Roll/Yaw Control of a Flexible Spacecraft Using Skewed Bias Momentum Wheels

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Roll/yaw attitude control of a bias momentum stabilized flexible spacecraft is presented. The roll/yaw normal controller with a nonminimum phase control logic is shown to be very insensitive to the structural mode uncertainty. However, if this controller is switched on at the termination of a thruster-controlled stationkeeping maneuver, it cannot provide sufficient nutation damping by itself to guarantee that the initial transients will remain within specifications. Consequently, a roll/yaw transition controller has been designed to meet the pointing requirements during the transition from the stationkeeping mode to the normal mode. The new transition controller design requires consideration of the solar array flexibility because a control logic, designed to stabilize the nutation mode neglecting the solar array flexibility, could destabilize the solar array flexible mode.

Introduction

THREE-AXIS bias momentum stabilization has been popular for many years for geosynchronous communications satellites; e.g., CTS, INTELSAT-V, SATCOM, TDRS, and INSAT. The ARABSAT and TV-SAT to be launched will also be bias momentum stabilized in geosynchronous orbit. The unique feature of the bias momentum stabilization is the ability to control yaw error passively without a direct yaw measurement.[1-6] The bias momentum provides gyroscopic stiffness to the environmental disturbances, primarily to the solar radiation pressure torque.

The increased demand of electrical power for communications and/or for direct TV broadcastings leads to large flexible solar panel arrays for three-axis-stabilized spacecraft. Higher control bandwidth to maintain an accurate body pointing of a three-axis-stabilized flexible spacecraft during thrusting maneuvers results in significant structural mode interactions with attitude control systems using thrusters.[7,8] Dynamic analysis and control design for large flexible space structures have also been the subjects of active research.[9-14]

This paper presents the results of an on-orbit roll/yaw attitude control system design for a bias momentum stabilized spacecraft shown in Fig. 1. This spacecraft utilizes the flight-proven INSAT attitude control subsystem. Wie and Plescia[8] have presented an attitude control system design for this spacecraft during stationkeeping maneuvers, where the structural modes strongly interact with the attitude control system using thrusters. The on-orbit normal mode control system consists of two momentum wheels skewed with respect to the pitch axis, a smaller yaw reaction wheel for backup mode, redundant two-axis Earth sensors to measure roll and pitch attitude references, and thrusters to provide momentum desaturation torques.

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Nonminimum phase control logic proposed by Terasaki[1] has been implemented for the roll/yaw normal controller of the INSAT spacecraft. Lebsock[6] has discussed the sizing criteria for the wheels as well as the on-orbit control system design for the INSAT spacecraft. The INSAT roll/yaw normal controller with a nonminimum phase control logic has a nutation damping ratio of 0.15 and a control bandwidth in the vicinity of the open-loop nutation frequency. Although Terasaki[1] and Lebsock[6] were not concerned with the flexible structural modes in the control design, this paper will show that their control logic is very insensitive to the structural mode uncertainty. All of the structural modes of the INSAT during on-orbit normal mode operations are gain stabilized by the steep roll-off at a frequency well below the first structural mode frequency. However, the roll/yaw normal controller has unusual transient responses; for example, the roll attitude initially increases before decreasing for the initial roll error correction, and the closed-loop roll transient peak exceeds the open-loop transient peak because of the nonminimum phase characteristics.

If the roll/yaw normal controller is switched on at the termination of a thruster-controlled stationkeeping maneuver, it cannot provide sufficient nutation damping by itself to guarantee that the initial transients will remain within specifications. Consequently, a roll/yaw transition controller has been designed for the spacecraft in this paper, to damp out the spacecraft residual rates which exist at the end of the stationkeeping operations to levels that the normal controller can handle. The new transition controller design requires consideration of the solar array flexibility, bcause a control logic designed to stabilize the nutation mode neglecting the solar array flexibility could destabilize the structural mode. Since there was no attempt to modify the flight-proven INSAT roll/yaw normal controller, separate normal and transition controllers have been designed for the spacecraft in this paper.

A cross-axis transfer function from the wheel yaw momentum command to the roll attitude output will be derived and used in the classical control analysis and design. This transfer function has nonalternating poles and zeros along the imaginary axis. In other words, the nutation and structural modes are *unstably* interacting with each other in the cross-axis channel. The roll/yaw control of this flexible spacecraft via gyroscopic coupling has a problem of unstably interacting structural modes similar to the one that exists in OSO-8[15] and Galileo spacecraft.[16] Digital simulation results will be used to validate a classical control design approach based on the approximate transfer function, and to justify the importance of structural mode interactions in the new transition controller design.

Control System Description

Figure 2 contains a sketch of the wheels configuration. The x axis is nominally in the flight direction, the y axis is normal to the orbit plane, and the z axis is directed toward the Earth. The spacecraft control axes are coincident with the spacecraft principal axes. Two skewed momentum wheels provide pitch bias momentum of 91.4 Nms with ± 2.5 Nms momentum modulation capability. The yaw reaction wheel has momentum modulation capability of ± 0.65 Nms. The pitch bias momentum vector has been chosen along the negative pitch axis to have a stable interaction of the orbit rate mode with the nutation mode in the closed-loop feedback control, which will be discussed in detail later.

These three wheels provide three sets of bias momentum: 1) the prime mode (MW 1 and 2), 2) backup mode 1 (MW 1 and RW), and 3) backup mode 2 (MW 2 and RW). For the prime mode, the pitch attitude is controlled by normal modulation of the pitch angular momentum (H_y) of the two momentum wheels about their nominal bias momentum (H) in response to the error signal from the pitch channel of the Earth sensor. The roll/yaw attitude is controlled by differentially modulating the yaw angular momentum (H_z) of the two momentum wheels in response to the error signal from the roll channel of the Earth sensor. Momentum command distribution matrices are used to convert the pitch and yaw momentum commands (H_{yc} and H_{zc}) to the wheel momentum commands (H_{1c}, H_{2c}, and H_{3c}). Similarly, the momentum measurement distribution matrix is used to convert the angular momentum of each wheel, as measured by its tachometer (H_{1m}, H_{2m}, and H_{3m}), to the angular momentum in body axes (H_{ym} and H_{zm}).

The roll/yaw normal controller actually consists of the yaw momentum control loop and the yaw momentum desaturation loop as shown in Fig. 3, which is similar to the INSAT normal mode control system.[6] The transition controller uses gyros to directly measure the roll rate. The gyros provide yaw attitude reference and three-axis rates during the stationkeeping operations, and will be turned off after the transition operations. The roll/yaw secular momentum due to external disturbance torques is stored either as yaw attitude error or in wheel momentum, until a desaturation torque is applied to the spacecraft. During the normal on-orbit operations, the yaw error is neither measured nor estimated. The yaw error in a limit cycle at orbit rate is controlled indirectly by the yaw momentum desaturation loop with the measurement of wheel yaw momentum. The yaw momentum contol loop provides active roll but passive yaw control with secularly increasing yaw error due to external disturbances.

Flexible Spacecraft Models

The hybrid-coordinate model[17] of a spacecraft with flexible appendages was used for the control analysis and design. A single solar array flexibility model referenced to the spacecraft center of mass is given in Table 1. The orientation of the solar arrays with respect to the spacecraft main body depends on orbital position and, thus, on orbital time. In this paper, solar array orientation at 6 a.m. is considered as a nominal configuration for the subsequent analysis and design. Orbit time of 6 a.m. yields out-of-plane bending modes in yaw axis and in-plane bending modes in the roll axis. The low-frequency characteristics of the first in-plane bending mode is caused by array yoke deformation. The antenna reflector flexibility is neglected because of its high frequency and smaller rigid-elastic coupling scalar.

During on-orbit normal operations, both arrays are always pointing to the sun, while the main body is pointing to the Earth. This results in very slowly changing modal frequencies and mode shapes. However, for control design purposes, this will be treated as a time-invariant system with a known range of modal characteristics.

The linearized equations of motion of the spacecraft with nominal 6 a.m. solar array configuration can be described as

Table 1 Single solar array flexibility model at 6 a.m.

Mode description[a]	Cantilever frequency, rad/s σ	Coupling matrix, $\sqrt{\text{kg-m}^2}$ Roll δ_x	Pitch δ_y	Yaw δ_z
OP-1	0.885	0	0	35.372
OP-2	6.852	0	0	4.772
OP-3	16.658	0	0	2.347
OP-4	33.326	0	0	0.548
T-1	5.534	0	2.532	0
T-2	17.668	0	0.864	0
T-3	33.805	0	0.381	0
IP-1	1.112	35.865	0	0
IP-2	36.362	2.768	0	0

[a]OP = out-of-plane, T = torsion, IP = in-plane.

Table 2 Spacecraft nominal parameters at 6 a.m.

Description	Value
Spacecraft inertia, kg·m²	3026 (roll)
	440 (pitch)
	3164 (yaw)
Pitch bias momentum, Nms	91.4
Nutation frequency, rad/s	0.03
Structural mode zeros, rad/s	0.885, 1.112
Structural mode poles, rad/s	1.928, 2.879
Wheels skew angle, deg	2.5
Motor time constant, s	4.0

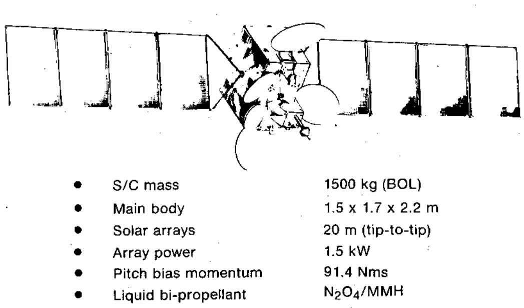

Fig. 1 Three-axis-stabilized geosynchronous communications satellite.

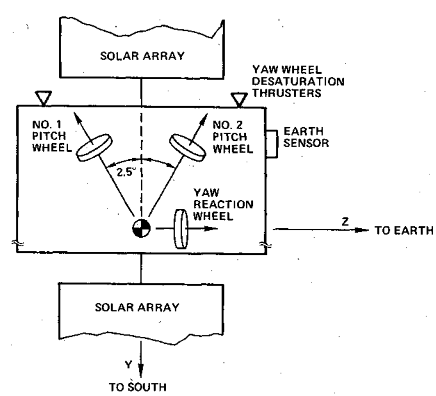

Fig. 2 Skewed wheels configuration.

Rigid body:

$$I_x\ddot{\phi}+(a+nH)\phi+(b+H)\dot{\psi}-nH_z+D_x\ddot{q}_x=M_x \tag{1}$$

$$I_y\ddot{\theta}+d\theta+\dot{H}_y+D_y\ddot{q}_y=M_y \tag{2}$$

$$I_z\ddot{\psi}+(c+nH)\psi-(b+H)\dot{\phi}+\dot{H}_z+D_z\ddot{q}_z=M_z \tag{3}$$

Solar arrays:

$$\ddot{q}_x+\sigma_x^2q_x+D_x\ddot{\phi}=0 \tag{4}$$

$$\ddot{q}_y+\sigma_y^2q_y+D_y\ddot{\theta}=0 \tag{5}$$

$$\ddot{q}_z+\sigma_z^2q_z+D_z\ddot{\psi}=0 \tag{6}$$

Wheel/motor:

$$T_m\dot{H}_y+H_y=H_{yc} \tag{7}$$

$$T_m\dot{H}_z+H_z=H_{zc} \tag{8}$$

where

$$a=4n^2(I_y-I_z),\qquad b=-n(I_x-I_y+I_z),$$

$$c=n^2(I_y-I_x),\qquad d=3n^2(I_x-I_z)$$

(ϕ,θ,ψ) are small roll, pitch, and yaw attitude errors about the Earth-pointing body-fixed reference (x,y,z), n the orbit rate, H the nominal pitch bias momentum, (I_x,I_y,I_z) the spacecraft principal moments of inertia, (q_x,q_y,q_z) the cantilever modal coordinates, $(\sigma_x,\sigma_y,\sigma_z)$ the cantilever modal frequencies, (D_x,D_y,D_z) the rigid-elastic coupling scalars of *two* solar arrays, (H_y,H_z) the angular momentum of the wheels in body axes, (H_{yc},H_{zc}) the wheels angular momentum commands in body axes, (M_x,M_y,M_z) the external torques, and T_m the motor time constant.

For simplicity, only a single dominant cantilever mode in each axis is included in the preceding equations. A passive structural damping ratio of 0.002-0.005 is usually added to the solar array equations, or it can be directly added to the transfer function later. The pitch dynamics is uncoupled from the roll/yaw dynamics. Thus, only the coupled roll/yaw dynamics and control will be discussed in this paper. The constants (a, b, c, and d) at geosynchronous orbit can be neglected compared to the relatively large value of pitch bias momentum.

Open-loop characteristics of the nutation dynamics can be easily observed from the following roll/yaw responses to the initial conditions, neglecting the orbit rate and the solar array flexibility;

$$\phi(t)=\phi_0+\frac{\dot{\phi}_0}{\lambda}\sin\lambda t-\frac{I_z\dot{\psi}_0}{H}(1-\cos\lambda t) \tag{9}$$

$$\psi(t)=\psi_0+\frac{\dot{\psi}_0}{\lambda}\sin\lambda t+\frac{I_x\dot{\phi}_0}{H}(1-\cos\lambda t) \tag{10}$$

where $\lambda\triangleq H/(I_xI_z)^{0.5}$ = nutation frequency.

These equations indicate that a positive initial roll rate produces a positive yaw response with a steady-state offset, and that the positive initial yaw rate produces a negative roll response with a steady-state offset. From these equations, the maximum roll/yaw errors due to a single roll/yaw thruster firing with a small pulse width can be estimated easily.

The roll/yaw control analysis and design can be accomplished using the classical transfer function approach. The cross-axis transfer function of the spacecraft, from the wheel yaw momentum command, $u\triangleq H_{zc}$, to the roll attitude error can be obtained as

$$\frac{\phi(s)}{u(s)}=\frac{(s^2/z_1^2+1)(s^2/z_2^2+1)}{H(s^2/\lambda^2+1)(s^2/p_1^2+1)(s^2/p_2^2+1)(T_ms+1)} \tag{11}$$

where

$$p_1\cong z_1/(1-2\delta_z^2/I_z)^{0.5},\quad p_2\cong z_2/(1-2\delta_x^2/I_x)^{0.5}$$

$$z_1=\sigma_z,\qquad z_2=\sigma_x$$

(σ_x,δ_x) and (σ_z,δ_z) are the cantilever frequencies and coupling scalars of the dominant in-plane and out-of-plane bending cantilever modes, respectively, of a singe array given in Table 1. The effect of the pitch bias momentum on the free-free structural frequencies is negligible here.

It should be emphasized that the above structural pole-zero relations were obtained using the wide spectral separation between the nutation and structural modes. Also, only a single dominant mode in each axis has been considered. The transfer function, Eq. (11), with the approximate pole-zero relations will be a basis of the subsequent classical control analysis and design in this paper.

Using the above pole-zero relations, the structural poles at 6 a.m. can be found as $p_1=1.93$ rad/s in yaw axis and $p_2=2.87$ rad/s in roll axis. These approximate values agree closely with the values obtained by a computer program for the model given by Eqs. (1-8) with additional higher frequency structural modes given in Table 1. The pole and zero of the orbit rate mode are of the same order of magnitude as the orbit rate, n, thus they are not included in Eq. (11). Exact relative location of the orbit rate mode pole and zero can be deter-

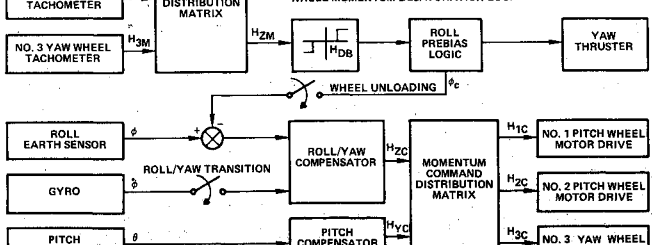

Fig. 3 On-orbit attitude control system.

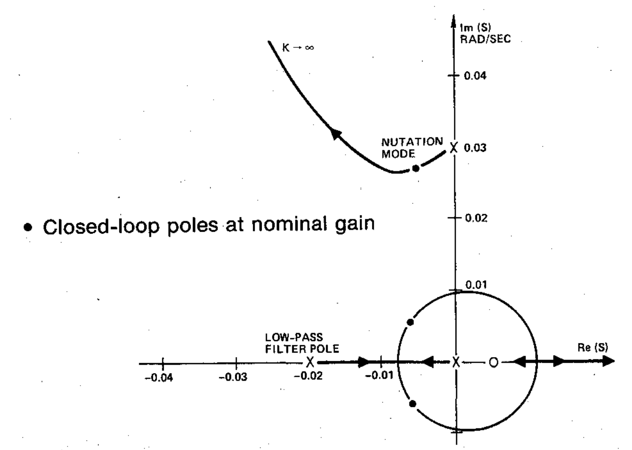

Fig. 4 Root locus for roll/yaw normal controller (structural modes are not shown).

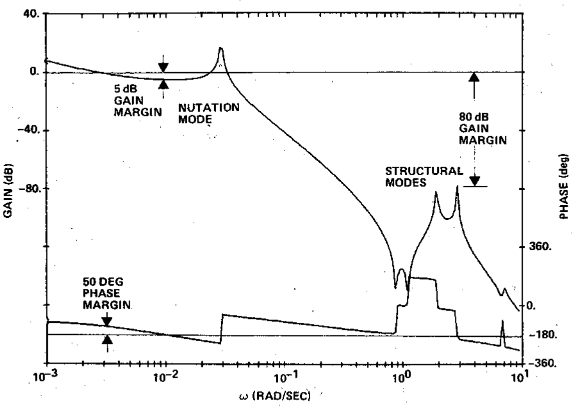

Fig. 5 Bode plot for roll/yaw normal controller.

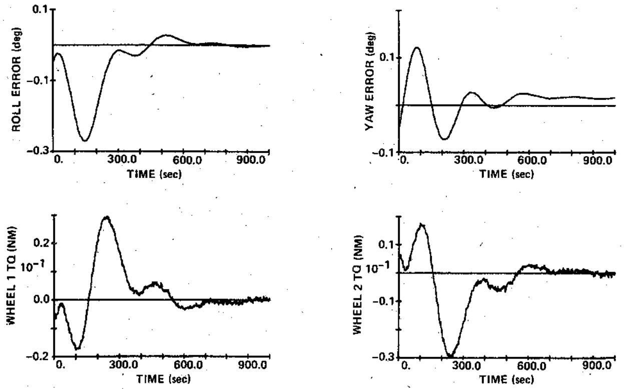

Fig. 6 Nominal responses of roll/yaw normal controller.

mined from the equations including the constants a, b, and c. The relative location determines the closed-loop stability of the orbit rate mode, and it depends on the pitch bias momentum direction. For a spacecraft with $I_z > I_y$, the pitch bias momentum vector is chosen along the negative pitch axis for a stable interaction between the nutation and orbit rate modes. The orbit rate mode stability problem due to the yaw momentum control loop is insignificant in practice; it depends on the yaw momentum desaturation loop and the solar torques. However, it is theoretically closed-loop stable by the stable interaction with the nutation mode.

It is interesting to note that the structural mode poles and zeros are *not* alternating along the imaginary axis. It has been known that if the structural poles and zeros alternate along the imaginary axis then the structural modes are stably interacting with the rigid-body mode.[10] Because of a stable interaction, a control logic designed to stabilize the nutation mode neglecting the orbit rate mode naturally stabilizes the orbit rate mode. However, a control logic designed to stabilize the nutation mode neglecting the structural modes could destabilize the structural modes, because they are not stably interacting.

Without the gravity gradient torque which is, in fact, negligible for small attitude errors in geosynchronous orbit, the orbit rate mode will have an exact pole-zero cancellation. In that case, the total roll/yaw angular momentum of the spacecraft plus wheels are uncontrollable with wheel yaw momentum, i.e., an internal torque cannot change the total angular momentum. Using wheel yaw momentum as a control, we can only transfer the angular momentum from the spacecraft to the wheels.

Roll/Yaw Normal Controller

The roll/yaw normal controller consists of the yaw momentum control loop and the yaw momentum desaturation loop

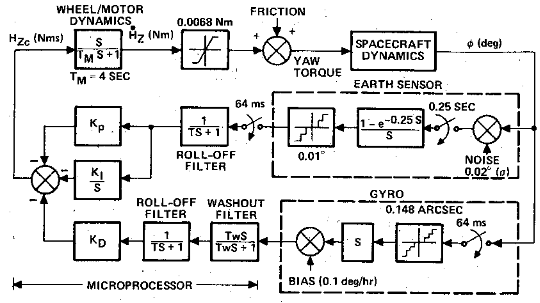

Fig. 7 Roll/yaw transition controller.

(see Fig. 3). Herein only the yaw momentum control loop will be discussed. The Terasaki nonminimum phase control logic,[1,6] which is a special case of the conventional proportional-integral-derivative (PID) controller, is represented as

$$u(s) = -\frac{K(1-T_z s)}{s(Ts+1)}\phi(s) \tag{12}$$

where K is the positive loop gain, T_z the time constant of the nonminimum phase zero, and T the time constant of the first-order low-phase filter. Referring to the conventional PID controller parameters, we have $K_P = -KT_z$, $K_I = K$, and $K_D = 0.0$. The low-pass filter in Eq. (12) is utilized for a phase-lag compensation of the nutation mode, which results in a gain stabilization of all of the structural modes by the steep roll-off at a frequency well below the first structural frequency. A nonzero passive structural damping is necessary for a gain stabilization of all of the structural modes. Because of a wide spectral separation, the controller parameters can be selected for a rigid spacecraft model. Stability analysis can then be performed by closing the control loop with the flexible spacecraft model.

Neglecting the solar array flexible modes, the closed-loop roll transfer function from the spacecraft initial conditions can be found as

$$\phi(s) = \frac{[(s^2+\lambda^2)\phi_0 + s\dot{\phi}_0 - \dot{\psi}_0 H/I_x](s+1/T)}{s^4 + s^3/T + \lambda^2 s^2 + s(H-KT_z)\lambda^2/HT + K\lambda^2/HT} \tag{13}$$

where the motor dynamics with a time constant of 4 s has also been neglected. Without a feedback control ($K=0.0$), Eq. (13) simply becomes the open-loop roll transfer functions from the initial conditions

$$\phi(s) = \frac{\phi_0}{s} + \frac{\dot{\phi}_0}{s^2+\lambda^2} - \frac{H\dot{\psi}_0/I_x}{s(s^2+\lambda^2)} \tag{14}$$

which can be inverted to get Eq. (9). The significance of Eq. (13) is that it shows individual as well as combined effects of the initial conditions on the closed-loop roll transient response. It is well known that the type of transient response is determined by the closed-loop poles, while the shape of the transient response is determined primarily by the closed-loop zeros. From Eqs. (13) and (14), we notice that the roll responses to the initial roll rate and initial yaw rate have a 90-deg phase difference. Such property is independent of the control logic. However, the yaw responses to the initial conditions do not have such a property.

The denominator of Eq. (13), which is the closed-loop characteristic equation, can be factored into two pairs of roots with natural frequencies and damping ratios given approximately by

$$\omega_1 \cong \lambda, \qquad \zeta_1 \cong \omega_2^2 T_z / 2\omega_1 \tag{15}$$

$$\omega_2 \cong (K/HT)^{0.5}, \qquad \zeta_2 = (H-KT_z)/(2\omega_2 HT) \qquad (16)$$

Using the design rules of $\zeta_1\omega_1 \cong \zeta_2\omega_2$, $\zeta_1 \cong = 0.175$, and $\zeta_2 \cong 0.707$,[4] the following approximate gain formula can be obtained:

$$T = (0.7\lambda)^{-1}, \quad T_z = (0.175\lambda)^{-1}, \text{ and } K = 0.0875H\lambda \qquad (17a)$$

or

$$K_P = -0.5H, \quad K_I = 0.0875H\lambda, \text{ and } K_D = 0.0 \qquad (17b)$$

The closed-loop nutation frequency ω_1 is chosen in the vicinity of the nutation frequency. After determining preliminary parameters using Eqs. (17) derived for a rigid spacecraft, a stability analysis for the flexible spacecraft can be performed using the s-domain root locus and the Bode plot with a Padé approximation of the digital Earth sensor sample-and-hold delay of 0.25 s and the microprocessor computational delay of 48 ms and sampling period of 64 ms.[8] A root locus vs K is shown in Fig. 4. Only the lower frequency region is shown because the structural poles do not move significantly. The pole of the in-plane bending mode in roll axis ($p_2 = 2.87$ rad/s) actually moves to the right, which results in a reduced damping ratio of 0.0019 from its open-loop damping of 0.002. Such a small effect on the closed-loop damping can be neglected in practice.

A Bode plot which shows such an insignificant structural mode interaction is given in Fig. 5. All of the structural modes are gain-stabilized with an 80 dB gain margin by the passive damping ratio of 0.002 and the steep roll-off at a frequency well below the first structural mode frequency. There is also a limitation on the active nutation damping, because the lower-frequency control mode is gain-stabilized with only a 5 dB gain margin. The gain margin is independent of the spacecraft inertia, but dependent on the pitch bias momentum which is measured quite accurately by the wheel tachometers. Thus, the 5 dB gain margin is acceptable. The nutation mode is phase-stabilized with a 50 deg phase margin.

From Fig. 5, it can be seen that the Terasaki control logic is quite robust with respect to the structural mode uncertainty. Thus, we could completely neglect the structural modes in the design of the roll/yaw normal controller, and even in the digital simulations. However, such a good robustness has been obtained by sacrificing some performance, especially during transient periods. During on-orbit normal operations, the environment is very quiet except for the solar radiation pressure torques. Thus the normal controller even with a small nutation damping ratio provides acceptable performance.

Figure 6 shows the results of digital simulation, where the initial conditions were chosen from a steady-state limit cycling at the end of a stationkeeping maneuver. The wheel torque demand is well below the saturation level and the structural modes are almost unexcited. However, the problem is that the roll transient peak exceeds the pointing requirement of 0.1 deg. Consequently, a roll/yaw transition controller has been designed to damp out the residual rates that exist at the end of the stationkeeping mode operations to levels that the normal controller can handle without exceeding the pointing requirement.

Roll/Yaw Transition Controller

The performance requirement of the roll/yaw transition controller was to maintain the short-term attitude errors within ± 0.1 deg in roll and ± 0.16 deg in yaw for all possible handoff conditions from the steady-state limit cycling of the stationkeeping mode.[8] A design goal was to have the roll/yaw rate damped within no more than 5 min without an excessive wheel torque saturation to typical on-orbit magnitudes which the roll/yaw normal controller can handle.

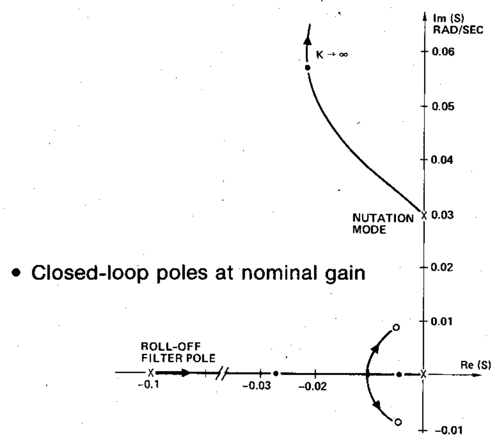

Fig. 8 Root locus for roll/yaw transition controller (structural modes are not shown).

To increase the active nutation damping ratio, a PID controller has been investigated in Ref. 18 where the roll rate signal is obtained by pseudo-differentiation of the roll error signal. The proportional gain is made almost zero to have the closed-loop nutation frequency in the vicinity of the open-loop nutation frequency. A different control logic for a bias momentum stabilized spacecraft can also be found in Ref. 19.

For the spacecraft in this paper, attitude error and rate are directly measured during the stationkeeping mode operations. Thus, the roll/yaw transition contoller can utilize the roll/yaw attitude error and rate measurements. A proportional and integral feedback of the yaw attitude error, in addition to the PID control of the roll error, can directly control the roll/yaw attitude error. However, the integral feedback of the yaw error may result in another switching transient, when the roll/yaw normal controller without the yaw feedback is turned on. Thus, the roll/yaw transition controller utilizes only the roll rate measurement from the gyro and the roll error measurement form the Earth sensor.

A direct roll rate feeback as well as proportional and integral feedback of the roll error signal into the yaw momentum control loop stabilizes the nutation mode with increased bandwidth and damping ratio. It could, however, destabilize the structural mode with passive damping ratio of 0.002 if the control design did not consider the solar array flexibility.

Because of a wide spectral separation between the nutation mode and the first structural mode, a gain stabilization of all of the structural modes using the first-order roll-off filter was possible with some degradation of the nutation mode stability and performance. A higher-order compensation such as a notch filtering[8] was avoided because of the structural mode uncertainty. A phase-lead stabilization of the structural modes was also avoided because of the control-loop time delay which produces larger phase lag at a higher frequency. The new transition control-loop block diagram is shown in Fig. 7. A high-pass filter to wash out the residual drift rate bias of the gyro is provided in the rate loop. This washout filter with small pole-zero separation near origin has no effect on overall stability and performance, thus it will not be included in the subsequent analysis.

For a preliminary design, the roll/yaw transition control logic shown in Fig. 7 can be described as

$$u(s) = -\frac{(K_D s^2 + K_P s + K_I)}{s(Ts+1)}\phi(s) \qquad (18)$$

where K_P is the proportional gain, K_I the integral gain, K_D the derivative or rate gain, and T the roll-off filter time constant. For a rigid spacecraft, the roll-off filter becomes a standard noise filter with corner frequency far outside the control bandwidth. Because the nutation mode is phase-lead stabilized by

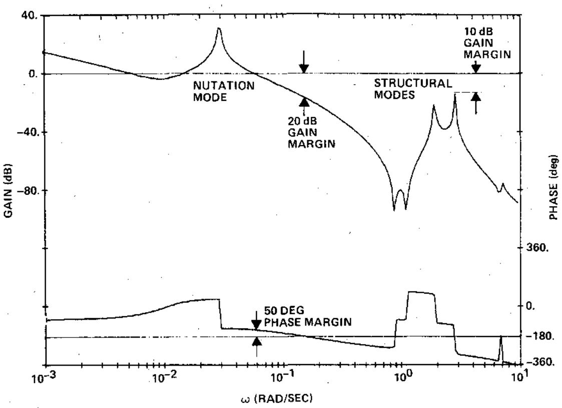

Fig. 9 Bode plot for roll/yaw transition controller.

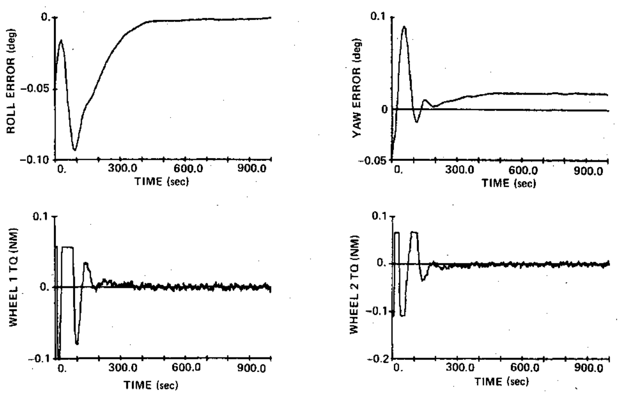

Fig. 10 Nominal responses of roll/yaw transition controller.

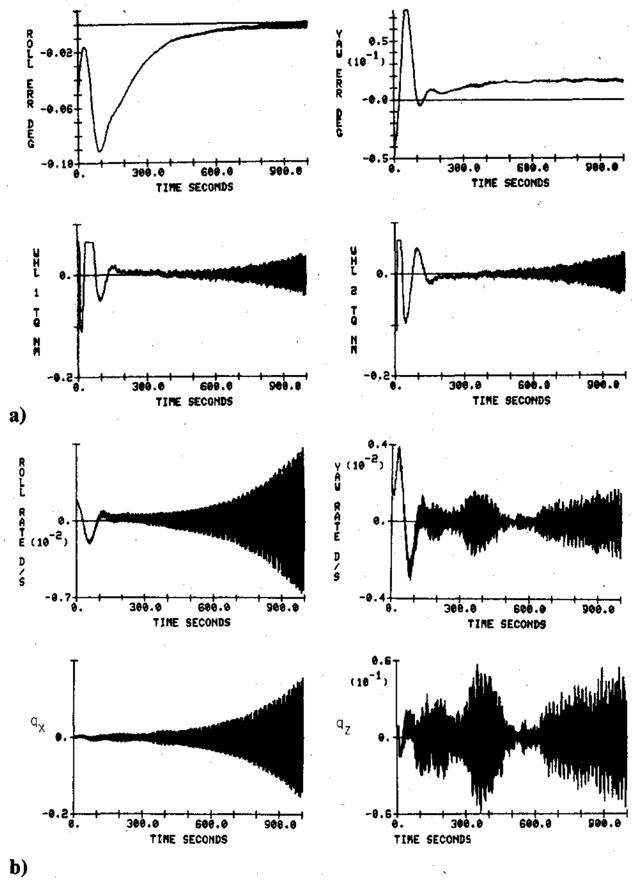

Fig. 11 Structural mode instability of transition controller without roll-off filter.

the rate feedback, it does not need any phase lag from the low-pass filter which has been utilized as a phase-lag compensation element in the Terasaki control logic. But the low-pass roll-off filter with corner frequency near the control bandwidth is necessary here to gain-stabilize the structural modes.

The closed-loop roll transfer function from the initial conditions neglecting the solar array flexibility can be obtained as

$$\phi(s) = \frac{[(s^2+\lambda^2)\phi_0 + s\dot{\phi}_0 - \dot{\psi}_0 H/I_x](s+1/T)}{s^4+s^3/T+s^2(1+K_D/HT)\lambda^2+s(H+K_P)\lambda^2/HT+\lambda^2 K_I/HT} \tag{19}$$

For the chosen closed-loop poles, the following approximate gain formula can be used for the selection of preliminary controller parameters:

$$T=(s_1+s_2+2\zeta\omega)^{-1} \tag{20a}$$

$$K_P=\omega^2(s_1+s_2)HT/\lambda^2-H \tag{20b}$$

$$K_I=s_1 s_2 \omega^2 HT/\lambda^2 \tag{20c}$$

$$K_D=[\omega^2+2\zeta\omega(s_1+s_2)-\lambda^2]HT/\lambda^2 \tag{20d}$$

where s_1 and s_2 are negative real roots, and ω and ζ are closed-loop nutation frequency and damping ratio, respectively. Since the roll transient peak is inversely proportional to the square of the closed-loop nutation frequency, the selection of the closed-loop poles is primarily dependent on the pointing requirement.

Figures 8 and 9 show a root locus and Bode plot for the roll/yaw transition controller, respectively. It can be seen that the structural model has a 10-dB gain margin. Figure 10 shows digital simulation results with the same initial conditions for Fig. 6. The transition controller meets the pointing requirement, but with a reduced structural mode stability margin compared to the normal controller with Terasaki control logic. Initially, there is torque saturation which has no critical effect on the overall stability. The different positive and negative torque saturation limits are due to the different effects of drag on the wheel with a biased speed; less drag effect for slowing down.

From Fig. 11, it can be seen that the transition controller without the roll-off filter destabilizes one of the structural modes. The in-plane bending mode ($p_2 = 2.87$ rad/s) in roll axis is destabilized by the transition controller without the roll-off filter. Figure 11 validates the analytic prediction of the 10-dB structural gain margin and also justifies the importance of the structural mode interaction in the new transition controller design.

However, a passive damping ratio of 0.01 could make it stable even without the roll-off filter. Thus, it is emphasized that the so-called control and observation spillover problem should not be exaggerated without considering the degree of instability, which might be naturally controlled by reasonable passive structural damping. A simple first-order filtering was adequate to prevent the structural mode instability which might occur in the transition mode operation. In general, a higher-order structural compensation is required in case of significant structure/control interactions.[7,8,15,16]

Conclusions

In this paper, a cross-axis transfer function for a three-axis bias momentum stabilized flexible spacecraft has been derived. The roll/yaw control design has been performed using this approximate transfer function. Digital simulations have validated such a classical design approach and justified the importance of the structural mode interaction in the new roll/yaw transition controller design. The Terasaki non-minimum phase control logic has been shown to be very robust with respect to the structural mode uncertainty. It is recommended that the degree of instability and the existence of passive structural damping should be considered in an attitude control system design for a three-axis bias momentum stabilized spacecraft with flexible solar panel arrays.

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