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Research Article

3-AXES ATTITUDE STABILIZATION OF SATELLITE IN THE SPIN UP MOMENTUM WHEEL MODE USING PD CONTROLLER

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ABSTRACT

An important task in the control of a satellite is orienting its antennas in such a way as to point in a certain direction with respect to the earth. In this paper, An Attitude Determination Control System (ADCS) is designed so that the satellite antenna will always point to the earth center. For this purpose, we used a Momentum Wheel (MW) actuator and magnetic torques to produce the required angular momentum.

In this paper, only the spin up of the momentum wheel mode is defined. In the MW spin up mode, the speed of the MW is kept high in order to set the changes in the three angles and their time rates to minimum values, resulting in a stable pitching motion for the satellite.

Key words: *Satellite, Attitude Determination and Control System, Momentum Wheel, PD Controller*

INTRODUCTION

Determination and attitude control system (ADCS) of the major subsystems of the satellite control subsystem, which covers a lot of the concepts. Satellite antennas have a limited ability to discuss the work efficiency of an antenna according to the power constraints must be taken into consideration. Maintain satellite attitude is essential because it will increase the lifetime of the satellite (Boussalis *et al.*, 1992).

The first step is to establish the precise orbit satellites that monitor the operation of the circuit is done with the control subsystems.

After the predetermined orbit satellites, in order to achieve the planned satellite maneuvers in the first step you should take steps to stabilize the situation stabilized the call. Categories and stabilization control circuit at any time because of the position and orientation (position) and the distortion is affected by environmental disturbances and cross-sectional areas are not seasonal.

Stabilization and its Variants

Satellite attitude control system for carrying out the missions entrusted the task of directing and keep it charged. Keeping and maintaining the stability of the satellite attitude control system is defined. In this regard, there are various ways to sustain the stabilization of rotating, three-axis stabilization, stabilization of the gravitational gradient methods and a combination of these three methods.

Roll Stabilization

In this way, the satellite rotates around the body coordinate system. This makes satellite rotation axis of the rotating system resistance against changes in the stiffness properties of the gyroscope spin axis which is said to be stable. The only way to control the rate of rotation of the rotary axis is taken into account.

3-Axis Stabilization

In this method, for each of the three axes of the coordinate system of the body, for optimal control components defined by the directions that come and try to be consistent. The main body of the satellite is no rotation. This control method is very complex and the method of stabilizing a rotating actuators and sensors with high accuracy and high capacities are needed. Attainable accuracy of the method of stabilizing a better drive, but it is more expensive.

Aiming accuracy can usually be less than 0.01 ° is the stabilization of the two main methods using 3-axis stabilization bias angular momentum (momentum wheels) and 3-axis stabilization with reactional wheel is divided. Due to its simplicity, low cost and good accuracy obtainable of the first 3-axis method rather than the second 3-axes method, satellite stabilization bias is selected and designed using bias angular momentum.

Stability of the Gravitational Gradient

The stability of LEO satellites are used in this method is not accurate as other methods, this method of controlling the boom in the use of satellites. Canvas control simple style and a long shaft that mass is concentrated at the end of it (Boussalis *et al.*, 1992).

Stabilization by Integration Methods

To take advantage of the properties of each method, we have integrated different techniques to be used. To achieve better accuracy and simplicity and lower power consumption and gradient methods, such as rotating and gravitational stabilization or the methods of magnetic stabilization and 3-axes are combined. The integrated approach is a compromise between the accuracy of simplicity, power consumption and cost are also being selected for the stabilization of the satellite orbit and mission.

MATERIALS AND METHODS

Control Rules

Most satellites that use magnetic torque makers, they use on / off control rule. Using PI control law using this operator provided (Chrone and Daugherty, 2004). Important thing is that the traction control on the satellite is deployed without restriction operator.

The implementation of adaptive control of control torque is desired by operators. If desired torque control is implemented by operators, optimal performance is achieved. At the beginning of the project, including dynamic and kinematic equations of satellite motion using Euler angles, are extracted, then the torque generated by the three magnetic torque makers and a momentum wheel for satellite under development are described.

Linear Dynamic Equations of Motion

If the agreement is designed to work with the principal axes of inertia of the products, can be removed from the dynamic equations. And they significantly simplify. In addition, the angular momentum can be approximated by an infinitesimal of angular momentum, which means the Euler angles and related derivatives is small. With these assumptions, the equations can Laplace transform and benefit from the advantages of linear control theory.

The Euler angles are small, the following equation holds (Hall, 2003):

$$\begin{bmatrix} \omega_{RIBx} \\ \omega_{RIBy} \\ \omega_{RIBz} \end{bmatrix} = \begin{bmatrix} 1 & \psi & -\theta \\ -\psi & 1 & \phi \\ \theta & -\phi & 1 \end{bmatrix} \begin{bmatrix} . \\ -\omega_o \\ . \end{bmatrix} = \begin{bmatrix} -\psi\omega_o \\ \omega_o \\ \phi\omega_o \end{bmatrix} \quad (1)$$

By the unknown $\overline{\omega}$ and $\overline{\omega} = \overline{\omega}_{BI} = \overline{\omega}_{BR} + \overline{\omega}_{RIB}$ is:

$$\begin{bmatrix} \omega_x \\ \omega_y \\ \omega_z \end{bmatrix} = \begin{bmatrix} p \\ q \\ r \end{bmatrix} + \begin{bmatrix} -\psi\omega_o \\ -\omega_o \\ \phi\omega_o \end{bmatrix} \quad (2)$$

The Euler angles are small $p \cong \dot{\phi}$ و $q \cong \dot{\theta}$ و $r \cong \dot{\psi}$ approximated. The approximate equation (2) becomes:

$$\begin{bmatrix} \omega_x \\ \omega_y \\ \omega_z \end{bmatrix} = \begin{bmatrix} \dot{\phi} - \psi\omega_o \\ \dot{\theta} - \omega_o \\ \dot{\psi} + \phi\omega_o \end{bmatrix} \quad (3)$$

Equation (3) we simply:

$$\dot{\omega}_x = \ddot{\phi} - \omega_o \dot{\psi}$$
$$\dot{\omega}_y = \ddot{\theta}$$
$$\dot{\omega}_z = \ddot{\psi} + \omega_o \dot{\phi} \qquad (4)$$

In this project we will deal with the satellite angular momentum bias. The satellite is a fixed initial bias is applied along the axis of inertia about the axis of the spacecraft angular stability is achieved. With this assumption, equation (1) and equation (2) and equation (3) into the linearized equations of motion become desirable:

$$\begin{cases} \overline{T}_{cx} + \overline{T}_{dx} = \omega_o h_w + h_w \dot{\psi} + \omega_o h_w \phi + I_{xx}\ddot{\phi} + 4\omega_o^2\left(I_{yy} - I_{zz}\right)\phi - \omega_o\left(I_{xx} - I_{yy} + I_{zz}\right)\dot{\psi} \\ \overline{T}_{cy} + \overline{T}_{dy} = -\dot{h}_w \qquad\qquad\qquad\qquad + I_{yy}\ddot{\theta} + 3\omega_o^2\left(I_{xx} - I_{zz}\right)\theta \\ T_{cz} + \overline{T}_{dz} = \omega_o h_w - h_w \dot{\phi} + \omega_o h_w \psi + I_{zz}\ddot{\psi} + \omega_o^2\left(I_{yy} - I_{xx}\right)\psi - \omega_o\left(I_{xx} - I_{yy} + I_{zz}\right)\dot{\phi} \end{cases}$$
$$\lfloor \textit{biased momentum wheel terms} \rfloor \qquad (5)$$

Equation (5) h_{wz}, h_{wy}, h_{wx} components of angular momentum wheels that the axis of rotation of the Z_B, Y_B, X_B satellite body coordinate system they are: $h_{wx} = I_{wx}\,\omega_{wx}$, $h_{wy} = I_{wy}.\omega_{wy} + h_{wyo}$, and $h_{wz} = I_{wz}.\omega_{wz}$. Where I_{wy}, I_{wx} و I_{wz} the moments of inertia and ω_{wy}, ω_{wx} و ω_{wz} the angular velocity of the wheels and the wheels are. The expressions $\dot{h}_{wz}, \dot{h}_{wy}, \dot{h}_{wz}$ are the angular momentum that wheel axes of the spacecraft body exercise. If $\dot{\omega}_{wx}$ the angular acceleration of the wheel axes X_B , then $\dot{h}_{wx} = I_{wx}\,\dot{\omega}_{wx}$ the negative angular momentum of the wheel X_B around its X_B axis, the satellite would enter. There are similar expressions for the components of the wheel Z_B, Y_B .

The Structures of the Reactional Wheels and Momentum Wheels (Goel and Kudva, 2004)

Block diagram of the reaction wheel momentum wheel is as follows:

- I_s =moment of inertia axes of the satellite
- I_w = moment of inertia of the motor axis
- K_m = constant coefficient $(T_m = K_m I_m)$
- ω_s = angular velocity of the wheel
- B = damping coefficient of viscous friction

In Figure 1, Block diagram of the reaction wheels and momentum wheels shown:

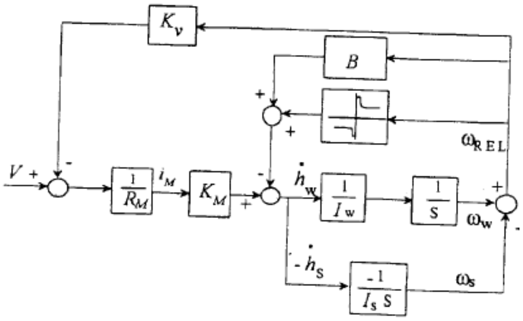

Figure 1: Block diagram of the reaction wheel momentum wheel

The reverse magnetic force on the rotor induction coils are directly linked to the difference between ω_w and $(\omega_{rel} = \omega_w - \omega_s)\omega_x$ speed. Because the speed of ω_s , the magnetic field coils cut altogether.
On the other hand, the total moment of the satellite must be zero to maintain balance. Regardless of friction are:

$$\frac{\dot{h}_w}{V} = \frac{I_w \dot{\omega}_\omega}{V} = \frac{S(\frac{K_m}{R_m})}{S + (\frac{1}{I_w} + \frac{1}{I_s})(\frac{K_\upsilon K_m}{R_m} + B)} \qquad (6)$$

In this case, according to the applied voltage dc motor armature constant stimulation, torque is produced opposite torque applied to the satellite. In equation (6) friction is removed and if it has to do with the fact that the nonlinear friction, for the analysis of nonlinear systems can be used. Also by linearization around an operating point, the state equations and linear control of the system is also used for the analysis. In equation (6) can be approximated by considering several factors to consider building momentum wheel. First, there ω_{rel} should be constant and the equilibrium condition and will therefore be considered linear friction:

$$\dot{h}_x = B\omega_{rel} \qquad (7)$$

Therefore, using the mass and inertia approximation, we have:

$$\frac{\dot{h}_w}{V} = \frac{I_w \dot{\omega}_w}{V} = \frac{S\frac{k_m}{R_m}}{S + \frac{k_v k_m}{I_w R_m}} \qquad (8)$$

To control the reaction wheel momentum control signal applied according to Figure 2.

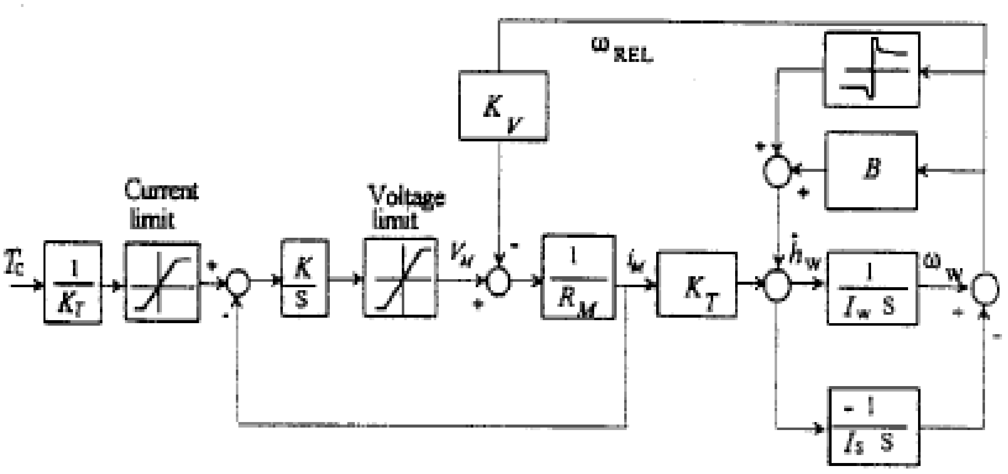

Figure 2: Block diagram of the control signal applied to the reaction wheel momentum [Goel and Kudva, 2004]

To find the transfer function of the reaction wheels or momentum wheels, Figure 3 and the fact that the value of K is chosen that $K >> \frac{K_o K_w}{I_w}$ so that in this case:

$$\frac{\dot{h}_w}{V} = \frac{1}{1+S\left(\frac{R_m}{K}\right)} \tag{9}$$

$\frac{R_m}{K}$ Represents the time constant is about ms (milliseconds) and can be used for practical purposes, it is ignored. Torque momentum equation reaction wheel momentum can't provide maximum torque required at each angular velocity and therefore limits dynamic is entered in torque- momentum curve.

Rate Momentum with Magnetic Torque Makers

Momentum wheel angular momentum vector in the direction of the axis of the screw and the screw is driven in the opposite direction. The components of the angular momentum of the wheels on the spacecraft fixed coordinate system are as follows (B.wie *et al.*, 2001).

$h_{wz} = o \qquad h_{wy} = -h_w \qquad h_{wx} = o$

By entering these values in equation (5) can be written:

$$\begin{Bmatrix} T_{cx} + T_{dx} \\ T_{cy} + T_{dy} \\ T_{cz} + T_{dz} \end{Bmatrix} = \begin{Bmatrix} I_{xx}\ddot{\phi} + \left[4\omega_o^2(I_{yy} - I_{zz}) + \omega_o h\right]\phi + \left[h - \omega_o(I_{xx} - I_{yy} + I_{zz}\right]\dot{\psi} \\ I_{yy}\ddot{\theta} + 3\omega_o^2\theta(I_{xx} - I_{zz}) - \dot{h} \\ I_{zz}\ddot{\psi} + \left[\omega_o^2(I_{yy} - I_{xx}) + \omega_o h\right]\psi - \left[h - \omega_o(I_{xx} + I_{zz} - I_{yy}\right]\dot{\phi} \end{Bmatrix} \tag{10}$$

The factors T_{cz}, T_{cy}, T_{cx} of control torque on the satellite configuration and T_{dz}, T_{dy}, T_{dx} torque components are a disturbance in the body coordinate system. Assuming that

$$h \,\square\, \max\left\{4\omega_o(I_{yy} - I_{zz}), \omega_o(I_{xx} - I_{yy} + I_{zz}), \omega_o^2(I_{yy} - I_{xx}), \omega_o(I_{xx} + I_{zz} - I_{yy})\right\} \tag{11}$$

$$\begin{Bmatrix} T_{cx} + T_{dx} \\ T_{cy} + T_{dy} \\ T_{cz} + T_{dz} \end{Bmatrix} = \begin{Bmatrix} I_{xx}\ddot{\phi} + \omega_o h\phi + h\dot{\psi} \\ I_{yy}\ddot{\theta} + 3\omega_o^2\theta(I_{xx} - I_{zz}) - \dot{h} \\ I_{zz}\ddot{\psi} + \omega_o h\psi - h\dot{\phi} \end{Bmatrix} \tag{12}$$

Equation (12) clearly be seen that for small errors in the equations of equations screw axis or off the roll axis. Yaw equations and roll axes by the angular momentum h are together. For this reason, we can separate the axle bolt, and discuss the role and Yaw (B.wie *et al.*, 2001).

Pitch Axis Control (Olivier, 2001)

From equation (12), the pitch axis linear equation is as follows:

$$T_{cy} + T_{dy} = I_{yy}\ddot{\theta} + 3\omega_o^2(I_{xx} - I_{zz})\theta - \dot{h} \tag{13}$$

When $I_{xx} = I_{zz}$ the equation (13) is reduced as follows:

$$T_{cy} + T_{dy} = I_{yy}\ddot{\theta} - \dot{h} \tag{14}$$

By substituting equation $T_{cy} = o$ in equation (14), the pitch equation becomes:

$$T_{dy} = I_{yy}\ddot{\theta} + k_\theta \tau_\theta \theta + k_\theta \theta \tag{15}$$

For $\tau_\theta > o$ و k_θ the equation of motion is damping coefficient by grade 2 and natural frequency ω_θ and of damping ξ_θ, obtained by the following equation:

$$\omega_\theta = \sqrt{\frac{k_\theta}{I_{yy}}}, \xi_\theta = \frac{\tau_\theta}{2}\sqrt{\frac{k_\theta}{I_{yy}}} \tag{16}$$

Momentum wheel because it is considered that the torque on the spacecraft, remove the external torques. Function control bolts are as follows:

$$G(s)H(s)=\frac{k_\theta(\tau_\theta s+1)}{I_{yy}s^2} \tag{17}$$

The open loop transfer function G (s) H (s), two poles at s = 0 is a zero.

And closed-loop transfer function is as follows:

$$\frac{\theta(s)}{T_{dy}}=\frac{1}{I_{yy}(s+\sqrt{k_\theta / I_{yy}})^2} \tag{18}$$

Error rate of change is determined by the sensor changes. The sensor rate of change can be very low. Therefore, practical and useful method is using a PD controller.

RESULTS AND DISCUSSION

Design and Simulation of PD Controller Operating Mode Gain Momentum Wheel Speed

Satellite angular momentum applied to the pitch axis with increasing momentum wheel speed is achieved and maintain this axis. The operational in the "spin up momentum wheel" mode is defined. DC motors are momentum wheel.

This type of wheel is controlled by the voltage applied. In the remainder of this section, the design of a PD controller for constant voltage to increase the momentum wheel speed and discussed. By applying this controller on a wheel momentum and simulation of the conceptual design shown and some physical limitations determined. Before the simulation of the mode of operation of uncontrolled due to the dynamic satellite in Figures 5, 4 and 3 are shown:

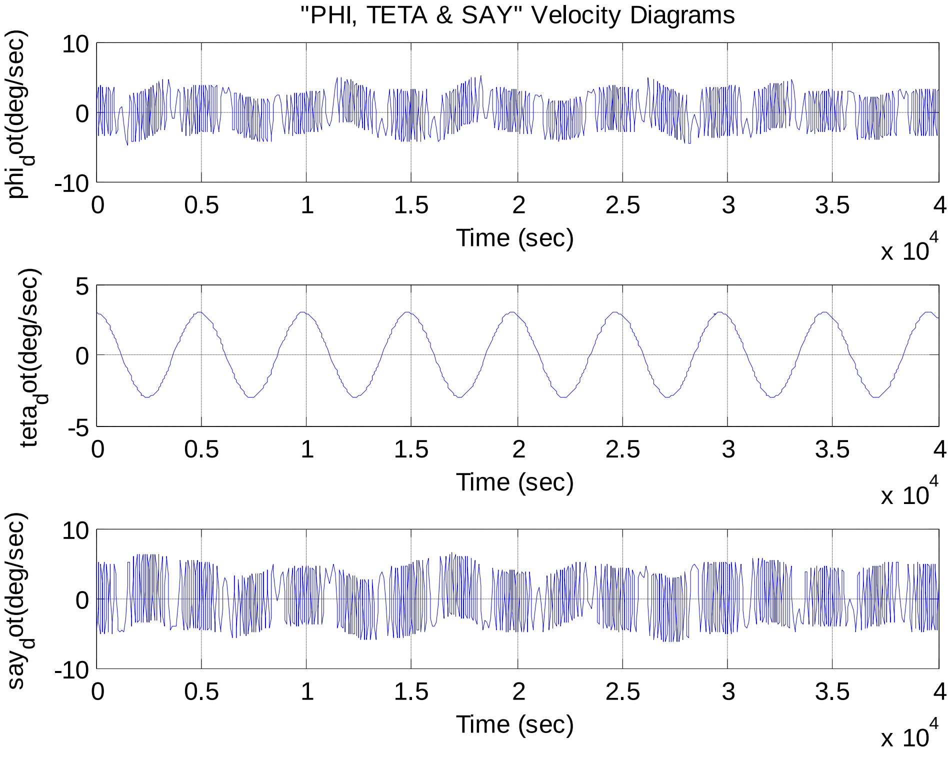

Figure 3: The curve angles ψ, θ, ϕ without using a PD controller

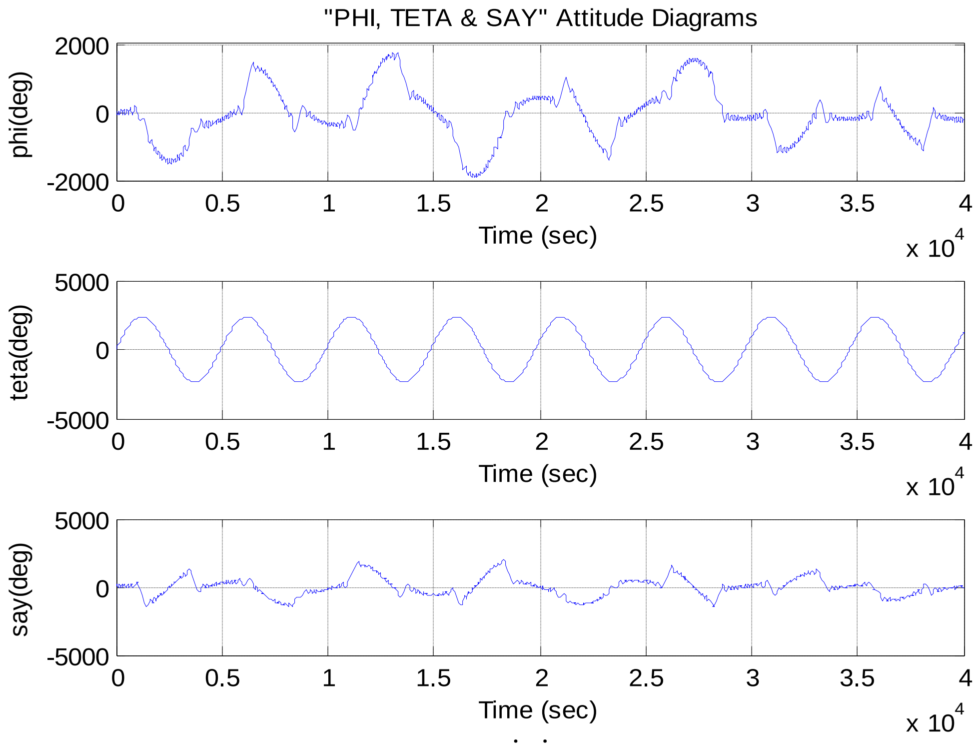

Figure 4: Changes in the rate curve $\dot{\psi},\dot{\theta},\dot{\phi}$ without using the PD controller

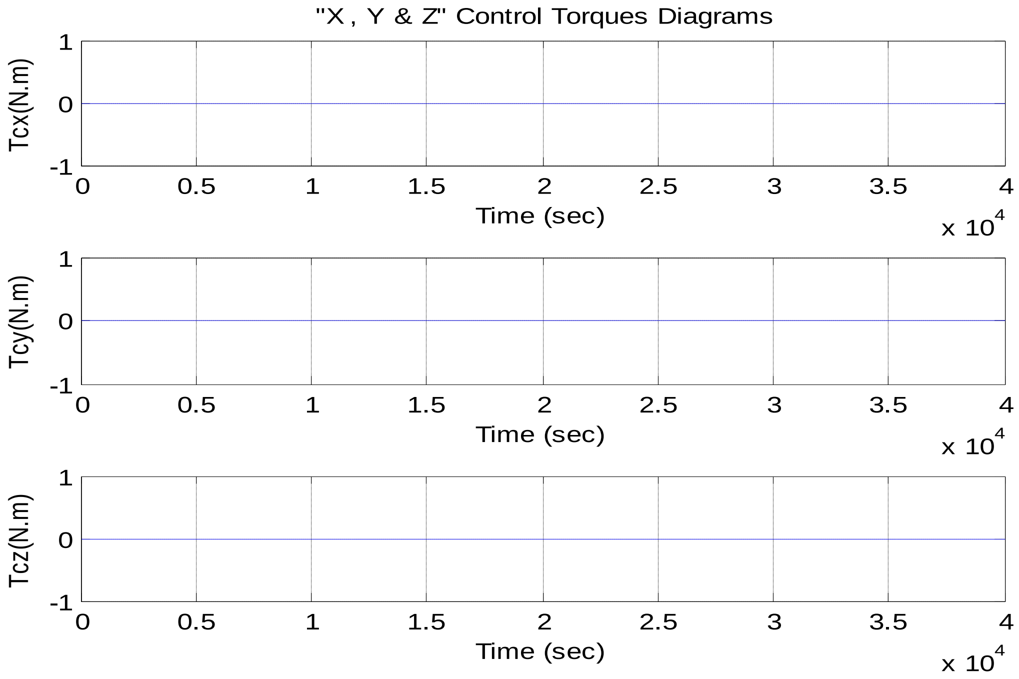

Figure 5: The control torque generated without using PD controller

Because there is not yet any control torque control curves along the three axes are zero.
The block diagram for the satellite attitude control system under design is shown in Figure 6:

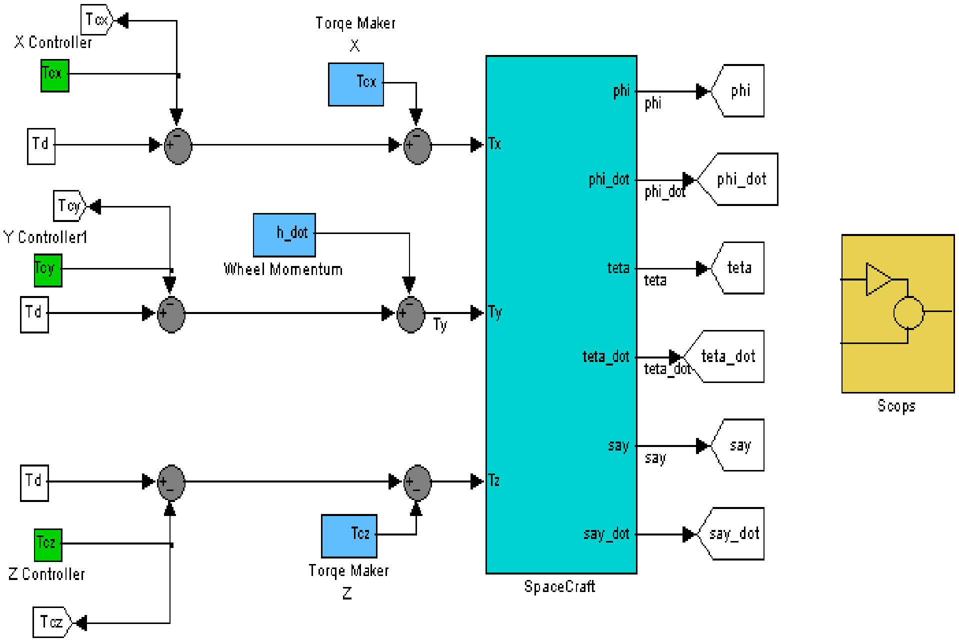

Figure 6: for the satellite attitude control system under design

PD Controller is Designed for the Spin up Momentum Wheel Mode

After the satellite in orbit after launch movement in two axes X_B و Z_B and angles ϕ و ψ and damping to zero, it must be stable satellite axis Y_B.

With increasing speed, momentum wheel, gyroscopic stiffness increases, resulting in increased stability. Because of the disturbance torque, wheel speed varies. On the other hand the wheel speed from zero to the nominal value of the desired angular momentum has to be achieved. Therefore, by applying a voltage to the voltage control and wheel, the changes of the angular momentum wheel must be such that have the least effect on the angles θ,ϕ و ψ (Hall, 2003). Wheel speed 955 rpm $\left(\omega_w = 100 \, {}^{rad}\!/_{Sec}\right)$ to rotate about the axis Y_B. Since the moment of inertia of the wheel is $I_w = 0/05 kg.m^2$, the angular momentum of the satellite is calculated in direction Y_B the following formula:

$$H_{wy} = I_w \omega_w \tag{19}$$

Changes in the satellite angular momentum axis Y_B as follows:

$$\frac{\dot{H}_{wy}}{V} = \frac{1}{1 + s\left(\frac{R_m}{k}\right)} \tag{20}$$

PD Controller Simulation for the Spin up Momentum Wheel Mode

PD controller with the simulation model can be used to indicate the system's stability. In this mode the controller design and simulation of the momentum wheel speed is so high that changes θ,ϕ و ψ and the rate of change $\dot{\theta},\dot{\phi}$ و $\dot{\psi}$ and the minimum is reached and the system is stable. All initial conditions in this mode to reduce fluctuations in fashion after throwing the same initial conditions are assumed. The mode control law is applied as follows:

$$T_{cy} = k_{\theta}(T_{\theta}.\dot{\theta}+\theta) \tag{21}$$

Also $T_{\theta} = 2\sqrt{\dfrac{I_{yy}}{k_p}}, k_{\theta} = 0.001$ are coefficients. The simulation results at the touch of a switch mode that was active in the 1500s, and start controls.

In Figures 7, 8 and 9, the results of the simulation are shown:

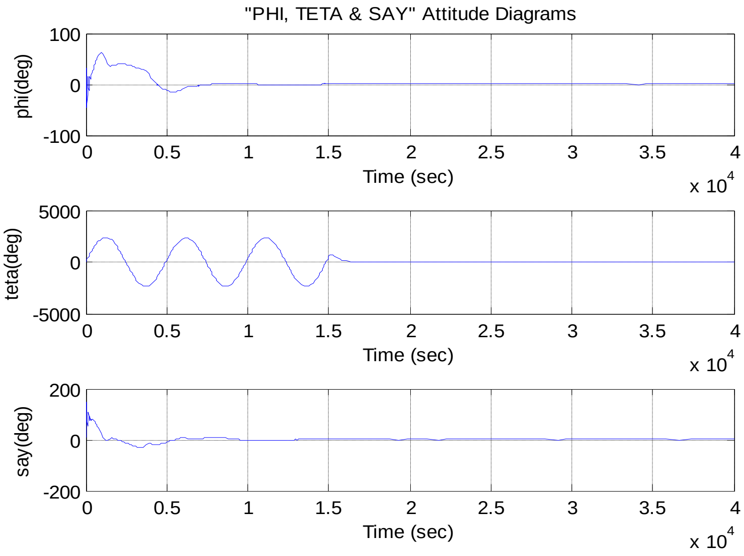

Figure 7: The angle of ψ,θ,ϕ the spin up momentum wheel mode

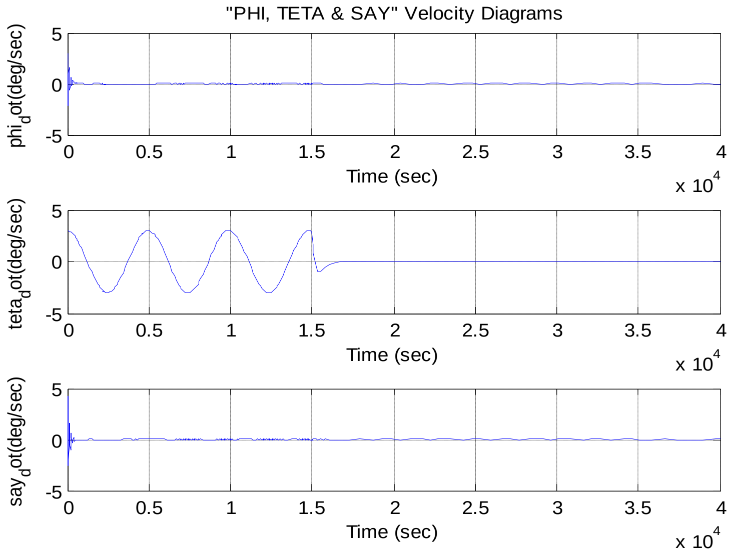

Figure 8: The curve of change $\dot{\psi},\dot{\theta},\dot{\phi}$ in spin up momentum wheel mode

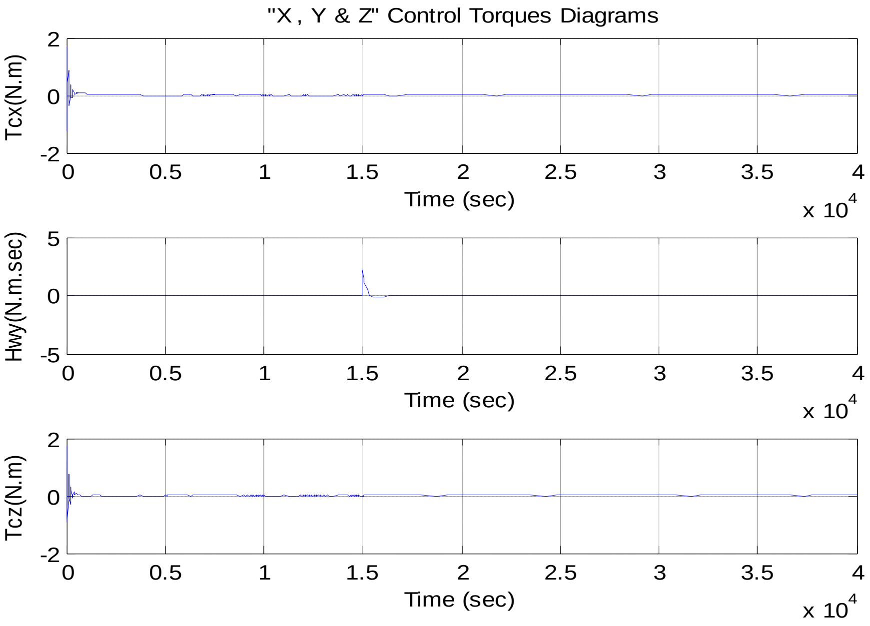

Figure 9: The generated control torque at spin up momentum wheel mode

As can be seen, there has not been a significant change in direction ψ,ϕ, but the angle θ is large variation in the normal mode angle is accurately controlled in the range that is beyond the scope of this article.

Conclusion

Satellite operators are used in the design of a momentum wheel and three magnetic torque makers. The torque produced by the Earth's magnetic field is perpendicular magnetic torque makers and there is no possibility of torque in any direction. The maximum angular momentum and magnetic moment torque makers Maximum wheel is 6N.ms. This was the good performance of the controller designed in simulation, it was shown (CD Hall., 2003).In this paper it is assumed that the dynamics can be assumed to be unchanged for a bunch of satellites and on the basis of the PD controller is designed. If the satellite dynamics of uncertainty, the coefficients of adaptive PD controller can be used. Also, since the satellite dynamics of multiple-input and multiple output, it is possible to control the PD controller to control multiple satellites used.

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