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Engineering Notes

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Earth-Pointing Recovery

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I. Introduction

GEOSTATIONARY satellites need to point at Earth continuously. The yaw axis (x axis) is directed toward the center of the Earth.[1] The pitch axis (z axis) of the satellite is along the negative orbit normal, and roll axis (y axis) is perpendicular to the other two such that the unit vectors along the three axes form an orthonormal triad. This completes the orbit frame of reference. The three-axis control is achieved using an Earth scan sensor mounted to measure the Earth chord about the roll and pitch axes. The momentum bias system indirectly controls the yaw. The loss of Earth pointing happens due to various reasons such as Earth sensor failure or dazzle, thruster failure, control glitch, etc. Recovery procedures for such satellites are geometrical in nature. The recovery is assisted by a sun sensor mounted along the positive or negative roll axis. The sun in the wide field of view of the sun sensors ensures sun acquisition along the roll axis. This approach requires the sun and Earth directions seen from the satellite to be orthogonal. This situation occurs at 0600/1800 hrs satellite local time. A roll rate is then initiated to capture the Earth by the Earth sensor. This procedure sometimes requires a long wait for the favorable orthogonal geometry. Next we cite two approaches that do not need to point the roll axis at the sun to recover Earth pointing. First is the technique in Ref. 2 that has been demonstrated on the Orbital Test Satellite after allowing the spacecraft to approach flat spin conditions. The second technique requires a star sensor. This method invokes the lost-in-space concept after arresting the angular rates on the satellite to obtain the attitude with respect to the inertial frame. Because the desired attitude in Earth pointing is known from the orbit data, the attitude maneuver to point to Earth can be deduced. This technique, however, requires storing onboard the star catalog, star identification algorithm, and onboard ephemeris. Finally, we cite one more technique[3,4] that needs the roll axis pointing at sun and then uses a star sensor. This approach eliminates the waiting time. The star sensor is mounted orthogonal to the roll axis and is able to capture Canopus when there is a roll rate. Canopus is a very bright star, and the star sensor is designed to detect its brightness. After holding on to the star within the field of view of the star sensor, the attitude maneuver for Earth pointing becomes straightforward.

The method suggested in this Note requires the roll axis pointing at sun and it eliminates the waiting time mentioned earlier. The Earth scan sensor and sun sensor are used. This approach consists of two consecutive three-axis attitude maneuvers. Simulations have been carried out to obtain the sensitivity of the recovery maneuver. Later in this Note, a simplified version of this method that can be invoked onboard is also presented. This simplification can be applied when a communication link failure occurs during the loss of the Earth. Incidentally, a reviewer has pointed out the similarity of a part of the proposed technique to a patent[5] that uses a slit sun sensor. Nevertheless, the mathematical treatment, simulation, and onboard version presented here based on the ideal sun direction without the need for sun vector measurements in the spacecraft body frame are the novel developments of the technique presented here.

II. Recovery Attitude Maneuvers

The recovery technique is as follows. When the loss of Earth pointing is noticed from the telemetry, the angular rates of the satellite are brought to zero. First, depending on the measurements of the sun sensor, the sun acquisition is carried out with a wide field-of-view coarse sun sensor on the positive or negative roll axis. The roll axis nearest to the sun is then made to point at sun. We then know whether the positive or negative roll axis is toward the sun, but the orientation of yaw and pitch axes are unknown. Let this orientation in the inertial frame be denoted as O_{sun}.

The first attitude maneuver is to obtain an orientation wherein the sun vector in the body frame is identical to the sun vector in the orbit reference frame corresponding to the Earth pointing orientation at the epoch of recovery. The azimuth and elevation of the sun in the body frame shall be the same as in the Earth-pointing orbit reference frame, denoted as O_{EP}.

The sun makes an azimuth α and elevation δ in the orbit reference frame O_{EP}. These can be determined using the satellite orbit at the time planned for initiating the recovery.[1] The inertial position of the sun is used to derive the sun vector $\boldsymbol{V}_s = [V_s(1), V_s(2), V_s(3)]$ in the orbit frame of reference. We then calculate the azimuth and elevation[1] as follows:

$$\alpha = \tan^{-1}[V_s(2)/V_s(1)], \qquad \delta = \sin^{-1}[V_s(3)] \tag{1}$$

A ground-based software realizes this with higher accuracy than the wide field-of-view coarse sun sensor. It shall be shown later that, with limited computations, the position of the sun can be obtained in the orbit reference frame fairly accurately for the geostationary satellite. Furthermore, in the Note we obtain the sun position from its ephemeris. Compute the angles, ang_x, ang_y, and ang_z, in the following manner.

When the positive roll axis points at the sun,

$$\text{ang\_x} = -\delta, \qquad \text{ang\_y} = 0, \qquad \text{ang\_z} = 90 - \alpha \tag{2a}$$

and when the negative roll axis points at the sun

$$\text{ang\_x} = \delta, \qquad \text{ang\_y} = 0, \qquad \text{ang\_z} = 270 - \alpha \tag{2b}$$

In this notation, the angle of rotation about the appropriate axis is indicated. The attitude matrix corresponds to a sequence[1] of rotation (2–1–3) and is denoted as A_{one}. When A_{one} is applied on O_{sun}, the azimuth and elevation of the sun will be the same as in the orbit reference frame of the satellite when it is Earth pointing. This orientation is O_{AE}. Next, the principle of Euler-axis rotation is used to capture the Earth center along the yaw. Set

$$\text{vec\_x} = \cos\alpha\cos\delta, \qquad \text{vec\_y} = \sin\alpha\cos\delta$$

$$\text{vec\_z} = \sin\delta \tag{3a}$$

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The second maneuver is a rotation with attitude quaternion q_manu given by

$$\text{q\_manu}(1) = \text{vec\_x}\ \sin(\nu/2), \qquad \text{q\_manu}(2) = \text{vec\_y}\ \sin(\nu/2)$$

$$\text{q\_manu}(3) = \text{vec\_z}\ \sin(\nu/2), \qquad \text{q\_manu}(4) = \cos(\nu/2) \quad \text{(3b)}$$

Here, angle ν is set to 180 deg first. The variation of ν from 0 to 180 deg initiates a rotation in the spacecraft. The second attitude maneuver q_manu is applied on O_{AE}. This instantaneous orientation will be represented as O_{ROT}. The Euler-axis rotation about the vector (vec_x,vec_y, and vec_z) ensures that the azimuth and elevation of sun is the same in O_{AE} and O_{ROT}. The trace of the yaw vector describes a circle denoted as C_{ROT}. If Earth-pointing recovery is not realized, the variation of ν is then extended from 180 to 360 deg by setting $\nu = 360$ deg in Eq. (3b). This is shown in Fig. 1.

Here, the circle C_{EP} is the trace of the yaw vector in normal Earth pointing mode O_{EP} when the Euler-axis rotation is about the roll y axis.

Because the azimuth and elevation of the sun in normal Earth pointing orientation O_{EP} matches with those of O_{ROT}, the intersection of these circles traced by the Euler rotation is Earth pointing. The instantaneous yaw (marked in Fig. 1) will coincide with the Earth pointing vector along the x axis by tracing the circle C_{ROT}. The effect of orbital motion during the time of the maneuvers can significantly contribute to the deviation in Earth pointing apart from the rate of rotation effected on the spacecraft.

III. Simulation

The satellite ephemeris and the sun ephemeris are used to compute the instantaneous azimuth and elevation of the sun in the satellite orbit frame of reference, O_{EP}. The time of recovery, t_recv, is decided subsequent to the occurrence of loss of lock. First, the safe mode of pointing at the sun by the nearest roll axis is realized. This is done based on the direction measurements of the sun in the field of view of the sun sensors mounted on the positive or negative roll axis. The polarity of the axis pointing at the sun is known onboard.

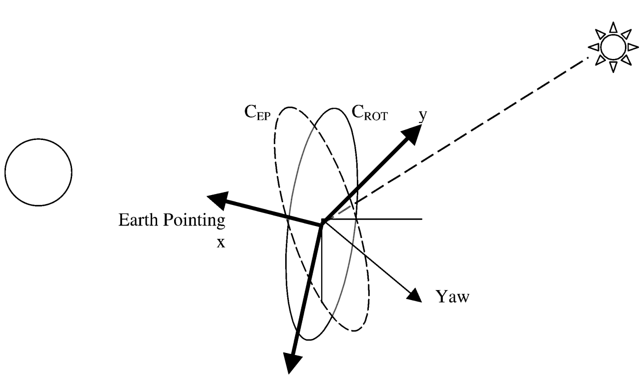

Fig. 1 Geometry of Earth recovery.

The attitude maneuvers $A_{\text{one}}^{\text{t\_recv}}$ and q_manu$^{\text{t_recv}}$ are computed for the selected time t_recv using Eqs. (1–3). For practical purposes the value of ν in q_manu$^{\text{t_recv}}$ is set to 180 deg, and the command is generated. The rotation about the Euler axis in Eq. (3b) is assumed to be at a nominal rate of 1 deg/s. To attain 180 deg at this rate, an elapsed time of 3 min is accounted for to arrive at the time t_recv. Also, for completing one circular trace, the duration is 6 min, and the orbital motion in this duration is 1.5 deg. The actual duration for effecting the attitude maneuvers $A_{\text{one}}^{\text{t\_recv}}$ and q_manu$^{\text{t_recv}}$ can be more or less than the expected duration calculated with the nominal rate of 1 deg/s about any axis. The deviation in time with respect to the decided time t_recv is ($\pm\Delta t$). In Fig. 2, the deviation ($\pm\Delta t$) is shown on either sides of the center of the time axis t_recv. The orbital motion during the deviated time adds to the inaccuracy in the recovery of exact Earth center pointing. We also consider the nominal accuracies of the sun sensor and that of the control system to be 0.5 and 0.25 deg, respectively. The results from a sensitivity study of the orbital motion with and without the sensor and control errors are shown in the Fig. 2.

The included angle in Fig. 2 is between the Earth center vector from the spacecraft and the instantaneous yaw axis. The thick line in the Fig. 2 typically represents the case when sensor or control inaccuracies are absent, and the graph indicates the variation of the minimum included angle. Note that while using the attitude maneuvers $A_{\text{one}}^{\text{t\_recv}}$ and q_manu$^{\text{t_recv}}$ at the time t_recv, the minimum included angle is not zero. This is due to the orbital effect when ν is set equal to 180 deg in q_manu. The Earth pointing may occur ahead or after the Euler rotation of 180 deg depending on the orientation O_{AE}. Furthermore, we use $A_{\text{one}}^{\text{t\_recv}}$ and q_manu$^{\text{t_recv}}$ for a time t_inst equal to (t_recv $\pm\Delta t$) on the time axis. The corresponding variation of the minimum of the included angle is shown in Fig. 2. The actual sun and satellite positions are obtained from the respective ephemeris at the request time t_inst. The minimum included angle is within the linear range of horizon sensor or other Earth sensors when Δt is less than ± 5 min. Next, in the simulation, we include a case where the sensor and control errors are as mentioned earlier. These errors are assumed to be Gaussian in nature. The dashed line in Fig. 2 indicates the minimum of the included angle for this case. This is seen to be somewhat shifted from the no-noise case. For a selected input time t_inst, the experiment corresponds to a single random run.

This study ensures that the Earth capture is assured for ± 3 min deviations from the target time t_recv. Communication signal levels with a wide angular coverage can also be used to assist the Earth capture.

IV. Onboard Version

Equations (2) and (3) are very simple to realize onboard, yet an accurate determination of the sun's azimuth α and elevation δ in the orbit frame of reference using Eq. (1) is a concern. The orbit model and sun's ephemeris can be made available onboard, and then the computation is straightforward. Here, a simple alternative is suggested for geostationary satellites. First, compute the azimuth angle. It sweeps 360 deg in one day. This assumed mean motion

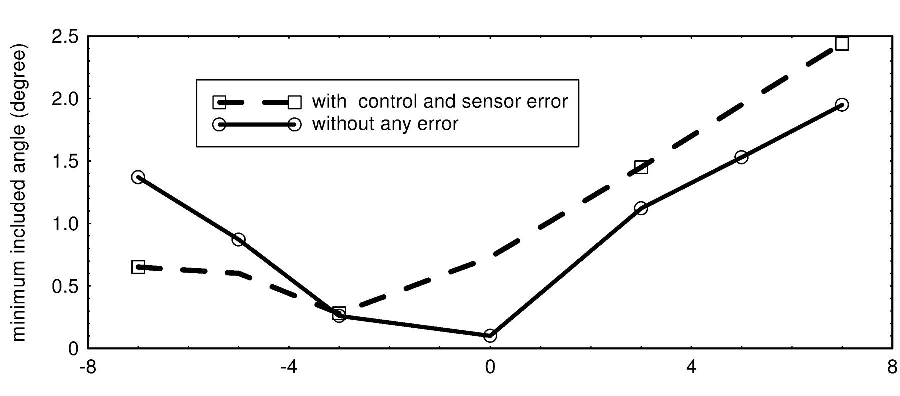

Fig. 2 Minimum included angle for shifts Δt from the planned time t_recv.

deviates from the actual due to the equation of time and because the period of the satellite's orbit is not exactly 86,400 s. The equation of time[6] can be fit by a polynomial and accounted for a span of every 100 days and can be made more accurate for a shorter duration. The approximate values of α and δ are computed as follows:

$$\alpha(\text{t\_recv}) = \alpha(\text{t\_init}) + \alpha(\text{mean}) + \alpha(\text{equation of time})$$
$$\alpha(\text{mean}) = \text{t\_diff} \times (360/86400)$$
$$\alpha(\text{equation of time}) = \sum \alpha_i (t)^i$$
$$\text{t\_diff} = [\text{t\_recv} - (\text{t\_init} + \text{nod} \times 86400)] \tag{4}$$
$$\text{nod(integral)} = (\text{t\_recv} - \text{t\_init})(\text{inseconds})/86,400$$
$$\delta = \sum \delta_i (\text{nod})^i \tag{5}$$

The following experiment was performed. The azimuth α(t_init) in the satellite orbit reference frame at the start of the year, that is, 1 s after midnight of the last day of the previous year, is known. Along with this initial azimuth value, the mean motion, and the correction obtained from a polynomial fit for the equation of time, an approximate azimuth is computed. The value of nod is the day number of the year. Here, Eq. (4) is very simple for onboard computation. The declination of the sun is also fit by a polynomial. In the experiment, a seventh-degree polynomial fit was used for both azimuth and declination. With the use of Eqs. (4) and (5), the angles α and δ are determined, and subsequently, the two attitude maneuvers are obtained from Eqs. (2) and (3). The accuracy of this model was typically within 0.3 deg for the satellite with inclination less than 0.1 deg, and eccentricity was less than 0.005. This algorithm can be useful for onboard applications after the sun pointing acquisition. It becomes useful when the communication link to the Earth is cut off.

V. Example

A spacecraft stationed at 40° east longitude with an inclination of 0.052 deg with respect to the equatorial orbit and eccentricity equal to 0.00152 is considered. The negative roll axis is set to the sun on 21 June at 1200 hours Universal time, denoted as time t_recv. The sun makes a right ascension of 90.24 deg and a declination of 23.43 deg in the Earth centred inertial frame. At this epoch, the sidereal angle and longitude sum up to 130.53°. The sun azimuth in the orbit frame of the spacecraft is 221.53 deg and the elevation is −23.43 deg. The pitch and yaw required to make the negative roll axis point at the sun are −48.54 and 23.43 deg, respectively. The A_{one} matrix is computed for pitch and yaw rotations of 48.54 and −23.43 deg, respectively, in Eq. (2). The Euler-axis vector in Eq. (3a) is −0.683, −0.606, and −0.406. When ν is set to be 180 deg, the second attitude maneuver q_manu is obtained. In the onboard version for the epoch t_init to be 1 s after the start of January 2005, it was found from ground simulation that α(t_init) was 40.275 deg. Furthermore, when using the method described in Eqs. (4) and (5) was used, the values of pitch and yaw in the A_{one} matrix are 48.26 and −23.33 deg, respectively, whereas the Euler-axis vector in Eq. (3a) is −0.685, −0.611, and −0.396. It is clear in this case that the onboard model (valid for equatorial orbit) is in close agreement with the exact model. This example illustrates the computational simplicity of the onboard approach. The two attitude maneuvers are precomputed and applied about 3 min in advance considering the realizable rate.

VI. Conclusions

A geometrical approach for Earth pointing recovery is presented that eliminates waiting for the orthogonal conditions. Earth scan sensor and sun sensor assist this recovery, which requires two attitude maneuvers. The sensitivity of this alternative approach has also been analyzed. A simplified onboard version is also presented and illustrated in an example.

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