such as Intelsat VI (in which the rotor spins at between 30 and 40 rpm), thruster pulse widths of 0.096 s provide a precession control increment of less than 0.01 deg.

Whenever the estimated attitude (in solar coordinates) exceeds the ground program thresholds, the spin axis controller computes attitude correction maneuver parameters (number of pulses to be fired and thruster pulse phasing with respect to the sun sensor pulse and actual attitude correction). During every iteration cycle (corresponding to 2.5 deg of orbit arc) following correction of the state estimates, the Earth-referenced attitude estimates (ϕ, ψ) are transformed to the sun-fixed reference frame (U and V). In this frame, for a fixed-attitude error, the estimates would be constant. Using this attitude estimate, thruster firing maneuvers are computed and automatically executed.

The SAC algorithm incorporates two primary control modes and two special purpose modes that temporarily override the normal processing logic. For each of the four operational modes, the number of pulses to be fired are computed using a ground commandable thruster effectiveness parameter δ. The pulse counts are derived from the individual quadrature components of the estimated attitude error (U and V). Because these angles are fixed with respect to the sun, the spin phase at which the thruster is fired (relative to the spin phase of the sun) is also constant. The sign of the estimated error (U and/or V), in conjunction with the fixed sun sensor/thruster mechanical alignment data, will thus define the thruster firing phase angle. As each maneuver is executed, the internal state estimates are corrected by the actual (quantized) maneuver magnitude (i.e., the correction angle equals the number of pulses times pulse effectiveness).

The two primary control modes of the SAC are: 1) an error threshold firing mode that generates thruster pulses to null the precession errors once a set threshold has been exceeded and 2) a fixed-time firing mode that operates similarly, but restricts firings to occur near the orbit nodal crossing for more efficient overall fuel usage. To minimize the spin axis attitude errors prior to stationkeeping maneuvers, the SAC incorporates two additional automatic sequences for nulling the estimated attitude error: immediate attitude trim mode and quick roll estimate and attitude trim mode. Either sequence can be executed via onboard stored command or direct ground command. In the IAT mode, the SAC uses the latest onboard attitude estimate to execute an immediate attitude trim bringing the spin axis to the desired orbit normal. In the QAT mode, the SAC initiates a quick roll attitude estimate followed by an immediate attitude trim maneuver to null the more important roll attitude error.

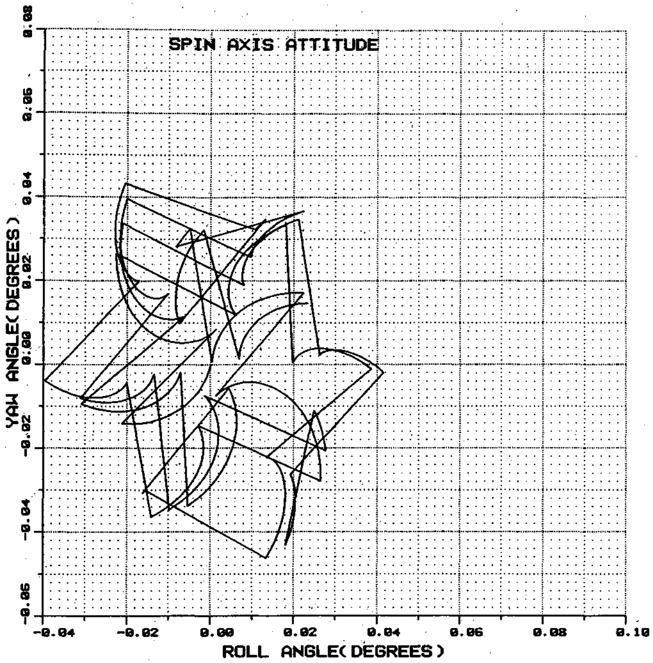

Fig. 12 Spin axis attitude response in ETF mode.

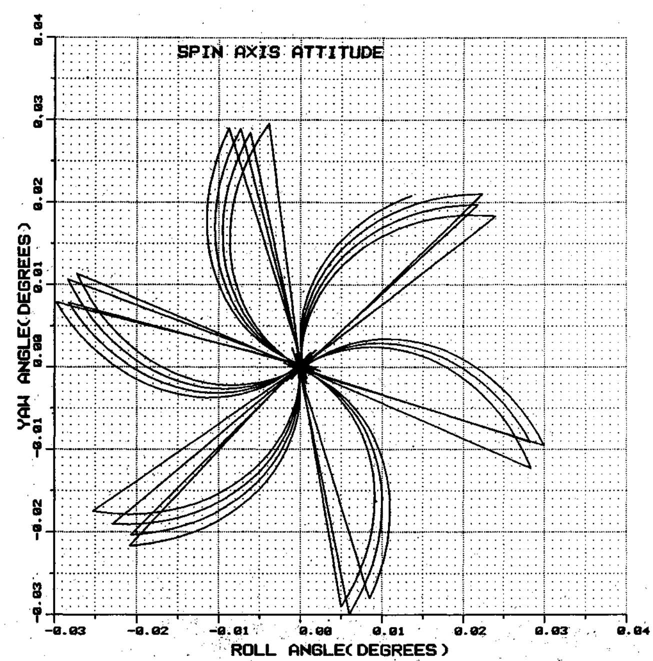

Fig. 13 SAC (ETF) closed-loop performance with perfect feedforward solar precession compensation (0.20 deg/day V axis, 0.03 deg thresholds).

Error Threshold Firing Mode

The ETF mode is the simpler of the two operating modes. Attitude is corrected whenever a ground commandable threshold is exceeded. The threshold logic monitors the two attitude estimates in the sun-fixed frame (U, V). When either of these estimates exceeds its threshold (U_o or V_o), a command to null the attitude in that particular direction is executed. For example, if $V_o = 0.03$ deg and V is calculated to be 0.031 deg, a V correction of 0.031 deg would be desired. The actual commanded correction will be as close to -0.031 deg as the thruster control granularity will allow. This granularity equals the attitude correction resulting from a single thruster pulse and is a ground-commanding parameter that is matched to actual thruster performance.

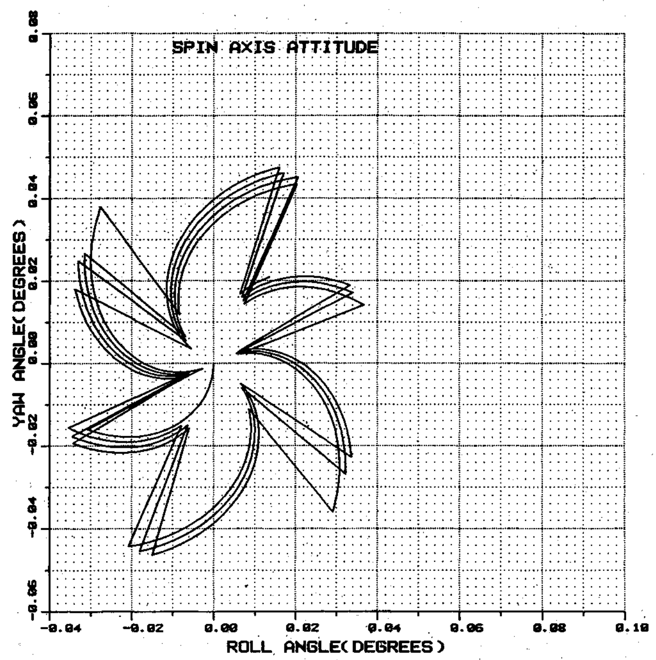

Fig. 14 SAC (ETF) closed-loop performance with 50% low feedforward solar precession (0.20 deg/day, V axis, 0.03 deg thresholds).

Fixed-Time Firing Mode

The FTF mode involves a more complex logic structure that determines corrections according to both error magnitude and spacecraft orbit location. Its objective is to restrict corrections to occur near the orbit node for the most efficient use of fuel. By correcting attitude near the orbit node, the small ΔV that occurs will assist in north-south stationkeeping (controlling orbit inclination). Figure 10 shows an example time history of the estimated spacecraft precession normal to the sunline during FTF mode operation. The controller's attempt to maintain the attitude within a ground-specified band can be seen. Major corrections occur once a day at the node. For large precession rates where one correction per day is not sufficient, additional logic keeps the estimate within the band by a series of minor (one thruster pulse) corrections made near the orbit node.

The portion ϵ of the SAC FTF mode attitude correction fuel consumption effective as a north-south ΔV maneuver is equal to $\cos(\omega_o t_o)$, where ω_o is the orbital rate and t_o the time from the orbit node crossing (i.e., $\omega_o t_o$ is the orbital angle from the node). For an attitude trim occurring at $t_o = 0$ h (orbit node), $\epsilon = 100\%$ and, at $t_o = 6$ h (antinode), $\epsilon = 0\%$. If an attitude trim occurs at t_o greater than 6 h, the fuel saving effectiveness becomes negative. Therefore, an SAC FTF mode attitude correction pulse firing at an orbital angle more than 90 deg away from the orbit node will cause the SAC fuel consumption to contribute to the north-south ΔV maneuver in a negative sense. Thus, it is desirable to design the FTF mode control logic such that t_o is ± 6 h from the orbit node crossing time.

Immediate Attitude Trim and Quick Roll Estimate and Attitude Time

Two ground-commanded control modes IAT and QAT are incorporated to null the spin axis attitude prior to and following orbit stationkeeping maneuvers. In the IAT mode, the SAC processor uses the latest onboard estimates to execute an immediate attitude trim where $U_c = U$ and $V_c = V$. In the QAT mode, the SAC initiates a quick roll attitude estimate using the average of less than the normal number (e.g., ETF or FTF modes) of chord length measurements (64) and the sun frame components of the attitude error estimates are computed with yaw ψ equal to zero. Then, an

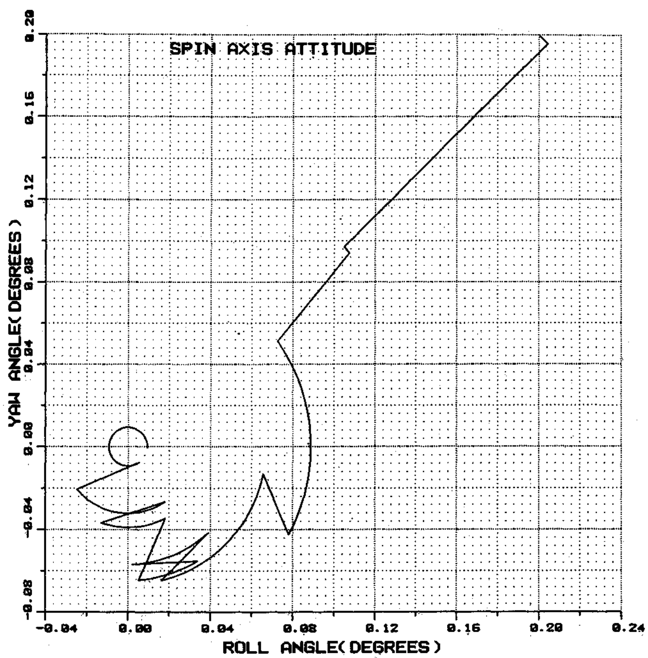

Fig. 15 SAC closed-loop response to initial 0.20 deg roll and yaw attitude errors.

immediate trim maneuver is executed in the same fashion as in IAT mode. To eliminate the transient response, the estimate of bias is not updated until 6 h after a QAT has occurred.

VI. SAC Performance

The nominal closed-loop performance of the SAC operating in ETF mode is illustrated in the simulation time histories given in Figs. 11 and 12. The simulation shows the system response to a worst-case constant V-axis solar torque generated precession of -0.20 deg/day and a -0.04 deg/day U-axis precession rate. Worst-case sensor noise of 0.09 deg, 3σ was assumed. Here, both the U and V thresholds were set to 0.03 deg. As can be seen in Fig. 12, the spin axis attitude error is maintained within a ± 0.04 deg window.

The effect of uncertainty in the estimator feedforward secular solar torque parameter $\hat{A}_o$ on SAC performance is illustrated in Figs. 13 and 14, which show the spin axis attitude profile with perfect and a 50% (low) mismatch between the actual secular torque (0.20 deg/day) value and the ground-commanded value. The result of the mismatch is a slight increase of roll error of 0.006 deg and an increased error of 0.017 deg in yaw. This effect is due to the (second-order) nature of the estimator and can be predicted using standard control techniques. The "high" roll estimator bandwidth make the roll axis much less sensitive to modeling errors.

The ability of the system to acquire or "capture" the spin axis attitude, i.e., achieve normal steady-state tracking from initial roll and yaw attitude errors (no solar torque) is illustrated in Fig. 15. For initial errors of 0.20 deg in both states the spacecraft roll attitude error is reduced to 0.02 deg ~½ h, while the overall estimation error is reduced to ~0.01 deg within 22 h.

VII. Conclusions

This paper has presented the design and performance of an onboard spin axis controller for either spin- or dual-spin-stabilized spacecraft. The design presented, implemented within the control system microprocessor, uses existing attitude control system sensors and signal processing to provide autonomous precession control of the satellite spin axis attitude with minimum hardware impact. The spin axis controller design will significantly increase spacecraft autonomy and reduce the ground station workload and cost for those applications in which precision payload pointing is required in the presence of severe environmental disturbances.

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