High-Fidelity Orbit Propagator for Precise Antenna Pointing in LEO Satellite Operation

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***Abstract*—In this paper, the development of an antenna pointing control system for the operation of satellites orbiting in Low-Earth Orbit is presented. A high-fidelity orbital dynamics model is developed for the prediction of the satellite's position and velocity. It includes higher geo-potential harmonics of the non-spherical Earth, the gravitational attraction from the Sun and the Moon, and other non-conservative forces, such as atmospheric drag and solar radiation pressure. A high precision orbit integrator based on the Burlish-Stoer algorithm plays a crucial role in integrating the equations of motion and find the satellite's states advance in time. The states from the Earth-Centered Inertial Coordinate are transformed to the Topocentric Horizon Coordinate to find the satellite's motion viewed from the control station on the ground. The ephemeris file that comprises vital information for the satellite tracking, such as azimuth and elevation angles, can be generated when the pass is detected. The antenna pointing control system then tracks the predicted trajectory. The proposed system is verified during the operation of the Thaichote satellite. It is shown that the predicted orbital states are very close to those determined from the onboard GPS receiver. It exhibits only some small residuals in position and velocity. The tracking performance is also shown through the high signal level received from the S-band antenna throughout the tracking course.**

Keywords—Satellite Orbit Propagation, Antenna Pointing Module, Thailand's Earth Observation Satellite

I. Introduction

Unlike satellites in Geostationary orbit, the orbital period of satellites in Low-Earth Orbit (LEO) are not synchronized with the Earth's orientation, and their positions are not fixed relative to the control station on the ground. Thaichote satellite [1], for instance, is operating at an altitude of around 822 km. There are about 4 passes per day, and each pass has a different trajectory when viewed from the ground station. Therefore, the station's S-band antenna used for telemetry and command needs to track the spacecraft along the trajectory to keep the satellite link. This can simply be performed by pointing the antenna along the predicted path.

Prediction of the satellite motion in LEO environment, however, is not trivial. Apart from the gravitational force of the central body, i.e. the Earth, there are some other perturbations, both conservative where the total energy is preserved and non-conservative, acting on the spacecraft. The equations of motion are naturally non-linear, and their closed-form solutions are possible only when some strict assumptions are given. The SGP4 orbit propagator [2], for example, uses the truncated version of the geo-potential harmonics up to 4 terms. Its accuracy has proven sufficient for many operations, and has been applied to satellite control for decades. However, for narrow beam data communication, such truncated dynamics may not be able to provide sufficient pointing accuracy between the dish-shape antenna and the moving spacecraft, hence, poor received communication signal.

In this research, we develop an orbit propagator to predict the satellite's position which can be used for precise pointing of the ground station's antennas. The modeling of perturbing force influencing the satellite motion will be described in the next section. The orbital integrator and satellite pass prediction module are then discussed. The experimental results obtained from the Thaichote satellite tracking system using the S-band antenna are shown, and the conclusion is drawn in the final section.

II. Orbital Dynamics

A. Perturbing Force Model

In LEO, the dominant force acting on a spacecraft is contributed by the non-spherical Earth. Its gravitation force can be modeled through the geo-potential function, U, which can be described using spherical harmonics function as

$$U = \frac{\mu}{r}\left[1 + \sum_{l=2}^{\infty}\sum_{m=0}^{l}\left(\frac{R_E}{r}\right)^{l} P_{lm}\left[\sin\phi_{sat}\right]\left\{C_{lm}\cos\left(m\lambda_{sat}\right) + S_{lm}\sin\left(m\lambda_{sat}\right)\right\}\right] \tag{1}$$

where μ is the gravitational constant, R_E is the Earth's mean equatorial radius, C_{lm} and S_{lm} are gravitational coefficients, l and m are order and degree of the harmonics, respectively, and r, ϕ_{sat}, λ_{sat} are radius, latitude and longitude of the satellite's position vector, respectively. In practice, the gravitational coefficients are determined using the orbiting satellites. In our work, we adopt WGS84 [1], GEM10B [2], and JGM-3 [3] gravity models. Degree and order of the gravitational harmonics can be included up to an arbitrary number, however, a truncated version with 36×36 terms is sufficient for our mission operations.

The gravitational attraction from the Sun and the Moon can cause long-term orbital plane drift, and they are modeled as point-mass third-body objects as

$$\ddot{\vec{r}}_{\otimes sat} = -\frac{Gm_{\oplus}\vec{r}_{\oplus sat}}{r_{\oplus sat}} + Gm_3\left(\frac{\vec{r}_{sat3}}{r_{sat3}^3} - \frac{\vec{r}_{\oplus 3}}{r_{\oplus 3}^3}\right) \quad (2)$$

where $\ddot{\vec{r}}_{\otimes sat}$ is the acceleration vector acting on the satellite caused by the object, $Gm_{\oplus}$ and Gm_3 are the gravitational constants of the Earth and the third-body objects, respectively. The position vector from to the third-body object with respect to the Earth and the spacecraft are described by $\vec{r}_{\oplus 3}$ and $\vec{r}_{sat3}$, respectively. To find such vectors, we employ the Jet Propulsion Laboratory Development Ephemeris (DE) model [5] to generate precise Solar and Lunar ephemeris.

Some non-conservative forces, though very small compared to their aforementioned conservative counterparts, can cause secular variations in some orbital elements, especially the decaying of the altitude caused by the atmospheric drag. It directly affects the satellite's groundtrack and causes a time-parabolic function drift. The Jacchia-Roberts model [6] has been adopted for the modeling of such effect. It computes the atmosphere density (ρ) from data on solar activity index, $F_{10.7}$, and from the geomagnetic index, K_p. The perturbing force is then calculated from

$$a_d = -\frac{1}{2}\rho C_D \frac{A_D}{m} v_r \vec{v}_r \quad (3)$$

where, $\vec{v}_r$ is the relative velocity vector with respect to the atmosphere, A_D is the effective area and m is the spacecraft's mass. The C_D value is estimated as a free parameter in the orbit determination.

The solar activity also affects the solar radiation pressure (SRP) acting on the spacecraft surface. A cylindrical cone of the Earth's shadow is assumed in the evaluation of the perturbing force. The acceleration of a satellite due to the SRP is modeled as

$$a_{rp} = \nu P_S AU^2 C_R \frac{A_R}{m} \frac{\vec{r} - \vec{r}_S}{\|\vec{r} - \vec{r}_S\|^3} \quad (4)$$

where P_S is the force due to solar radiation at one astronomical unit (AU) acting on a unit area, A_R is the area exposed to the Sun rays, r_S is the Sun position vector in inertial coordinates. Equation (4) is commonly used in orbit determination programs with the option of estimating C_R as a free parameter. Orbital perturbations resulting from shadow transits are treated by the introduction of shadow function ν, that measures the degree of Sun's occultation by a body like the Earth or the Moon.

B. The Orbit Propagator

The satellite's position and velocity along each axis in an Earth-Centered Inertial Coordinate (IJK) form the orbital state vector $x \cong \begin{bmatrix} r_I & r_J & r_K & v_I & v_J & v_K \end{bmatrix}^T$, which can be obtained by solving the equations of motion:

$$\dot{x} = \begin{bmatrix} v_I \\ v_J \\ v_K \\ -\frac{\mu}{r^3} r_I + a_{p_I} \\ -\frac{\mu}{r^3} r_J + a_{p_J} \\ -\frac{\mu}{r^3} r_K + a_{p_K} \end{bmatrix} \quad (5)$$

where a_{pi} denotes the combined acceleration from each perturbing force described in the previous subsection.

An analytic solution can be found in some special cases where the truncated version of perturbations is applied and the equations of motion are linearized. The accuracy of the generated ephemeris, however, may be limited. Its numerical approach counterpart is more preferable when a high-fidelity solution is required.

In our propagator, we apply the Bulirch-Stoer algorithm [3] to integrate the spacecraft's motion. This extrapolative method is considered to be the fastest and most accurate, thanks to the Richardson extrapolation incorporated into the algorithm. This integrator is particularly suitable for near-circular orbits where the dynamics topology is quite uniform throughout the integration course.

III. ANTENNA POINTING MODULE

To find the satellite's position viewed from the ground station, it requires the vector transformation from The Earth-Centred Coordinate to the Topocentric Horizon Coordinate shown in Fig. 1. The origin is at the ground station and the local horizon forms the fundamental plane. The X axis points due north, the Y axis points east, and the Z axis point radial outward from the site. The data that are vital for commanding the antenna pointing are azimuth and elevation angle. Range and Doppler shift can also be utilized for orbit determination purpose. Note that, as the globe is not a perfect sphere, geodetic latitude (ϕ) shall be used instead of geocentric latitude (ϕ'), and they relate through

$$\tan\phi' = (1 - e_{\oplus}^2)\tan\phi \qquad (6)$$

where the Earth's eccentricity $e_{\oplus} = 0.081819221$.

The antenna pointing reference trajectory can be generated when the satellite pass with high enough elevation angle and long enough pass duration is detected during the orbit propagation course.

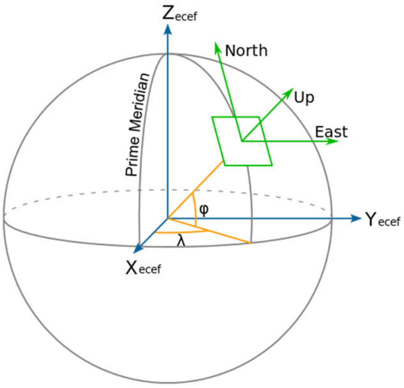

Fig. 1 Topocentric Horizon Coordinates.

IV. Experimental Results

The proposed antenna pointing module has been tested on the S-Band antenna at the Thaichote mission control center. The orbit propagator uses the determined orbital vector obtained from the GPS-based navigation system as the initial condition. It then propagates the equations of motion forward in time. Fig. 2 and 3 show the position and velocity residuals between the predicted states and the determined states using the onboard GPS receiver during the prediction period of 10 orbital revolutions. It can be seen that the propagated satellite orbit is very close to the measured one. The RMS of position and velocity residuals are only 416 m and 0.417 m/s, respectively.

After performing the coordinate transformation, typical azimuth and elevation angles of the satellite motion over the ground station are shown in Fig. 4, and 5. The received signal level of the S-band antenna, when pointing along a reference trajectory is shown in Fig. 6. It is obviously shown that the signal strength is high above the predefined threshold of -43 dB throughout the trajectory.

V. Conclusion

We have presented the development of an antenna pointing algorithm for the LEO satellite tracking system. The key part is on the prediction of the satellite states, where high-fidelity force models, i.e. the higher geopotential harmonics, the third-body attraction, as well as some non-conservative forces are taken into account. Using an accurate integrator based on the Burlish-Stoer algorithm and the coordinate transformation toward the Topocentric Horizon frame, the ephemeris data can be generated for the antenna pointing control system. The experimental results show that the predicted satellite trajectory is precise. Although using a simple structure with open-loop control, the S-band antenna pointing performance is sufficiently accurate. The received signal level is high above the threshold throughout the tracking course.

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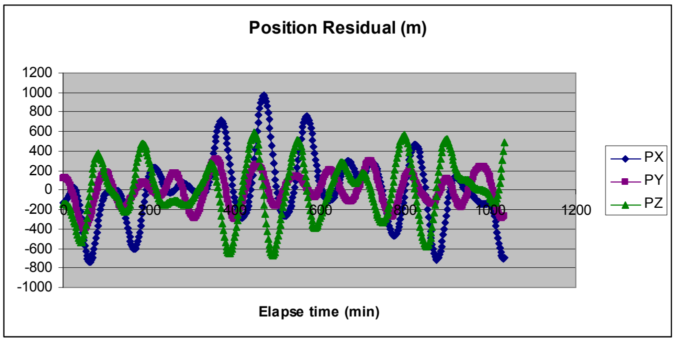

Fig. 2 Position residuals.

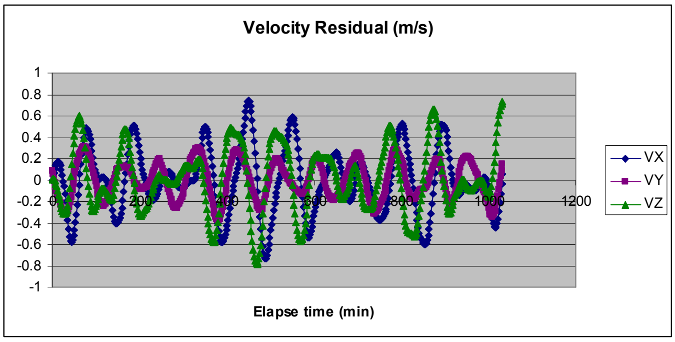

Fig. 3 Velocity residuals.

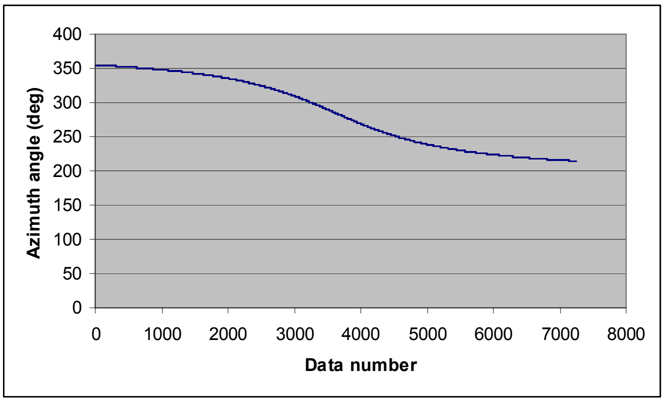

Fig. 4 Predicted azimuth angle profile.

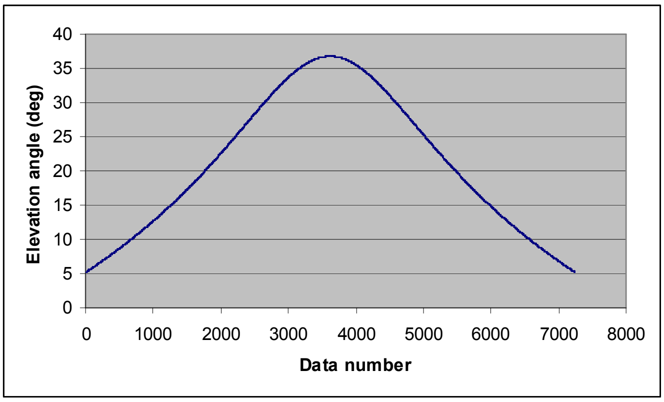

Fig. 5 Predicted elevation angle profile.

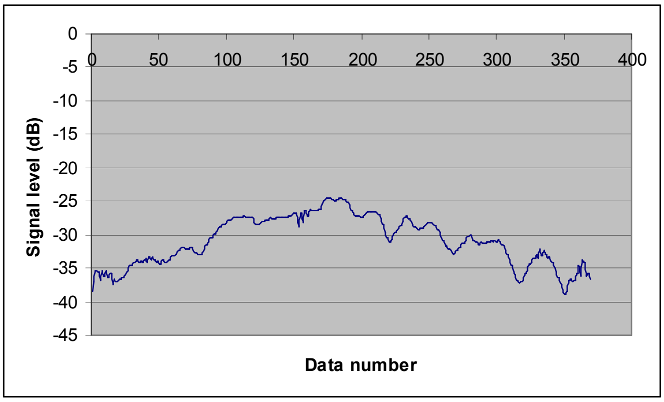

Fig. 6 Received signal strength.

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