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Development of Precise Ephemeris Generation Module for Thaichote Satellite Operations

Manop Aorpimai, Ponthep Navakitkanok

***Abstract*—**In this paper, the development of the ephemeris generation module used for the Thaichote satellite operations is presented. It is a vital part of the flight dynamics system, which comprises, the orbit determination, orbit propagation, event prediction and station-keeping maneouvre modules. In the generation of the spacecraft ephemeris data, the estimated orbital state vector from the orbit determination module is used as an initial condition. The equations of motion are then integrated forward in time to predict the satellite states. The higher geopotential harmonics, as well as other disturbing forces, are taken into account to resemble the environment in low-earth orbit. Using a highly accurate numerical integrator based on the Burlish-Stoer algorithm the ephemeris data can be generated for long-term predictions, by using a relatively small computation burden and short calculation time. Some events occurring during the prediction course that are related to the mission operations, such as the satellite's rise/set viewed from the ground station, Earth and Moon eclipses, the drift in groundtrack as well as the drift in the local solar time of the orbital plane are all detected and reported. When combined with other modules to form a flight dynamics system, this application is aimed to be applied for the Thaichote satellite and successive Thailand's Earth-observation missions.

***Keywords*—**Flight Dynamics System, Orbit Propagation, Satellite Ephemeris, Thailand's Earth Observation Satellite.

I. Introduction

THAICHOTE is Thailand's first commercial Earth-Observation Satellite (THEOS) [1]. It was launched into a Low-Earth Orbit (LEO) in October, 2008. In order to serve its main payload, a high-resolution multi-spectral Earth imaging system, the satellite was inserted into a Sun-Synchronous orbit with the local solar time of the descending node at 10 A.M., and a repeat-groundtrack condition of 26 days, 369 orbits has been assigned for exact revisit to the areas of interest on the ground. Frozen orbit is also preferable for altitude variation minimization. These specialist orbit conditions can be evaluated by incorporating the perturbations effects [2]. The required orbital configuration for the Thaichote mission is summarized in Table I.

The flight dynamics system (FDS) takes the responsibility for keeping the satellite's orbit at such requirements throughout its designed life-time. It also propagates the satellite position forwards in time and generates ephemeris data required for the mission operations. At its operational altitude in LEO, the spacecraft will experience some natural perturbation forces that can cause orbit configuration deviated from its predefined conditions. The prediction of those important events, therefore, is crucial for the operations of the satellite.

Manop Aorpimai is with the Thai-Paht Satellite Research Centre, Mahanakorn University of Technology, Nongchok, Bangkok, Thailand (phone: +66 2 9883655, fax: +66 2 9884040, email: manop@mut.ac.th).

Ponthep Navakitkanok is with the Orbit and Space Object Analysis Division, Geo-Informatics and Space Technology Development Agency (GISTDA), Ministry of Science and Technology, Thailand (phone: +66 38 490858, fax: +66 38 490858, email: pornthep@eoc.gistda.or.th).

TABLE I
THAICHOTE MISSION ORBITAL CONFIGURATION

Orbital Parameters	Value
Altitude	822 km
Inclination	98.76 deg.
Frozen eccentricity	0.001146
Mean Motion	14+5/26 revolutions/day
LST of Descending Node	10.00 A.M. +/- 2 min.
Groundtrack revisit	26 days (369 revolutions)
Groundtrack Maintenance	+/- 40 km w.r.t. the reference longitude

In this paper, we describe an in-house development of the Thaichote's ephemeris generation module, which comprises 2 important parts, the orbit propagation and event prediction. The mathematical models and algorithm used in the calculation are explained and some prediction results are shown and discussed.

II. Flight Dynamics System

The flight dynamics system comprises 4 main modules, i.e. orbit determination (OD), orbit prediction (OP), event prediction (EP), and station-keeping maneouvre (SK). The OD module retrieves navigation data from the on-board GPS receiver on a daily basis. The satellite orbital states, i.e. position and velocity vectors, are estimated using the differential correction algorithm, where the residuals between the observed and the estimated states are minimized using a Weighted Least-Squares cost function. Some solved-for parameter, such as drag coefficient, is also included in the estimated state vector for the evaluation of the orbital decay rate. By using the converged estimated solution from OD, the OP module propagates the satellite's states advance in time, as well as converts them into different coordinates appropriate for each sub-module application. High-fidelity force models, as well as a precise numerical integrator play the crucial role in this orbit propagation process.

Important events occurring throughout the prediction period, such as equator crossing, satellite rise/set with respect to the assigned ground station, Earth and Moon eclipses, drift in groundtrack and the local solar time of the orbital nodes, are predicted in the EP module. This information is useful for the satellite operations and planning, such as the transponder activation, antenna pointing for data transmission, payload operation and orbit maneouvre planning.

The SK module handles the calculation of the required delta-v vector, and execution time and firing duration. It also accounts for the propellant expenditure and fuel remaining. Fig. 1 represents the functional architecture and interface of the FDS.

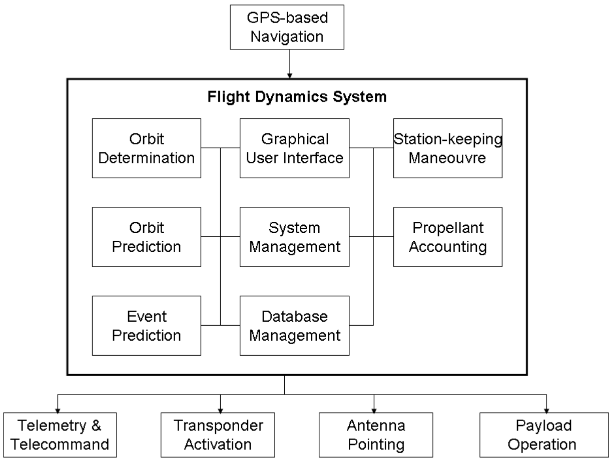

Fig. 1 The flight dynamics system structure

III. Orbit Propagation

The generation of the spacecraft ephemeris data relies upon the state solutions of the equations of motion, where the estimated states from the OD module are used as the initial conditions. The accuracy of the solutions depends on the numerical integration method, as well as the mathematical modeling of the perturbation forces.

The Bulirch-Stoer algorithm [3] has been applied for our ephemeris generation. It incorporates the Richardson extrapolation into the algorithm. This extrapolative method is considered to be the fastest and most accurate, and it is particularly suitable for our near-circular orbit where the dynamics topology is quite uniform throughout the integration course.

At the operational altitude of about 822km, the dominant force acting on the spacecraft is contributed by the non-spherical Earth. The gravity model used in our FDS is optional between WGS84, GEM10B, and JGM-3. Degree and order of the gravitational harmonics can be included up to a higher number, however, a truncated version with 36×36 terms is generally sufficient for the mission operations [4].

The gravitational attraction from the Sun and the Moon can cause long-term orbital plane drift, and they are modeled as point-mass objects. Their position vector with respect to an Earth equatorial plane coordinate system is important not only for the acceleration calculation, but also for the prediction of the eclipses. The Jet Propulsion Laboratory Development Ephemeris (JPL DE405) model [5] has been adopted for 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 from data on solar activity index, $F_{10.7}$, and from the geomagnetic index, K_p. 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.

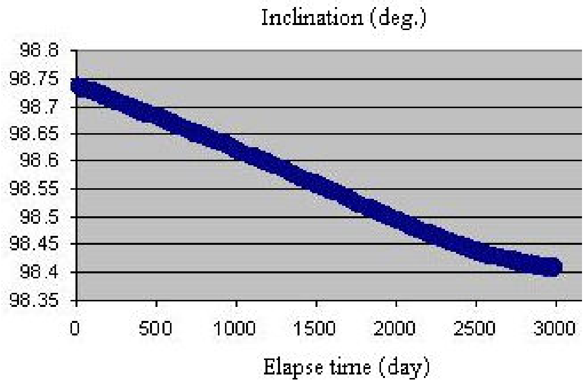

Fig. 2 Long-term inclination drift caused by the 3rd-body attraction

The combined effects from all perturbations can be integrated both for short-term and long-term applications. Fig. 2 shows an example of a 10-years evolution of the Thaichote's orbital inclination. It is clearly seen that orbit maneouvre is required to be executed periodically, in order to keep the inclination at its assigned value.

IV. Event Prediction

Events that are important for satellite operations throughout the orbit propagation course are detected in the EP module, both in a day-by-day and event-by-event basis. Some relevant parameters are also calculated. They include the equator crossing position and time, satellite rise/set time and its position in the topocentric coordinates, which is useful for transponder activation and antenna pointing routines. The prediction of eclipses both from the Earth and the Moon will help in imaging planning and electrical power management.

Some station-keeping-related parameters generally require long-term prediction. The vital parameters include groundtrack error, local solar time of the equator crossing nodes and the biased eccentricity vector from the mission's frozen condition. Fig. 3 shows a 6-months prediction of the groundtrack error with the control band of ± 40 km marked. Short-term simulation may be also required for verification due to the uncertainty of the atmospheric drag model. The drift in local solar time shown in Fig. 4 is caused mainly by the change in orbital inclination, perturbed by the Sun's attraction as seen in Fig. 2. The local solar time control window has been set within ± 2 minutes.

The bias eccentricity vector can be found by converting from the osculating to mean orbital elements. A small variation around the frozen condition as shown in Fig. 5 confirms the minimal change in altitude profile over the ground as shown in Fig. 6.

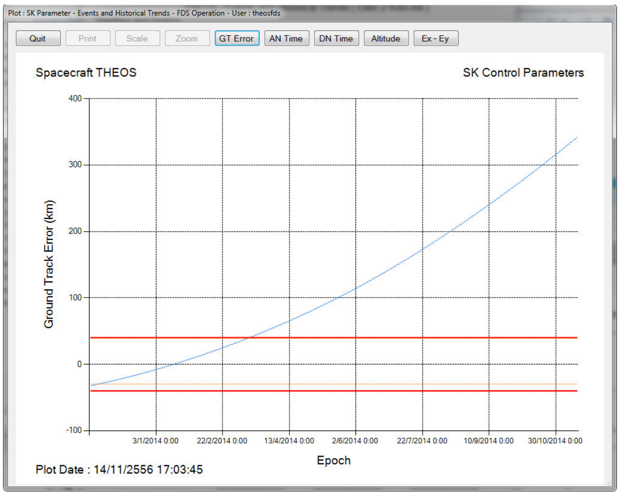

Fig. 3 Groundtrack error prediction

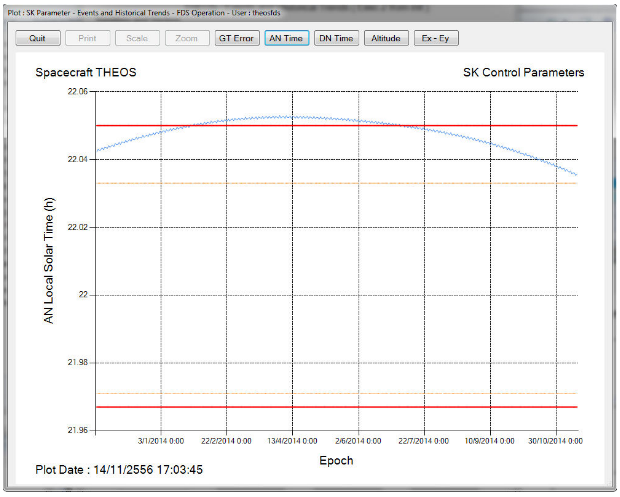

Fig. 4 Drift in local solar time of the ascending node prediction

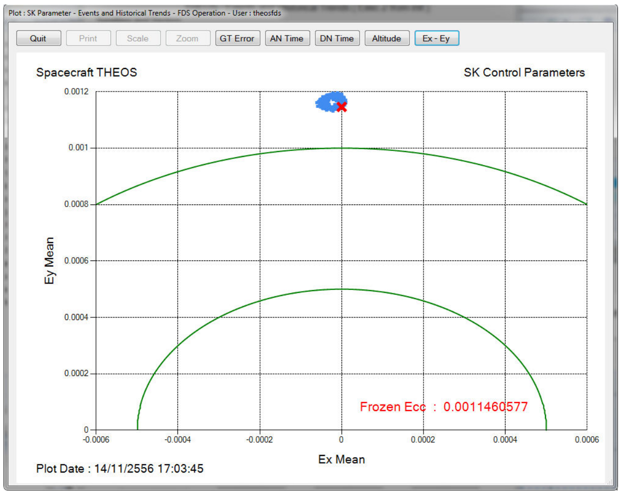

Fig. 5 Evolution of eccentricity vector around the frozen condition

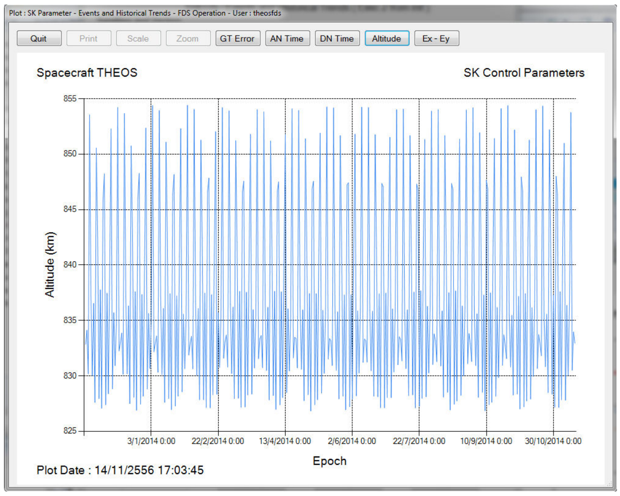

Fig. 6 Prediction of orbital altitude variation

V. Conclusion

We have presented the development of the ephemeris generation module of the Thaichote satellite. 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, and some other non-conservative forces, such as atmospheric drag and solar radiation pressure are taken into account. Using an accurate integrator based on the Burlish-Stoer algorithm the ephemeris data can be generated for years in advance by using a small computation burden and short calculation time. The event prediction and station-keeping maneuver planning can directly employ this ephemeris data by rearranging them into formats that are convenient for the mission operations.

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