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Magnetic Attitude Control of a Momentum-Biased Satellite in Near-Equatorial Orbit

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A novel closed-loop magnetic attitude control scheme is described for momentum-biased, near-equatorial orbit satellites. Unlike the other schemes proposed so far, the controller presented here performs both the attitude correction and nutation damping, thus obviating the need for a separate nutation damping mechanism. The magnetic torquer is placed along the roll axis of the spacecraft. The roll error ϕ, obtained from the Earth sensor, is filtered out into two components; one varying at orbital frequency ϕ_o, and the other varying at nutational frequency ϕ_n. The control dipole moment M_c of the magneto-torquer is governed by the control law, $M_c = K_2'\phi_n - K_1'\phi_o$, where K_1' and K_2' are constants. Analytical expressions for time response of the system and conditions for stability are derived, using linearized equations of motion. The roll/yaw dynamics of the satellite were simulated on an analog computer, and the simulation and analytical results matched well. Also, the simulation results indicate that enough damping is provided in the yaw channel. Design aspects such as choice of feedback gains and saturation characteristics of the magneto-torquer are discussed.

Introduction

THE current trend in satellite attitude control shows that three-axis stabilization with momentum-biased wheels will be adequate to meet the stringent pointing requirements of present and future communication satellites. The effect of environmental forces is compensated by reaction control systems using hydrazine monopropellant, cold gas, etc. The need for long-life communication satellites has led to the development of the Magnetically Suspended Momentum Wheel (MSMW), which has an operational life of fifteen years. However, the use of the chemical thrusters for roll/yaw control for such a long time becomes questionable due to the increased number of thruster firings, propellant weight, and associated contamination problems. Electrical thrusters, though gaining importance, do not have a demonstrated life of more than two years. Hence, there is a need for the evolution of semipassive controllers utilizing the environmental forces such as Earth's magnetic field, solar radiation pressure, etc.

The utility of magnetic torquers for satellite control has been well established for near-Earth orbits.[1-5] Control laws for both error reduction and nutation damping for spin-stabilized and dual-spin-stabilized satellites were obtained by Wheeler[4] and Alfriend,[5] respectively.

The extension of magnetic torquing to satellites in geosynchronous altitude is receiving considerable attention due to the successful demonstration in the Lincoln Experimental Satellite 5 (LES 5) and RCA's SATCOM. The Lincoln Experimental Satellite[6] is the first geosynchronous satellite to use electromagnets for attitude control purposes. Two electromagnets placed along the transverse axes of the spacecraft and suitably excited by the outputs from four photo diodes, kept the spin axis of the satellite within 2.6 deg of the orbit normal. Modi[7] proposed magnetic solar hybrid controller for three-axis nutation damping and attitude control of dual-spin satellites in near-equatorial orbit. It is shown that the bang-bang controller is useful in achieving any arbitrary orientation. The RCA SATCOM[8] employs air-cored coils to counteract the effect of the solar radiation pressure. Spencer[9] developed proportional control laws for slewing maneuvers and wheel dumps for a momentum-biased astronomical observatory in equatorial orbits. Recently, Lacombe[10] analyzed the feasibility of magnetic torquing in the case of a single-degree-of-freedom angular momentum type control of a large satellite.

The present work investigates a closed-loop magnetic attitude control system for roll/yaw control of a momentum-biased body-stabilized APPLE spacecraft, the first Indian experimental communication satellite, to be launched in the middle of 1980 by the Ariane launch vehicle. The investigation is concerned with the development of a suitable control law to govern the dipole moment of the magnetotorquer placed along the roll axis of the satellite (Fig. 1) so as to correct the attitude error and damp out nutational oscillations. An interesting feature of the proposed controller is that the nutational oscillations, arising due to transverse torquing, are also damped out, thus eliminating the need for WHECON[11] or half-precession cycle damping by reaction jets. The yaw control is obtained by roll/yaw coupling established by the momentum wheel, and is quite adequate for most of the applications. The proposed control law can be very easily adopted for yaw control as well. An additional feature of the control system is that it can provide long-term attitude stability such as that required by certain meteorological missions, (.004 deg/30 min), which otherwise cannot be met by momentum wheels alone.

The analysis, design and simulation results of the proposed control scheme are discussed in the subsequent sections. The implementation of the control law is also covered. The subsequent discussion and results refer to a satellite in geosynchronous orbit. However, the results can be applied to satellites in near-Earth equatorial orbits.

Geomagnetic Field at Synchronous Altitude

Extensive observations of the geomagnetic field by several satellites now provide a realistic picture of the Earth's magnetic field configuration. The magnitude of the geomagnetic field at any satellite station or longitude can be extrapolated from the observations made by the ATS1 and ATS6 satellites. The equatorial magnetic field at a distance of about 6.65 times that of Earth's radius consists of a steady-state geomagnetic field on which temporal variations are superimposed. On magnetically quiet days, the geomagnetic field is along the orbit normal pointing southward. External

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field sources cause a diurnal variation of 10 to 15% in the total magnetic intensity. During magnetic storms, the field at synchronous altitude can increase by 50%, decrease to zero, or even reverse in its direction; such intense magnetic storms may occur about a maximum of four times per year, depending upon solar activity. In general, the effects of storms do not significantly change the pitch component of the magnetic induction in the 6-18 hours of satellite local time.[8] The storms usually occur about noontime. The recovery of the geomagnetic field after a severe storm may require several days.

It is assumed here that the pitch component of the magnetic field is about 1 mG. Although the inplane component of the magnetic field may vary from 0 to 40 gammas depending upon the satellite location, it is assumed to be zero, as the error caused by it will be taken care of by the pitch control system.

Equations of Motion

Let XYZ represent a reference coordinate system with the origin at the center of mass of the satellite as shown in Fig. 1. The X-axis is along the radius vector, towards Earth's center from the satellite. The Y-axis is along the velocity vector and the Z-axis is along the orbit normal positive towards the south, thus completing the orthogonal triad. The principal axes of the spacecraft x, y, and z are to be nominally aligned with the reference axes X, Y, and Z, respectively. The orientation of the body-fixed axes with respect to the reference coordinate system can be obtained through the standard yaw (ψ), roll (ϕ), and pitch (θ) angles.

The linearized equations of motion are given as

$$\ddot{\psi}+a_1\psi+b_1\dot{\phi}=T_x/I_x \tag{1}$$

$$\ddot{\phi}+a_2\phi+b_2\dot{\psi}=T_y/I_y \tag{2}$$

where

$$a_1=-W_o(h+W_oI_y)/I_x \tag{3}$$

$$a_2=-W_o(h+W_oI_x)/I_y \tag{4}$$

$$b_1=[h+W_o(I_x+I_y)]/I_x \tag{5}$$

$$b_2=-[h+W_o(I_x+I_y)]/I_y \tag{6}$$

$$h=h_w-I_zW_o \tag{7}$$

and where I_x, I_y, and I_z are the moments of inertia about the x, y, and z axes; W_o is orbital velocity; and h_w is the angular momentum of the momentum wheel.

Laplace transformation of the linearized equations yields

$$\begin{bmatrix} s^2+a_1 & b_1s \\ b_2s & s^2+a_2 \end{bmatrix}\begin{bmatrix}\psi(s)\\ \phi(s)\end{bmatrix}=\begin{bmatrix}\dfrac{T_x(s)}{I_x}+s\psi_0+\dot{\psi}_0+b_1\phi_0\\ \dfrac{T_y(s)}{I_y}+s\phi_0+\dot{\phi}_0+b_2\psi_0\end{bmatrix} \tag{8}$$

where ψ_0 and ϕ_0 are the initial yaw and roll angles, respectively, and $\dot{\psi}_0$ and $\dot{\phi}_0$ are the initial yaw and roll rates, respectively.

The characteristic equation of the system is given by

$$(s^2+a_1)(s^2+a_2)-s^2b_1b_2=0 \tag{9}$$

The roots of the above characteristic equation indicate that the system has two natural frequencies: 1) the nutation frequency, $W_n=h/\sqrt{I_xI_y}$, and 2) the orbital frequency W_o. If there is no nutation, an error in yaw gets converted to a roll error after a quarter of an orbit. The problem now is to determine a control law so that nutation and the orbital

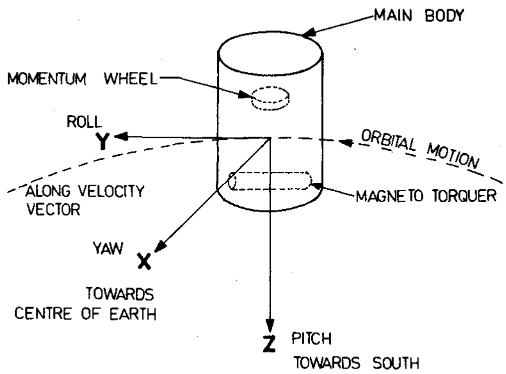

Fig. 1 Satellite configuration.

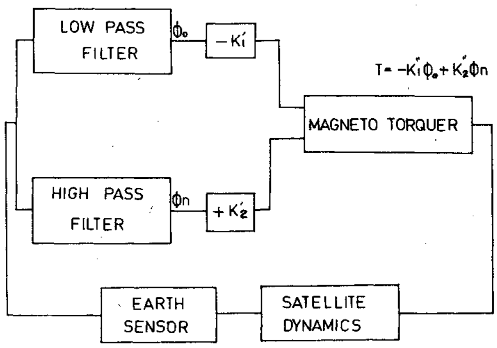

Fig. 2 Schematic of the proposed magnetic controller.

frequency components are damped out, i.e., the momentum wheel axis coincides with total angular momentum vector, and the momentum vector aligns with the orbit normal.

Proportional Controller

Let us now consider an electromagnet placed along the roll axis of the satellite. The dipole moment of the magnet is varied proportionally to the roll error. The torque produced by the interaction of the magnetorquer with the geomagnetic field is given as

$$\bar{T}=\bar{M}\times\bar{B}_E \tag{10}$$

where M is the dipole moment of the magneto-torquer and B_E is the geomagnetic field. Since the magnitude of B_E is fairly constant, a yaw torque proportional to roll error is obtained, i.e.,

$$T_X=K'\phi(t) \tag{11}$$

where K' is a feedback constant. The new characteristic equation is given as

$$(s^2+a_1)(s^2+a_2)-s^2b_1b_2+Kb_2s=0 \tag{12}$$

Equation (11) can be rewritten as

$$(s^2+\lambda_1s+W_1^2)(s^2+\lambda_2s+W_2^2)=0 \tag{13}$$

It can be shown that

$$\lambda_1=-Kb_2/W_2^2 \tag{14}$$

$$\lambda_2=Kb_2/W_2^2 \tag{15}$$

Equations (14) and (15) show that a proportional controller with a positive value of K will provide damping of the nutational motion whereas the orbital frequency component would increase. On the other hand, a negative value of K reduces the amplitude of the low-frequency component but causes nutational instability. Thus the dual objectives of error reduction and nutation damping cannot be simultaneously achieved by the proportional controller alone.

Proposed Controller

In this section a new control law is proposed such that the dipole moment of the magneto-torquer, and hence the torque developed, are governed by the following relationship:

$$M_c = -K_1'\phi_o + K_2'\phi_n \tag{16}$$

and

$$T_x = -K_1''\phi_o + K_2''\phi_n \tag{17}$$

where ϕ_o is the orbital frequency component of the Earth sensor output, and K_1'', K_2'' are feedback gains. The schematic diagram of the proposed controller is given in Fig. 2.

The signals ϕ_o and ϕ_n can be obtained by passing the output of the Earth sensor through a low-pass filter (LPF) and a high-pass filter (HPF), respectively. The transfer function of the LPF is given as

$$\phi_o(s) = \phi(s)/(1+\tau s) \tag{18}$$

Also, the output of the HPF

$$\phi_n(s) = \phi(s)\tau s/(1+\tau s) \tag{19}$$

From Eqs. (18) and (19)

$$T_x(s) = \left[\frac{-K_1''+K_2''\tau s}{1+\tau s}\right]\phi(s) \tag{20}$$

The modified system equations are

$$\begin{bmatrix} s^2+a_1 & b_1 s+\dfrac{K_1-K_2\tau s}{1+\tau s} \\ b_2 s & s^2+a_2 \end{bmatrix} \begin{bmatrix} \psi(s) \\ \phi(s) \end{bmatrix} = \begin{bmatrix} \psi_0 s+\dot\psi_0+b_1\phi_0 \\ \phi_0 s+\dot\phi_0+b_2\psi_0 \end{bmatrix} \tag{21}$$

where $K_1 = K_1''/I_x$ and $K_2 = K_2''/I_x$

The determinant of Eq. (21) is given as

$$\Delta(s) = \frac{\tau s^5+s^4+s^3[\tau(a_1+a_2)-b_1b_2\tau]+s^2(a_1+a_2-b_1b_2+b_2K_2\tau)+s(a_1a_2\tau-b_2K_1)+a_1a_2}{(1+\tau s)} \tag{22}$$

The characteristic equation can be taken as

$$s^5+s^4/\tau+s^3(a_1+a_2-b_1b_2)+s^2[(a_1+a_2-b_1b_2)/\tau+b_2K_2] +s(a_1a_2-b_2K_1/\tau)+a_1a_2/\tau=0 \tag{23}$$

Stability Analysis

Equation (23) can be rewritten as

$$(s^2+\alpha_1 s+W_o'^2)(s^2+\alpha_2 s+W_n'^2)(s+\alpha_3)=0 \tag{24}$$

Comparing the coefficients of Eqs. (23) and (24), we obtain,

$$\alpha_2 = \frac{W_n^2(b_1b_2-b_2K_2\tau+W_n^2)}{b_2K_1+W_n^4\tau} \tag{25}$$

$$\alpha_3 = \frac{W_n^2(1/\tau-\alpha_2)+\sqrt{W_n^4(1/\tau-\alpha_2)^2+4b_2K_1W_n^2/\tau}}{2W_n^2} \tag{26}$$

$$\alpha_1 = 1/\tau-\alpha_1-\alpha_2 = -b_2K_1/W_n^2 \tag{27}$$

Also,

$$W_0'^2 = a_1a_2/W_n^2\alpha_3\tau$$

Equations (25-27) show that α_1, α_2, and α_3 are all positive for practical values of satellite parameters, thus ensuring system stability for positive values of the feedback gains K_1 and K_2.

Time Constants of the Control System

Equation (24) can now be used to determine the time constants of the two modes of oscillations, namely, the orbital mode τ_o and the nutational mode τ_n. Equation (25) can now be rewritten as

$$\Delta(s) = (s^2+\alpha_1 s+W_o^2)(s^2+\alpha_2 s+W_n^2) \tag{28}$$

since $\alpha_3 = 1/\tau$.

Also, from Eq. (21) we see that

$$\phi(s) = [\phi_0 s^3+\dot\phi_0 s^2+(a_1\phi_0-b_2\dot\psi_0+b_1b_2\phi_0)s +a_1(\dot\phi_0+b_2\psi_0)]/\Delta(s) \tag{29}$$

Considering first the nutation motion, the term $(s^2+\alpha_1 s+W_n^2)$ has a pair of complex conjugate roots at nutational frequency. Hence, the time constant for nutation damping can be readily obtained as

$$\tau_n = 2/\alpha_2$$

The term $(s^2+\alpha_1 s+W_o^2)$ reprsents an overdamped system, i.e., it has two real roots. Hence, there are two time constants for the orbital mode component which are approximately in the ratio of 10:1. The subsequent analysis is done to find out which of the two time constants dominates.

Inverse Laplace transformation of Eq. (29) gives

$$\phi(t) = [e^{-\tau_1 t}(c_1\tau_1+c_2)-e^{-\tau_2 t}(c_1\tau_2+c_2)]/(\tau_2-\tau_1) +e^{-\alpha_2 t/2}(c_3\cos W_n t+c_4\sin W_n t) \tag{30}$$

where

$$c_1 = -(b_1b_2\phi_0+\alpha_1\phi_0+b_2\dot\psi_0)/W_n^2 \tag{31}$$

$$c_2 = (a_1\dot\phi_0+a_1b_2\psi_0)/W_n^2 \tag{32}$$

$$c_3 = \phi_0(1+b_1b_2/W_n^2)+(\alpha_1\dot\phi_0+b_2\dot\psi_0)/W_n^2 \tag{33}$$

$$c_4 = \dot\phi_0-a_1b_2\alpha_2\psi_0/\alpha_1-\phi_0\alpha_1 \tag{34}$$

$$\tau_1 = [\alpha_1+\sqrt{\alpha_1^2-4W_0^2}]/2 \tag{35}$$

$$\tau_2 = [\alpha_1-\sqrt{\alpha_1^2-4W_0^2}]/2 \tag{36}$$

It is obvious from Eq. (29) that both the time constants aid each other in reducing the error. However, the smaller time constant dominates over the other and hence it can be used for design purposes, i.e.,

$$\tau_0 = 1/\tau_1 = 1/\alpha_1 \tag{37}$$

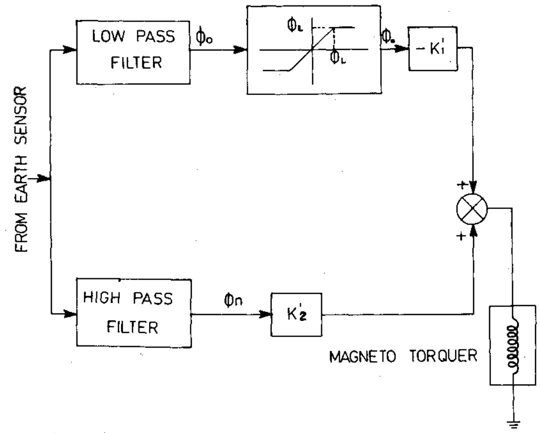

Fig. 3 Magnetic controller including saturation characteristics.

Design Considerations

As seen from earlier analysis, the system time constants depend upon the feedback gains K_1 and K_2. The magnitude of the gains is governed by factors such as maximum control torque and allowable pointing error. The control torque produced by the magneto-torquer is limited due to the weak magnetic field at synchronous altitude, and weight and power constraints. Also, the feedback gains should be selected so as to keep the satellite attitude error within allowable limits in the presence of disturbance torques. A conservative estimate of the control torque is arrived at by assuming a secular disturbance torque acting on the roll axis of the spacecraft. A value of 10^{-6} N-m is estimated as the constant disturbance torque for the APPLE spacecraft. The control torque should be about ten times greater than the disturbance torque and hence the controller should be capable of producing a torque of 10^{-5} N-m.

A realistic estimate of the control torque can be obtained by considering the effect of solar radiation torques on the satellite attitude. The solar radiation torques are cyclic in nature, varying at orbital frequency. Assuming the solar torques T_{XD} and T_{YD} as

$$T_{XD} = A \sin W_o t$$

and

$$T_{YD} = B \cos W_o t$$

$$\begin{bmatrix} \psi(s) \\ \phi(s) \end{bmatrix} = \frac{1}{\Delta(s)} \begin{bmatrix} s^2 + a_2 & -b_1 s + \dfrac{K_1 - K_2 \tau s}{1+\tau s} \\ -b_2 s & s^2 + a_1 \end{bmatrix} \begin{bmatrix} A' W_o / (s^2 + W_o^2) \\ B' s / (s^2 + W_o^2) \end{bmatrix} \tag{38}$$

Where

$$A' = A/I_x \quad \text{and} \quad B' = B/I_y$$

Hence

$$\phi(s) = [B' s^3 + (B' a_1 - A' W_o b_2) s] / \Delta(s) (s^2 + W_o^2) \tag{39}$$

Our interest is to obtain an expression for the oscillatory component, $\phi_o(t)$ of the attitude error due to solar torques. Inverse Laplace transformation of Eq. (39) yields

$$\phi_{os}(t) =$$

$$\frac{(B' W_o^2 + A' W_o b_2 - B' a_1)}{\alpha_1 W_o (W_o^2 \alpha_2^2 + W_n^4)} (W_o \alpha_2 \cos W_o t - W_n^2 \sin W_o t)$$

$$\approx -(B' W_o^2 + A' W_o b_2 - B' a_1) \sin W_o t / \alpha_1 W_o W_n^2 \tag{40}$$

The amplitude of the oscillation is given as $(B' W_o^2 + A' W_o b_2 - B' a_1)/\alpha_1 W_o W_n^2$. If, ϕ_m is the allowable error, then α_1 should be so chosen to limit $\phi_{os}(t)$ to be less than ϕ_m. Hence,

$$\alpha_1 > (B' W_o^2 + A' W_o b_2 - B' a_1) / W_o W_n^2 \phi m$$

Substituting for a_1, b_2, A' and B',

$$\alpha_1 > (A+B)/h\phi_m \tag{41}$$

Since

$$\alpha_1 \approx |b_2| K_1 / W_n^2 \tag{42}$$

$$K_1 > (A+B)/I_x \phi_m \tag{43}$$

The magneto-torquer can be either air-cored or iron-cored, and its detailed design is available elsewhere (Ref. 12). Iron-cored torquers are preferred to air-cored torquers since the former requires less power and weight. However, the iron-cored torquers exhibit saturation characteristics and hence the controller should be designed such that the linearity of the feedback is maintained within desirable limits. If ϕ_L is the error up to which proportional torque is maintained, the control torque should satisfy the condition,

$$T_x > (A+B)\phi_L / \phi_m \tag{44}$$

The constant $K_2(K_2'')$ can be determined by assuming a maximum half conning angle r_{max} and equating $K_2'' r_{max}$ with T_x. When both attitude error and nutation are present, the output of the LPF is sent through a limiter shown in Fig. 3. The output of the limiter is summed up with the nutational component and the combined signal can be used to drive the magneto-torquer. Also, the required control law can be realized with one filter alone, for example the LPF. Assuming $K_1'' = K_2'' = K^*$, the torque can then be varied according to

$$T = K^* (\phi - 2\phi_o) \tag{45}$$

Certain applications impose severe constraints on the long-term attitude stability. For instance, a Very-High-Resolution Radiometer (VHRR) in geosynchronous orbit demands long-term stability of the order of 0.004 deg for 30 min for accurate prediction of the wind velocity. Such a requirement cannot be met by a normal momentum biased spacecraft. The proposed magnetic controller reduces the drift to a large extent. This can be practically reduced to zero for a unidirectional disturbance. Equation (40) gives the drift rate for a cyclic disturbance and the gain K_1 can be suitably selected to achieve the required drift rate.

Backup Action when Geomagnetic Field Reverses

The magnetic control system loses its stability in the event of magnetic field reversal due to solar flares. Such a phenomenon, as already mentioned, occurs at the most once a year. The control system can be put out of action by ground commands after observing the roll behavior. During this period, the hydrazine gas jets may be used. The fuel consumption for this backup action is negligible.

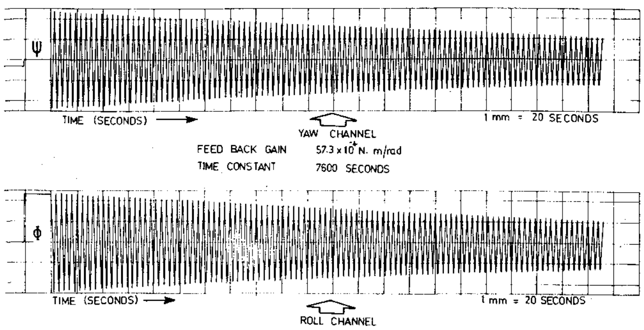

Fig. 4 Nutation damping by the controller.

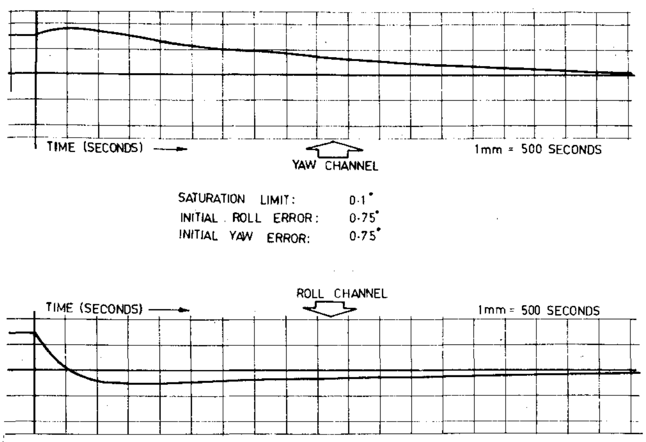

Fig. 5 Roll and yaw response including saturation.

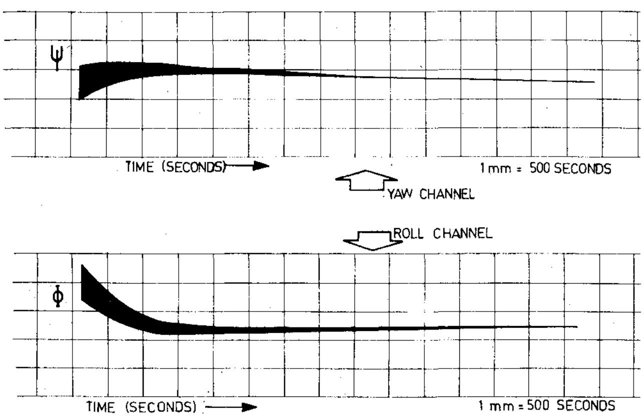

Fig. 6 Error reduction and nutation damping.

Simulation Results

The proposed control scheme was simulated for a geosynchronous communication satellite, using an EAI 680 analog computer. The linearized system equations were used. The relevant satellite and controller parameters are listed below:

Moments of inertia: $I_x = 150$ kg-m^2

$I_y = 140$ kg-m^2

$I_z = 80$ kg-m^2

Nominal angular momentum of the wheel $h_w = 22$ N-m-s
Feedback gains $K_1 = K_2 = 3.819 \times 10^{-5}$
Filter time constant = 100 s.
$\phi_m = 0.1$ deg.
$r_m = 0.1$ deg

The different cases studied are listed below:

1) Nutation damping by the proposed controller (Fig. 4)
2) Error reduction by the proposed controller, with roll and yaw error including saturation characteristics (Fig. 5)
3) Error reduction and nutation damping (Fig. 6)

The time constants obtained from simulation plots match well with the analytical expressions derived in the earlier section.

Conclusions

A new magnetic attitude control concept is proposed for momentum-biased satellites in near equatorial orbits. Such a control concept is highly desirable for long-life communication satellites. The closed-loop control law provides error control as well as nutation damping. The linearized equations of motion are analyzed to check for stability and to derive analytical expressions for system time constants and roll response. The proposed scheme was simulated using an EAI 680 analog computer for a typical communication satellite. The simulation results match well with the analytical results.

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