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Donald Chu
Swales Aerospace

Sam Chen
NOAA NESDIS

Derrick Early
Swales Aerospace

Doug Freesland
ACS Engineering

Alexander Krimchansky
NASA GSFC

Bo Naasz
NASA GSFC

Alan Reth
Chesapeake Aerospace

Kumar Tadikonda
SAIC

John Tsui
NOAA NESDIS

Tim Walsh
NOAA NESDIS

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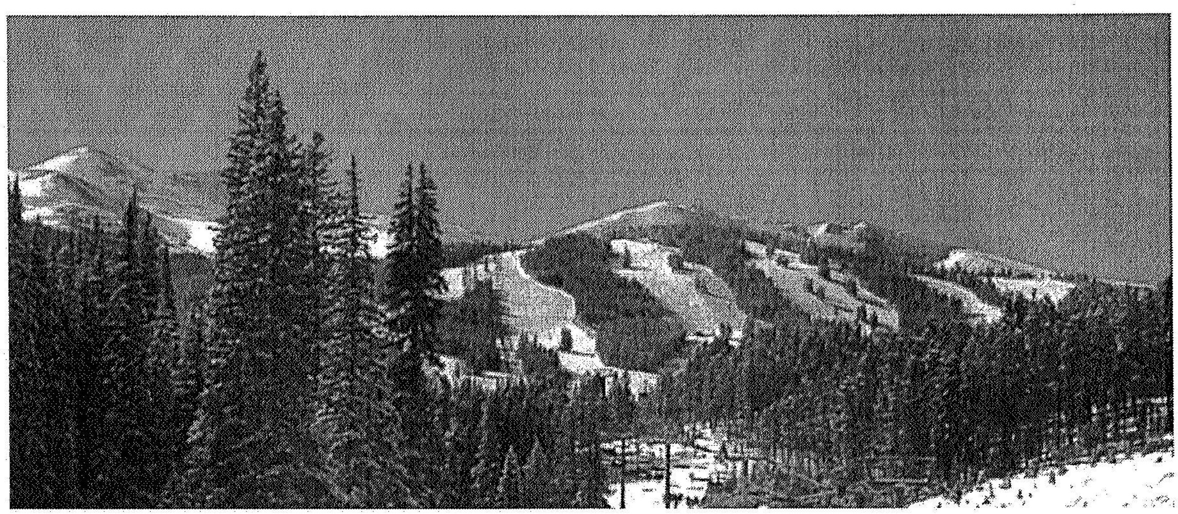

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GOES-R STATIONKEEPING AND MOMENTUM MANAGEMENT

Donald Chu[1], Sam Chen[2], Derrick Early[1], Douglas Freesland[3], Alexander Krimchansky[4], Bo Naasz[5], Alan Reth[6], Kumar Tadikonda[7], John Tsui[2], Tim Walsh[8]

The NOAA Geostationary Operational Environmental Satellites (GOES) fire thrusters to remain within a 1° longitude-latitude box and to dump accumulated angular momentum. In the past, maneuvers have disrupted GOES imaging due to attitude transients and the loss of orbit knowledge. If the R-series of spacecraft to be launched starting in 2012 were to follow current practice, maneuvers would still fail to meet Image Navigation and Registration (INR) specifications during and after thruster firings. Although maneuvers and recovery take only one percent of spacecraft lifetime, they sometimes come at inopportune times, such as hurricane season, when coverage is critical.

To alleviate this problem, thruster firings small enough not to affect imaging are being considered. Eliminating post-maneuver recovery periods increases availability and facilitates autonomous operation. Frequent maneuvers also reduce longitude/latitude variation and allow satellite co-location. Improved orbit observations come from a high-altitude GPS receiver, and improved attitude control comes from thruster torque compensation. This paper reviews the effects of thruster firings on position knowledge and pointing control and suggests that low-thrust burns plus GPS and feedforward control offer a less disruptive approach to GOES-R stationkeeping and momentum management.

INTRODUCTION

The GOES spacecraft fire thrusters to maintain station and dump angular momentum. On GOES N-P, daily momentum dumps and bi-monthly east-west (longitude) maneuvers will be done during a daily 10-minute housekeeping period. Large north-south (inclination) maneuvers are done once a year, and INR performance may take 6 hours to recover. By performing small maneuvers on a daily basis, one could certainly reduce box size but might also reduce availability because of the time to recover position knowledge and stabilize attitude. This paper attempts to assess the feasibility of daily maneuvers in the light of expected orbit estimation and attitude control performance. [8]

[1] Swales Aerospace, 5050 Powder Mill Road, Beltsville, MD 20705
[2] NOAA-NESDIS, Spacecraft Operations Control Center, Suitland, MD 20746
[3] ACS Engineering, 3535 Sheffield Manor Terrace Suite 403, Silver Spring, MD 20904
[4] NASA-GSFC, GOES-R Project (Code 417), Greenbelt, MD 20771
[5] NASA-GSFC, Flight Dynamics Analysis Branch (Code 595), Greenbelt, MD 20771
[6] Chesapeake Aerospace, PO Box 567, Grasonville, MD 21638
[7] SAIC, 1710 SAIC Drive, M/S 2-6-9, McLean, VA 22102
[8] NOAA-NESDIS, GOES-R Program Office, 1315 East-West Highway Silver Spring, MD 20910

BACKGROUND

GOES spacecraft are stationed at 75° and 135° west longitude. Without maneuvers, both would oscillate about the -105° longitude due to the Earth's nonspherical gravitational field. Their orbital inclinations would also increase by about 1° per year. To keep the spacecraft close to their nominal longitudes in the equatorial plane, thruster firings are necessary. Tangential thruster firings lower or raise the semi-major axis to set the spacecraft drifting east or west. Out-of-plane thruster firings cancel inclination drift caused by solar and lunar gravity. In addition, thrusters are needed to dump angular momentum due to solar radiation pressure.

As shown in Table 1, north-south maneuvers require about 50 m/s velocity change (Δv) each year and east-west maneuvers require about 1.3 m/s Δv each year. If maneuvers were done every day, the daily Δv's would be about 13 cm/s and 0.35 cm/s. Momentum dumps of about 25 Nms are also needed each day for spacecraft with unbalanced solar arrays. It is difficult for GOES to have balanced solar arrays because of the need for a clear field-of-view to space for the instrument cryogenic radiators. For thrusters with 1 m moment arms, the momentum dump Δv is about 1 cm/s. Daily north-south burns are largest followed by momentum dumps and finally east-west burns.

Table 1 GOES Maneuver Schedule

	thruster axis	magnitude Δv (m/s)	period (days)	daily Δv (m/s)
east-west stationkeeping	tangential	0.27-0.29	~ 80	0.0035
momentum dump	any	0.01	1	0.01
north-south stationkeeping	out-of-plane	45-50	365	0. 13

Synchronous Elements

Because of the ambiguity as to right ascension of the ascending node Ω and argument of perigee ω, Keplerian elements are not well-suited for work with geosynchronous orbits. A commonly used alternative is the set of *synchronous* orbital elements.[1] Semi-major axis a is replaced with drift D

$$D = -3\delta a/2A \tag{1}$$

where A is the nominal geostationary semi-major axis and δa is the deviation from it. Eccentricity becomes a two-dimensional vector with magnitude equal to the Keplerian eccentricity but pointing toward perigee

$$\vec{e}^T = \begin{pmatrix} e_x & e_y \end{pmatrix} = e\begin{pmatrix} \cos(\Omega+\omega) & \sin(\Omega+\omega) \end{pmatrix} \tag{2}$$

Inclination becomes a two-dimensional vector with magnitude equal to the Keplerian inclination pointing along the orbit normal projection in the equatorial plane

$$\vec{i}^T = \begin{pmatrix} i_X & i_Y \end{pmatrix} = i\begin{pmatrix} \sin\Omega & -\cos\Omega \end{pmatrix} \tag{3}$$

The eccentricity and inclination vectors are shown in Figure 1. Mean anomaly is replaced by sidereal angle measured from the geocentric equatorial x-axis

$$s = G + \lambda = G_0 + \psi \cdot (t - t_0) + \lambda \tag{4}$$

where G is the Greenwich sidereal (or hour) angle, λ is the sub-satellite longitude and ψ is the Earth's sidereal rotation rate, i.e. 360.985647°/day.

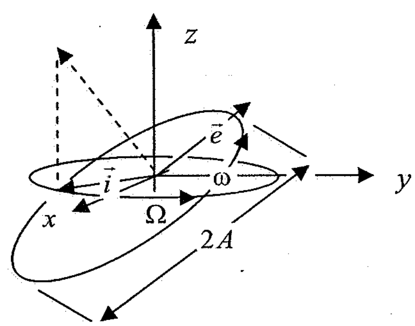

Figure 1 Orbital Elements

Position and Velocity

From the synchronous elements, the east-west (λ), north-south (θ) and radial (r) coordinates of the spacecraft may be computed as

$$\begin{aligned} \theta &= -i_x \cos s - i_y \sin s \\ r &= A - A(3D/2 + e_x \cos s + e_y \sin s) \\ \lambda &= \lambda_0 + D(s - s_0) + 2e_x \sin s - 2e_y \cos s \end{aligned} \tag{5}$$

Here, the subscript $_0$ denotes epoch position, *i.e.* where the spacecraft started out. As can be seen from the first equation, latitude variation is equal to the inclination magnitude. Similarly, the last equation shows that longitude variation is equal to twice the eccentricity magnitude.

Thruster burns may be used to control drift, eccentricity and inclination according to the following equations[9]

$$\begin{aligned}\Delta v_r &= v\left(\Delta e_x \sin s_b - \Delta e_y \cos s_b\right)\\ \Delta v_t &= v\left(\Delta D + 2\Delta e_x \cos s + 2\Delta e_y \sin s_b\right)\\ \Delta v_o &= v\left(\Delta i_x \sin s_b - \Delta i_y \cos s_b\right)\end{aligned} \tag{6}$$

where the subscripts r, t and o indicate the radial, tangential (increasing longitude) and orthogonal (increasing latitude) directions. The subscript b indicates that the sidereal angle is at the thruster *burn* time.

Geosynchronous stationkeeping strategies are described in Soop and in journal articles. Among them are the articles by Kamel on maximizing the time between longitude maneuvers and on eccentricity control and a more recent one by Kelly on combining longitude and eccentricity control.[2-4] In these papers, maneuvers are treated as instantaneous because commonly used chemical thrusters are so powerful that burns do not take much time. More recently, papers have addressed low-thrust stationkeeping.[5-6]

ENVIRONMENTAL EFFECTS

Even in geosynchronous orbits, spacecraft may move with respect to the ground. This is because of perturbations from the Earth's nonspherical gravitational field, solar radiation pressure and lunisolar gravity. The nonspherical geopotential causes spacecraft longitude to drift. Solar radiation pressure drives orbit eccentricity and applies a torque on the spacecraft. Lunar and solar gravity cause orbital inclination to drift. These effects are all countered by thruster firings.

Nonspherical Geopotential

GOES-East and -West are stationed on either side of a geopotential minimum at 105.3° west longitude. Without stationkeeping maneuvers, the spacecraft longitude would oscillate about this point with a period of more than 2 years.[7] Thrusters are fired when the spacecraft reaches the "lower" edge such that the spacecraft travels "up" across the box and turns around at the other edge. This is depicted in Figure 2 for a 0.1° box, and the maneuver period is about 25 days. Longitudinal acceleration over the box is about 0.001°/day^2 which makes the box width for daily maneuvers 0.00025° or 4 μrad. This corresponds to 140 meters east-west movement of the sub-satellite point.

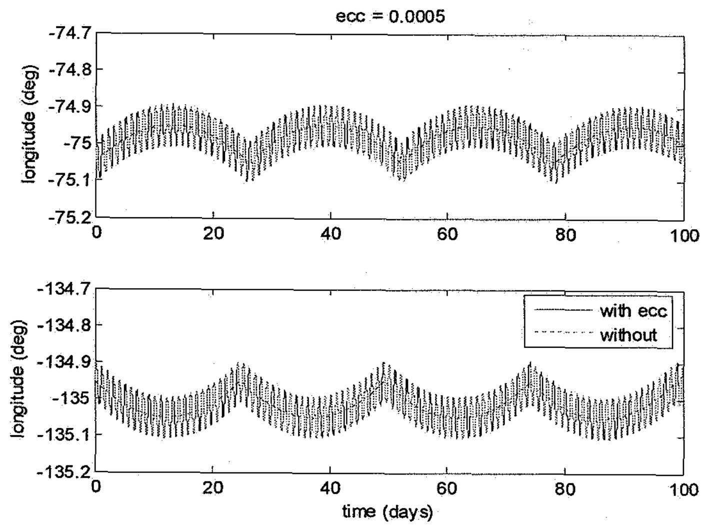

Figure 2 Longitude Stationkeeping

Solar Radiation Pressure - Eccentricity

Eccentricity is driven by solar radiation pressure. As solar radiation pressure accelerates the spacecraft on one side of the orbit, the radius stretches out on the other side. There, the solar radiation pressure slows the spacecraft making the orbit still more elongated. Integrated over a day, the average rate of change in the eccentricity vector is

$$\frac{d\vec{e}}{dt} = \frac{3 p_S A}{2 v m} \begin{pmatrix} -\sin s_S \\ \cos s_s \end{pmatrix} \tag{7}$$

where p_S is the solar radiation pressure, A is the spacecraft area, m is the spacecraft mass, v is the spacecraft velocity and s_S is the sidereal angle of the Sun in the equatorial plane as shown in Figure 3.

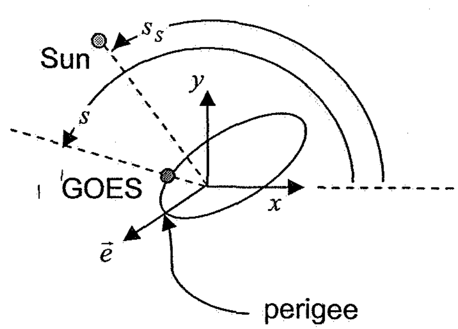

Figure 3 Solar Radiation Pressure and Eccentricity

For ballistic coefficients (mass-to-area ratio) over 50 kg/m^2, eccentricity grows less than 0.4×10^{-6} per day and can be controlled by choosing the maneuver time. Because longitude variation is twice the eccentricity, this eccentricity causes less than 1 μrad longitude variation. GOES I-M eccentricity is currently controlled by choice of maneuver time, and the typical eccentricity is less than 0.0005 which gives a longitude variation of only 17.5 μrad. Figure 4 shows the almost circular path of the eccentricity vector over the course of a year. If this level of eccentricity variation is acceptable, the trajectory can be centered on the origin with relatively small burns.

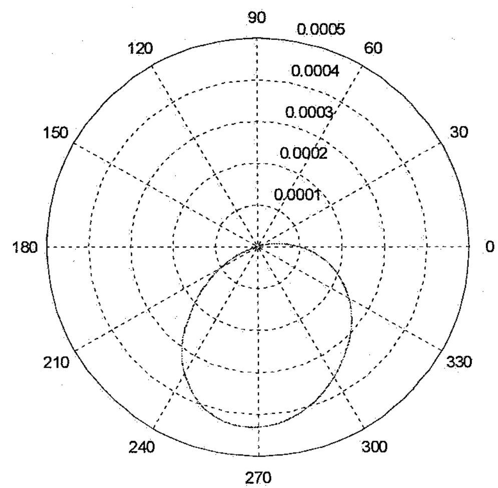

Figure 4 Annual Eccentricity Drift

Solar Radiation Pressure - Angular Momentum

Spacecraft angular momentum increases due to solar radiation pressure and offsets between the spacecraft center of pressure and center of mass. For an axially-symmetric spacecraft with a single solar array, the primary torque component is in the orbit plane perpendicular to the Sun line. To compute the momentum build-up, one can integrate Euler's equation in coordinates rotating with the spacecraft at angular velocity $\vec{\omega}$

$$\vec{M} = \frac{d\vec{H}}{dt} + \vec{\omega} \times \vec{H} \tag{8}$$

Here $\vec{M}$ represents the solar radiation torque and $\vec{H}$ represents the spacecraft angular momentum. $\vec{H}$ is actually the sum of the spacecraft rigid body angular momentum $I\vec{\omega}$ and the wheel angular momentum $\vec{H}_w$

$$\vec{H} = I\vec{\omega} + \vec{H}_w \tag{9}$$

For a perfect attitude control system with no angular acceleration, the differential equation for the wheel angular momentum is

$$\dot{\vec{H}}_w = \vec{M} - \vec{\omega} \times \left(I\vec{\omega} + \dot{\vec{H}}_w \right) \tag{10}$$

Because it is tied to the Sun line, the angular momentum rotates in spacecraft body coordinates. To estimate the size of the momentum buildup, we used GOES I-M dimensions minus the solar sail which balances the torque from the solar array.[8] Assuming that the spacecraft center-of-mass is at the center of the spacecraft body, the angular momentum builds up in inertial space as shown at the top of Figure 5. In spacecraft body coordinates, it cycles between the in-plane spacecraft axes, with growing amplitude. Over 24 hours, the angular momentum increases by about 20 Nms.

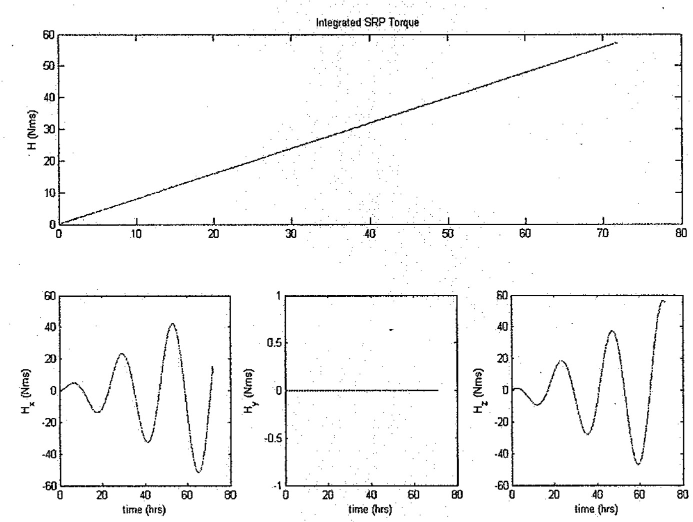

Figure 5 Angular Momentum Growth

Lunisolar Gravity

Orbit inclination is the deviation of the orbit normal from to the celestial north pole and is driven by solar and lunar gravity. Because the Earth is also accelerated by the Sun and Moon, the spacecraft acceleration relative to the Earth is written as

$$\frac{d^2\vec{r}}{dt^2} = -\frac{\mu}{r^3}\vec{r} + \sum_{k=1}^{2} \mu_k \left(\frac{\vec{r}_k - \vec{r}}{\left|\vec{r}_k - \vec{r}\right|^3} - \frac{\vec{r}_k}{r_k^3} \right) \tag{11}$$

where $\vec{r}$ and $\vec{r}_k$ are the spacecraft and Sun/Moon position vectors in Earth-centered coordinates, and μ and μ_k are the Earth and Sun/Moon gravitational constants.[9]

Because the inclination vector is parallel to the angular momentum vector, its direction obeys Euler's equation

$$\dot{\vec{h}} = \vec{r} \times \frac{d^2\vec{r}}{dt^2} \tag{12}$$

where $\vec{h}$ is the spacecraft angular momentum taken about the Earth-spacecraft center of mass and normalized by the spacecraft mass. In terms of $\vec{h}$, the components of the inclination vector are

$$\vec{i} = \begin{pmatrix} \sin^{-1}(h_x/h) \\ \sin^{-1}(h_y/h) \end{pmatrix} \tag{13}$$

Both Sun and Moon push the inclination vector toward the vernal equinox, and although the Sun's pull on the spacecraft is larger than that of the Moon, the Moon actually has the greater influence because it is the differential force that drives inclination. Figure 6 shows the annual 1° inclination drift.

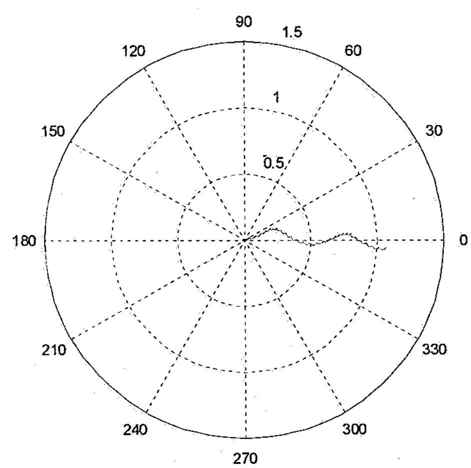

Figure 6 Annual Inclination Drift

MANEUVER RECOVERY

There are two reasons for the INR maneuver recovery period. First, because of Δv uncertainty, the new orbit is not well known until there is enough post-burn tracking data. GOES-R 3σ position uncertainty must be less than 100 m on each axis. Second, when thrusters are fired, the solar array may vibrate and cause attitude oscillations. GOES-R 3σ attitude and angular velocity errors must be less than 360 μrad and 100 μrad/s on each axis respectively. Because the instruments are only required to meet their INR requirements once the spacecraft does, GOES-R must recover quickly.

Orbit Knowledge

GPS has long been seen as the key to onboard orbit determination and rapid maneuver recovery.[10,11] Until receivers sensitive enough to track GPS sidelobe signals were developed, geosynchronous coverage was spotty, and the time needed to converge was long. With new receivers having 22-25 dB-Hz sensitivity, it is possible to track multiple GPS vehicles from geostationary altitude most of the time as shown in Figure 7.[12, 13]

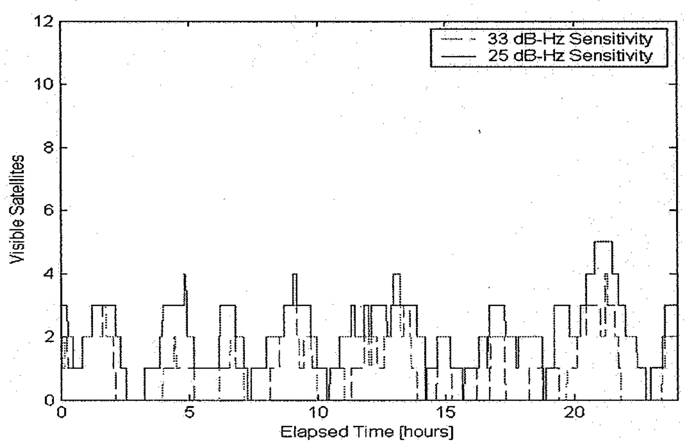

Figure 7 GPS Availability

High-fidelity simulations have been done showing that the GOES-R position knowledge requirement of 100 m per axis can be met during normal operations.[14] This study assumed the GPS Block IIA antenna pattern and 9 dB receiving antenna gain. To estimate the time required for orbit knowledge to converge following a maneuver, a simple covariance analysis was performed in which the initial velocity covariance was increased from its steady-state value by a value corresponding to the Δv uncertainty.

The covariance P was propagated using the orbit state transition matrix Φ and the process noise covariance Q

$$P \leftarrow \Phi P \Phi^T + Q \tag{14}$$

The information matrix P^{-1} was updated by adding the information in each observation

$$P^{-1} \leftarrow P^{-1} + G^T W G \tag{15}$$

where W was the observation weighting matrix and G was the derivative of the GPS range observations with respect to spacecraft position.

Figure 8 shows the position knowledge uncertainty (3σ) following north-south maneuvers with different Δv uncertainties. The observation interval is 10 seconds, the observation standard deviation is 1 m, and there are 4 GPS vehicles in view. With daily maneuvers, the expected Δv uncertainty would be less than 1 cm/s. Such small uncertainties are practically undetectable. The solid curve is for a 1 m/s Δv uncertainty comparable to that of an annual 50 m/s north-south maneuver. The dash-dot middle curve is for a 10 cm/s Δv uncertainty. For GOES-R, where position is required only to 100 m, it should be possible to fly through daily maneuvers without violating orbit knowledge requirements.

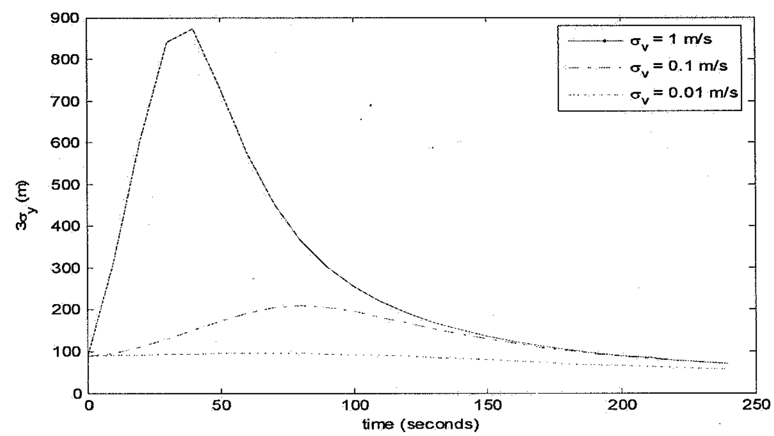

Figure 8 Post-Maneuver Position Knowledge

Attitude Control

Thruster firings affect attitude control rather than knowledge. To quantify their effect, the spacecraft was modeled as shown in Figure 9. The main body was replaced with a sphere free to rotate through the angle φ about a diameter perpendicular to the page. Its 2000 kg mass and 1.5 m radius were chosen to be close to those of GOES I-M. The solar array was replaced by a point mass at the end of a cantilever beam. Its 80 kg mass, 5 m length, 0.25 Hz natural frequency ω_n and 0.01 damping ratio ς were also chosen to match the GOES I-M solar array.

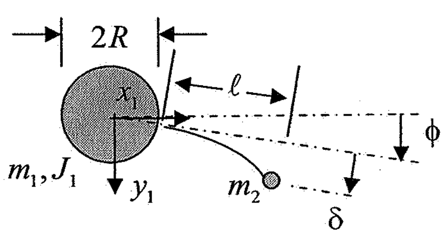

Figure 9 Spacecraft-Solar Array Model

For small angular displacements ϕ and small linear displacements δ, the open-loop equations for the spacecraft and the solar array are

$$(m_1 + m_2)\ddot{x}_1 = F_x \tag{17}$$

$$(m_1 + m_2)\ddot{y}_1 + m_2(R+\ell)\ddot{\phi} + m_2\ddot{\delta} = F_y \tag{18}$$

$$J\ddot{\phi} + m_2(R+\ell)\cdot(\ddot{y}_1 + \ddot{\delta}) = M_T + M_C \tag{19}$$

$$m_2(\ddot{y}_1 + (R+\ell)\ddot{\phi} + \ddot{\delta}) = -c\dot{\delta} - k\delta \tag{20}$$

where F_x and F_y are thruster forces, M_T and M_C are thruster and control torques and J is a combined moment of inertia

$$J = J_1 + m_2(R+\ell)^2 \tag{21}$$

The solar array damping and spring constants c and k are derived from the damping ratio and natural frequency as

$$c = 2\varsigma\omega_n m_2 = 0.5924\, kg/s \tag{22}$$

$$k = \omega_n^2 m_2 = 46.5281\, kg/s^2 \tag{23}$$

A proportional-integral-derivative (PID) control system as shown in Figure 10 was chosen because an integral term was needed to eliminate the steady-state error to the long thruster pulse. The control system roots were chosen to lie in a Butterworth pattern with natural frequency equal to the 0.01 Hz bandwidth ω_c.

$$\begin{aligned} K_r &= 2\omega_c J \\ K_p &= 2\omega_c^2 J \\ K_i &= \omega_c^3 J \end{aligned} \tag{24}$$

The bandwidth was chosen to be well below the solar array vibration frequencies so as not to excite them. Knowledge of the applied torque was assumed accurate to 5% and was used in feedforward fashion to reduce the input torque M.

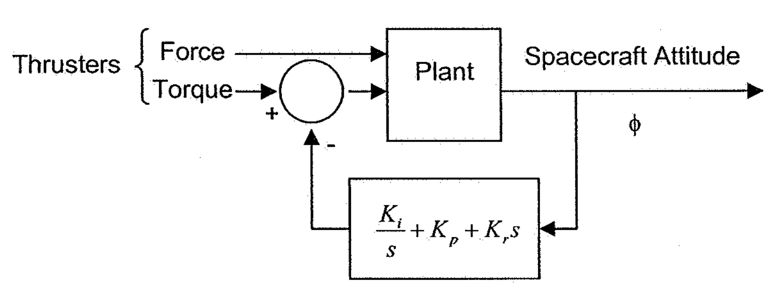

Figure 10 Attitude Control System

To impart 20 Nms of angular momentum to the spacecraft with a 0.1 N thrust and a 1 m moment arm, one would fire for 200 seconds. Figure 11 shows the attitude error and angular velocity predicted for such a firing. The attitude error peaks at the start and end of the pulse but stays below the 360 μrad requirement. The angular velocity also stays well below the 100 μrad/s requirement. If controller bandwidth could be doubled to 0.02 Hz, the peak errors would drop by a factor of 4 to 50 μrad. This is still well below the solar array natural frequency. Although not shown here, the response to a pure force, rather than a pure torque, was also computed but was much smaller. Thus, it seems quite plausible that attitude control can be maintained through thruster firings.

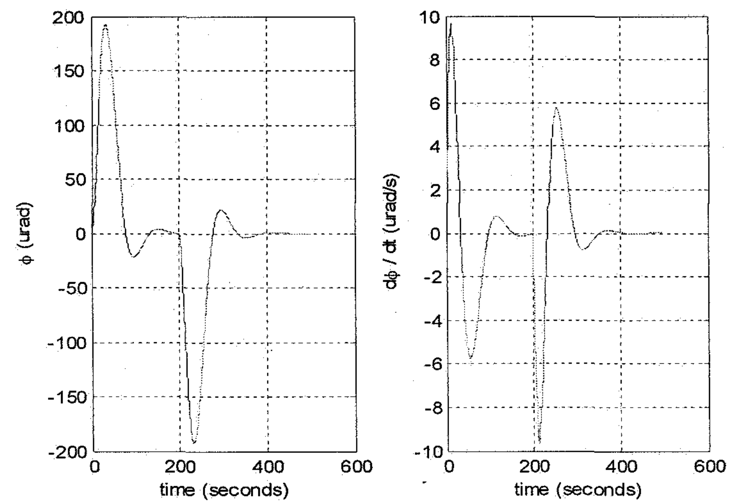

Figure 11 Thruster-Induced Attitude/Angular Velocity Error

CONCLUSION

This paper has examined the feasibility of doing GOES-R stationkeeping and momentum dumping with frequent low-thrust burns assuming a high-sensitivity GPS receiver and attitude control system with thruster torque compensation. The results suggest that:

1. Longitude and latitude control can be tightened from the current 1° to 0.1° or less.
2. Orbit knowledge requirements can be met during and after daily thruster firings.
3. Attitude control requirements can be met during and after daily thruster firing.

While encouraging, these results are preliminary, and concerns remain as to choice of thruster, development of a flight-qualified high-sensitivity GPS receiver and design of the feedforward attitude control system.

ACKNOWLEDGMENT

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