Attitude Acquisition System for Communication Spacecraft

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A new method for the attitude acquisition of a momentum-biased communication spacecraft that is spin stabilized in the transfer orbit is presented. The initial spin axis attitude is parameterized as a quaternion vector using the transfer orbit horizon and sun sensors. The acquisition control system employs direct rate and quaternion feedback to autonomously despin the spacecraft and align the momentum wheel axis along the orbit normal. The quaternion feedback control gains are selected to perform the maneuver along the Euler axis. The control torques are realized by thruster pulse width modulation. A simple thruster pulse width scaling is included to keep the maneuver axis along the Euler axis. A typical attitude acquisition including momentum wheel spin up to the nominal bias level is completed in 1800 s. The momentum wheel axis is brought within 1 deg of the orbit normal.

Nomenclature

D_i = ith axis rate damping gain
I = inertia matrix
K_i = ith axis quarternion feedback gain
T = externally applied torque vector
T_{ci} = torque commanded along the ith axis
ω = angular velocity vector
ω_T = body rate threshold

Introduction

GEOSYNCHRONOUS communication satellites are generally three-axis stabilized using a bias momentum wheel along the pitch axis. During the transfer orbit they are spin stabilized and the spin axis attitude is controlled to coincide with the apogee kick motor's (AKM) firing attitude for injection into the geostationary orbit. Two techniques, viz., dual-spin turn (DST) and classical sun-Earth acquisition are widely used to acquire the Earth after AKM firing. The DST[1] is a very simple and attractive technique but depends on the passive energy dissipation in the propellant tanks for damping the residual nutation. In fact, for certain spacecraft configuration, this passive damping alone might result in excessively large damping time constant and necessitate passive or active nutation damping resulting in increased weight. The classical sun-Earth acquisition system requires a sun sensor that is sparingly used in the nominal mission. Both of these methods require gas jets to orient the momentum vector from the AKM firing attitude to orbit normal and despin the spacecraft. The classical sun-Earth acquisition system also requires the spacecraft to be at a particular orbital position (6 a.m. or 6 p.m.) for Earth lock and thus impose further operational and time constraints.

The reorientation of the spacecraft from the AKM firing attitude to the nominal mission mode attitude involves large angle maneuvers. The use of quaternions for large angle maneuvers is very attractive since they do not have any geometrical singularity and are well suited for attitude propagation using strap-down attitude determination algorithm. The use of quaternions for feedback control was first suggested by Mortenson[2] in 1968. Reference 3 presents the stability and control analysis of three-axis large angle reorientation for perigee maneuvers. Hubert and Miller[4] employed a quaternion-based attitude determination system to simultaneously control precession error and damp nutation of a spin-stabilized shuttle compatible orbit transfer system for perigee maneuvers. Recently, Wie et al.[5] presented a quaternion feedback regulator for eigenaxis rotational maneuvers. An excellent overview of the papers on quaternion-based attitude determination and control published since Mortenson's work is given in Ref. 5.

This paper considers an innovative spinning initialization of inertial parameters (SIIP) technique to acquire the Earth. The SIIP uses the transfer orbit horizon and analog sun sensors to initialize the spacecraft attitude in inertial space. The spin axis attitude is parameterized in terms of quaternions.

The attitude acquisition control system uses direct rate and quaternion feedback to autonomously despin the spacecraft and bring the momentum wheel axis aligned along the orbit normal. The pitch wheel is then spun up to complete the Earth lock. The large angle maneuvers are realized using reaction jets. The quaternion feedback gains are scaled proportional to the principal moments of inertia to perform the maneuver along the Euler axis. Control torques proportional to the error are obtained by pulse width modulating the thruster firing time. A simple pulse width scaling is used to account for thruster saturation and keep the rotational axis aligned with the Euler axis. Simulation results for a typical communication spacecraft including various error sources such as gyro bias, misalignment error, gyro scale factor, and SIIP initialization errors showed that the momentum wheel axis is brought within 0.7 deg of the orbit normal in less than 1800 s. Simulation results with identical quaternion feedback gains (non-Euler axis maneuver) showed that an additional 20 s of thruster firing is required and the momentum wheel axis offset from the orbit normal is 2.2 deg. Thus, an optimal maneuver resulting in reduced propellant consumption and smaller terminal errors is achieved by scaling the gains proportional to the inertias.

Spinning Initialization of Inertial Parameters

Figure 1 shows the spacecraft fixed coordinate system and the transfer orbit attitude sensor geometry. The spacecraft spins about the X axis during the transfer orbit. The momentum wheel axis coincides with the Z axis. The horizon sensor assembly (HSA) gives the scanned Earth chordwidth. The sun sensor assembly (SSA) is a hybrid analog/digital device that gives a pulse and data outputs. The data output determines the sun angle, i.e., the angle between the spin axis and the sun line. The pulse output establishes the inertial reference for phasing

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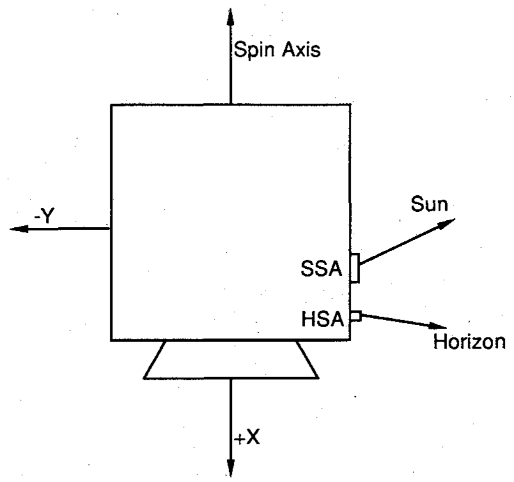


Fig. 1 Spacecraft coordinate system and sensor geometry.

the thruster firing during spin axis precession maneuvers. The spacecraft attitude is determined on the ground using the telemetered Earth chordwidth and sun angle data. The data from horizon and sun sensors are usually available for 3–4 h centered on apogee.

Figure 2 shows the block diagram of the attitude determination and control system using SIIP. The attitude acquisition system nulls the body rates and reorients the Z axis so that it is aligned within a tolerable limit along the orbit normal. The transfer orbit horizon and sun sensors are used to initialize the spacecraft attitude in inertial space as a four-element quaternion vector. The rate measuring assembly (RMA) consisting of three-axis rate integrating gyros are used to propagate the quaternions. The RMA gives the incremental angles accumulated over the quaternion update interval. The attitude control logic generates torque commands that are proportional to the error quaternions and body rates. The spacecraft carries the RMA for three-axis attitude control during stationkeeping maneuvers. Thus, the attitude acquisition system uses existing hardware and attitude determination methods.

The sequence of events beginning from AKM firing to the final attitude acquisition are summarized in the following.

1) Determine the reference attitude at the sun sensor reference pulse using standard techniques following the apogee burn.

2) Initialize the quaternion propagation at the sun sensor reference pulse.

3) Propagate the quaternions using the gyro-based quaternion update algorithm.

4) Autonomously despin the spacecraft and hold the body rates below a threshold.

5) Perform three-axis attitude control and align the momentum wheel axis along the orbit normal.

6) Spin up the momentum wheel to the bias momentum level maintaining zero body rates.

7) Switch to normal on-orbit mode.

Table 1 gives a typical event timeline.

The acquisition control system aligns the body axes closely with the orbit referenced yaw, roll, and pitch axes. Normally, when the acquisition is completed, the on-orbit Earth sensor will be viewing the Earth. The final pitch attitude error, in fact, can be large since the momentum wheel control logic will enable the X and Y axes to be aligned with the orbit yaw and roll axes.

Attitude Acquisition Control

Quaternion Kinematics

We will review briefly the spacecraft equations of motion and quaternion kinematics. The rotational equations of motion for a rigid spacecraft is given as

$$I\dot{\omega} + \tilde{\Omega} I \omega = T \tag{1}$$

where

$$\tilde{\Omega} = \begin{bmatrix} 0 & -\omega_3 & -\omega_2 \\ \omega_3 & 0 & -\omega_1 \\ -\omega_2 & -\omega_1 & 0 \end{bmatrix}$$

The gyroscopic coupling term $\tilde{\Omega} I \omega$ arises due to the cross products of inertia and unequal transverse moments of inertia.

The Euler's principal rotation theorem[6] states that a rigid body can be brought from an arbitrary initial attitude to an arbitrary final orientation by a single rotation of the body through a principal angle ϕ about a principal line 1, the principal line being a judicious axis fixed in the body and in space. The quaternions, also called Euler parameters, are expressed in terms of the principal angle and the principal line as follows:

$$q_0 = \cos(\phi/2)$$
$$q_1 = l_1 \sin(\phi/2)$$
$$q_2 = l_2 \sin(\phi/2)$$
$$q_3 = l_3 \sin(\phi/2)$$

l_1, l_2, and l_3 are the direction cosines of the principal line in the reference frame. We will call the principal line the Euler axis. Since the principal angle ϕ is always smaller than the algebraic sum of the three successive Euler angles, it is desirable to perform the large angle maneuver about the Euler Axis. Such a maneuver being an optimal (shortest) rotation results in reduced maneuver time and/or propellant consumption.

It is obvious that the quaternions satisfy the constraint equation

$$q_0^2 + q_1^2 + q_2^2 + q_3^2 = 1$$

The most impressive property of the quaternions is that the time rate of change of quaternions is a linear differential equa-

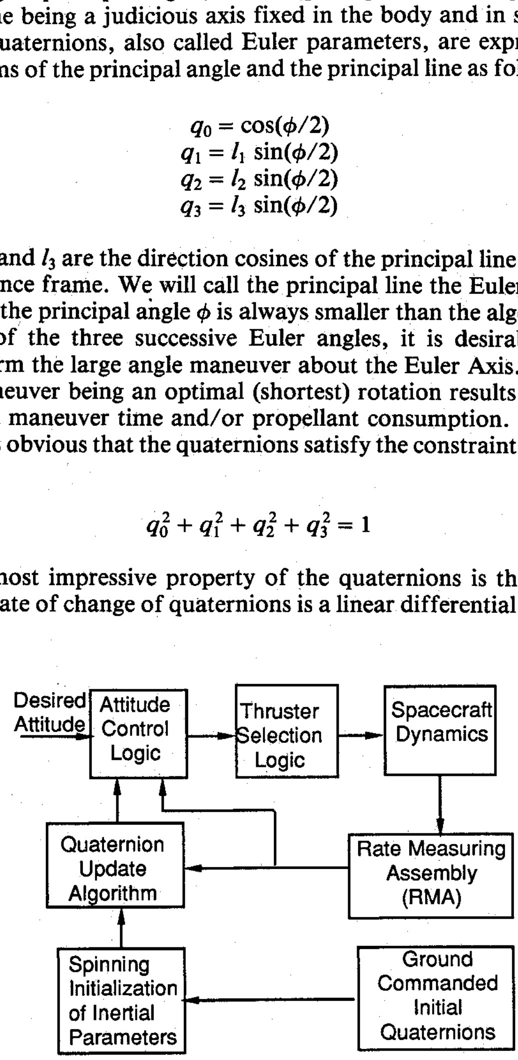


Fig. 2 Attitude determination and control using SIIP.

Table 1 Typical SIIP event timeline

Time after sun pulse, mm:ss	Event
– 40:00	Power up gyros
– 30:00	Command initial quaternion based on attitude determination
– 10:00	Gyro warm-up completed
0:00	Start propagating quaternions; start feedback control
1:30	Spacecraft despun and acquisition begins
4:00	Acquisition complete and wheel spin up commanded
30:00	SIPP maneuver completed

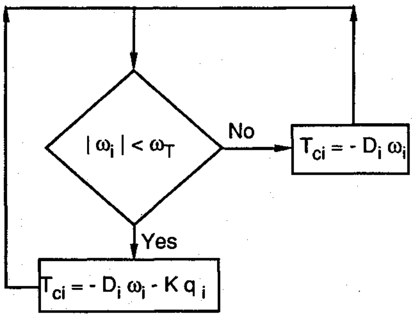
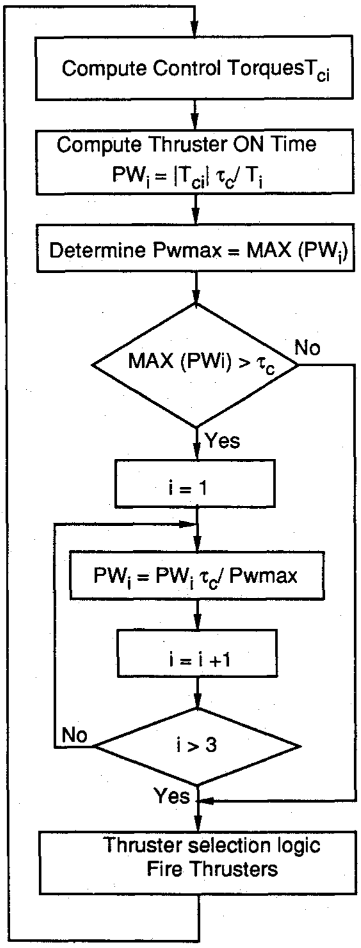


Fig. 3 Attitude control logic.



τ_c = control interval
T_i = Magnitude of thruster torque along i^{th} axis
PW_i = i th axis thruster ON time

Fig. 4 Thruster pulsewidth scaling logic.

tion involving the body rates. The quaternion kinematic equation is

$$\dot{q} = [\omega]q \tag{2}$$

where

$$[\omega] = \frac{1}{2}\begin{bmatrix} 0 & -\omega_1 & -\omega_2 & -\omega_3 \\ \omega_1 & 0 & \omega_3 & -\omega_2 \\ \omega_2 & -\omega_3 & 0 & \omega_1 \\ \omega_3 & \omega_2 & -\omega_1 & 0 \end{bmatrix}$$

Table 2 System parameters

$$\text{Spacecraft inertia, } I = \begin{bmatrix} 1458 & 0 & 0 \\ 0 & 1706 & -98 \\ 0 & -98 & 1824 \end{bmatrix} (\text{Kg-m}^2)$$

Wheel inertia = 0.175 Kg-m^2

Control Parameters	
Quaternion gains (case 1):	[40, 40, 40]
Quaternion gains (case 2):	[31.9, 37.4, 40.0]
Rate damping gains (N-m-s/rad):	[129.2, 151.1, 161.6]
Body rate threshold (deg/s):	1.7
Control interval (s):	0.2
Gyro sampling interval (s):	0.1
Initial spin rate (rpm):	5.0

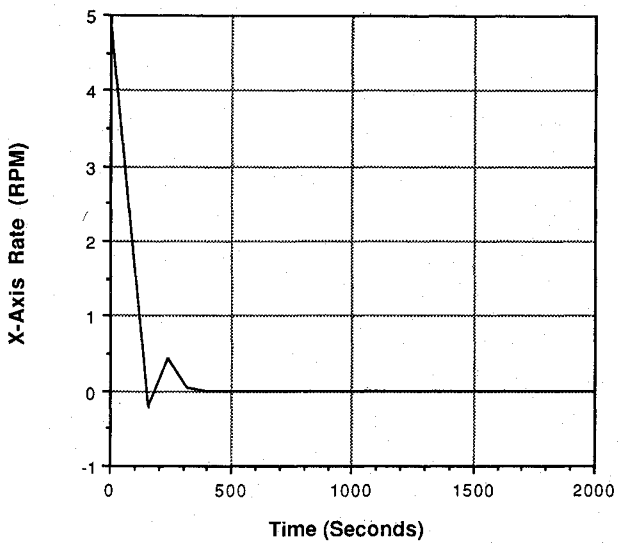


Fig. 5 Spin rate vs time (case 1).

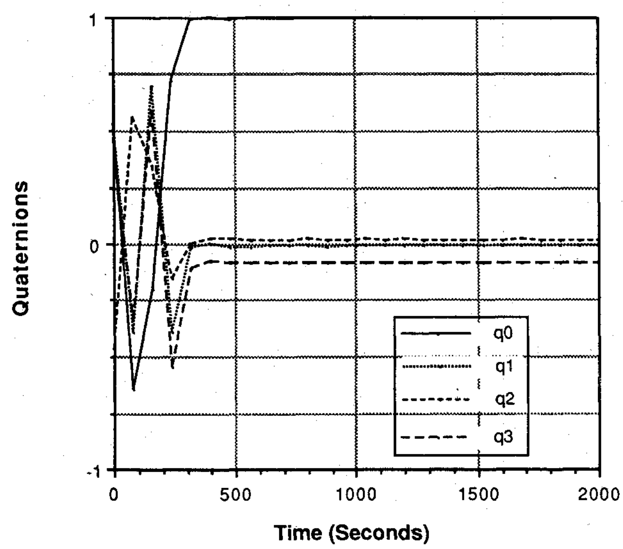


Fig. 6 Quaternions vs time (case 1).

Equation (2) is the starting point for the most quaternion-based attitude determination algorithm. Any one of the methods, such as the rotation vector concept, Runge-Kutta method, and Taylor series expansion, can be used for propagating the quaternions. Assuming the target frame to be the inertial frame, the error quaternion for feedback control can be propagated using the previous equations. The initial error quaternion is obtained from the ground-generated attitude data.

Table 3 Typical SIIP error sources

Error source	Value (units)
Ground-commanded SIIP initalization	1 (deg)
Gyro misalignment	0.3 (deg)
Gyro bias uncertainty	0.41 (deg/h)
Gyro scale factor	300 (ppm)

When the error quaternion reaches [1, 0, 0, 0], the spacecraft body axes coincide with the reference frame and the attitude acquisition is complete.

Acquisition Control Law

Figure 3 shows the attitude control logic diagram. The quaternion feedback control suggested by Mortensen[2] takes the following form:

$$T_{ci} = -K_i q_i - D_i \omega_i \qquad i = 1, 2, 3 \tag{3}$$

Mortensen[2] chose the control gains K_i to be inversely proportional to the principal moments of inertia. Wie and Barba[3] used identical gains for large angle perigee maneuvers. Since the quaternions q_1, q_2, and q_3 contain the direction cosines of the principal axis, the control torque vector lies along the Euler axis. Wie et al.[5] using Lyapunov stability analysis, showed that the identical gain control law results in a globally stable control system that is robust to inertia uncertainty. But the resulting rotation can take place about the Euler axis only if the principal moments of inertia are equal. For small body rates and negligible cross products of inertia, it is obvious from Eqs. (1) and (3) that the quaternion feedback gains must be scaled proportional to the spacecraft moments of inertia to perform the maneuver about the Euler axis. The gyroscopic coupling torques, however, act as disturbances and the feedback control might take place about an axis other than the principal axis.

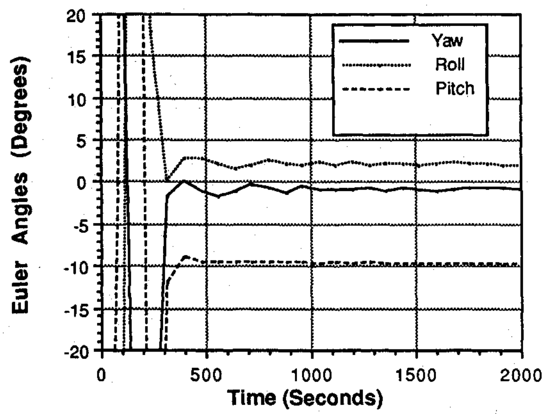


Fig. 7 Euler angles vs time (case 1).

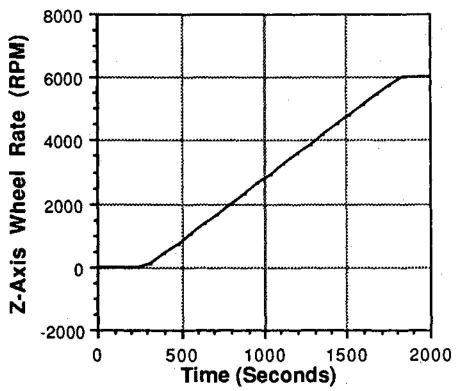


Fig. 8 Momentum wheel speed vs time (case 1).

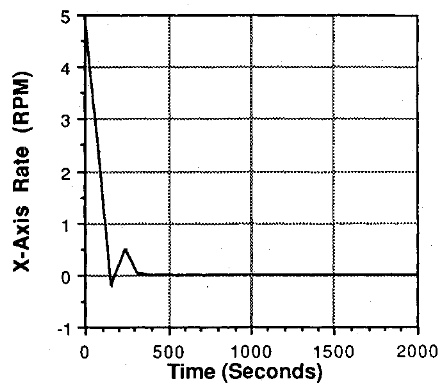


Fig. 9 Spin rate vs time (case 2).

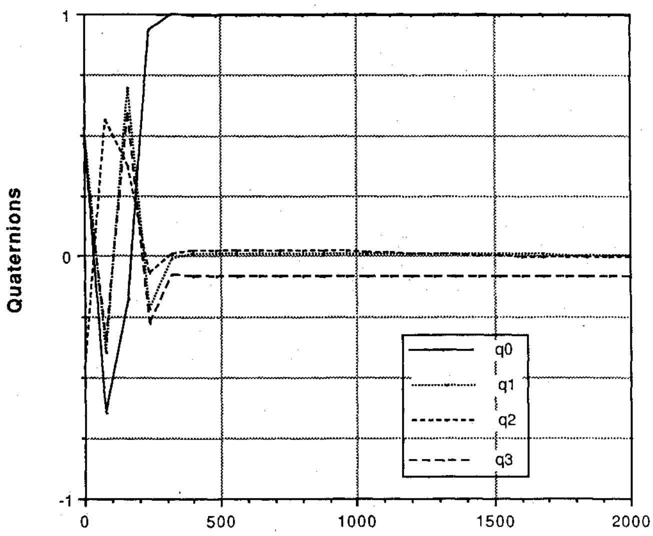


Fig. 10 Quaternions vs time (case 2).

Wie et al.[5] presented a feedback control law including gyroscopic decoupling torque and quaternion feedback gains scaled proportional to the inertia matrix to perform Euler axis maneuvers. Using Lyapunov stability analysis, Wie et al.[5] showed the control law to be globally stable.

Such gyroscopic decoupling is unnecessary if the body rates are kept very small during the attitude maneuvers. In fact, it is desirable to keep the body rates low so that the solar arrays and other flexible structures can be deployed with minimal attitude perturbations. Hence, the control law given by Eq. (3) is modified so that initially it reduces the body rates to lie below a threshold. The control logic checks whether the body rates exceed a threshold and commands the torques as per

$$T_{ci} = -D_i \omega_i \qquad \text{for} \qquad |\omega_i| < \omega_T \qquad i = 1, 2, 3 \tag{4}$$

Once the body rates are reduced below the threshold, the acquisition control law is the same as given by Eq. (3). The control gains K_i are scaled as

$$K_i = kI_i \qquad i = 1, 2, 3 \tag{5}$$

where k is a constant and I_1, I_2, and I_3 are the principal moments of inertia along X, Y and Z axes.

The large angle maneuvers are usually realized using reaction jets or reaction wheels. We will restrict our dicussion to control using reaction jets. The proportional torques required by the control system are realized by pulse width modulating the thruster on time. Because of the finite thruster torques, the pulse width computed over a control cycle can exceed the control interval. Thus, merely scaling the quaternion feedback

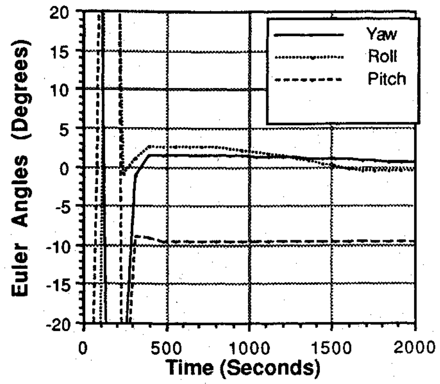


Fig. 11 Euler angles vs time (case 2).

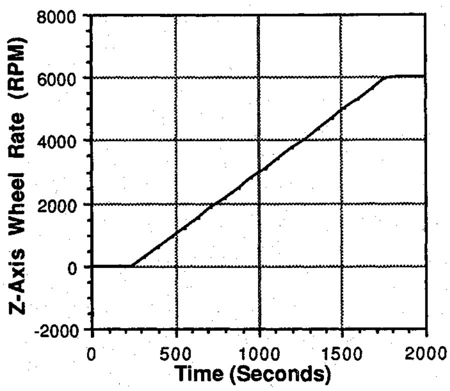


Fig. 12 Momentum wheel speed vs time (case 2).

gains will not assure Euler axis rotation. A further scaling is necessary to account for actuator (pulse width) saturation. Figure 4 presents a pulse width scaling logic that will keep the maneuver axis closely aligned with the Euler axis. Every control interval, thruster on time for each axis is determined. If the maximum of the three thruster on times exceeds the control interval, a relative scaling of the thruster on times is performed. Otherwise, the scaling is not applied.

Simulation Results

This section presents the simulation results for a typical communication spacecraft. Table 2 gives the mass properties and control parameters. The spacecraft is initially spinning at 5 rpm. The body rate threshold is selected as 1.7 deg/s. The thrusters produce 5 N-m torque about each axis. The minimum thruster pulse width is 20 ms. The control interval is 5 Hz, whereas the quaternion propagation is done at 10 Hz.

Table 3 lists the various error sources included in the simulation. The gyro errors include 0.41 deg/h bias uncertainty, 300-ppm scale factor error, and 0.3 deg misalignment error. Simulations showed the gyro scale factor error to be the dominating error source. The ground generated initial attitude determination error is 1 deg.

We consider two cases to obtain a comparison of Euler and non-Euler axis maneuvers. The first case used identical quaternion feedback gains for all the axes. The second case uses control gains scaled proportional to the principal moments of inertia. In both of the cases, thruster pulse width scaling shown in Fig. 4 is applied.

The target frame is selected to coincide with the orbit yaw, roll, and pitch axes at the expected maneuver completion time. Exact knowledge of maneuver completion time is not required. Uncertainty in maneuver completion time shows up as a pitch error that will be corrected by the Earth lock control system. We are primarily interested in closely aligning the spacecraft momentum vector with the orbit normal. When the error quaternion $q_0 \geq 0.999$, the momentum wheel is commanded to spin up to its nominal bias level. Alternatively, the wheel spin up can be commanded when the error quaternions q_1, q_2, and q_3 simultaneously lie below a threshold.

Figures 5–8 show the results for case 1. Figure 4 shows the angular rate along the X axis. The 5-rpm initial spin rate is reduced below the threshold in 100 s. The time history of the truth quaternions is presented in Fig. 6. The error quaternion q_0 reaches 1, whereas the other quaternions become zero. Figure 7 shows the Euler angles that define the relative orientation between the target frame and the body frame. Figure 8 shows the wheel spin up. The momentum wheel spin up begins at 250 s and is completed at 1800 s. The attitude errors at 2000 s are -0.9 deg in yaw, 2.0 deg in roll, and -9.7 deg in pitch. The momentum wheel offset from the orbit normal is 2.2 deg. These terminal errors will enable the on-orbit Earth sensor to view the Earth. The attitude errors remain nearly constant during and after the wheel spin up.

Figures 9–12 show the results for case 2. Figure 9 shows the angular rate along the X axis. The initial rate nulling performance is similar to case 1. Figure 10 presents the truth quaternions. It can be seen from Fig. 11 that the attitude errors are corrected during and after wheel spin up, unlike case 1 where the attitude errors remained constant. Figure 12 shows the wheel spin up. There is no significant difference in the maneuver completion time, but the attitude errors at 2000 s are 0.4 deg in yaw, -0.5 deg in roll, and -9.5 deg in pitch. The momentum wheel offset from the orbit normal is 0.7 deg.

A comparison of thruster firing times for the two cases showed that case 1 required an additional 20 s of thruster actuation. Thus, case 2 is an optimal maneuver resulting in smaller terminal errors and reduced fuel consumption.

Conclusions

A novel technique has been presented for the attitude acquisition of a communication spacecraft. The attitude initialization and propagation does not require any new hardware or attitude determination methods. Simulation results were presented for a spin-stabilized spacecraft in the transfer orbit. The technique, however, can be applied readily to other spacecraft configurations and large angle perigee and apogee maneuvers. The quaternion feedback gains are chosen to perform the maneuver along the Euler axis. The Euler axis maneuver results in reduced propellant consumption and smaller terminal errors compared to a non-Euler axis maneuver. Simulation results including various error sources showed that the wheel axis is brought within 0.7 deg of the orbit normal. The attitude acquisition including wheel spin up is completed in 1800 s.

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