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# Automated Flight Dynamics System for Thaichote Satellite

Ponthep Navakitkanok and Manop Aorpimai

*Abstract*— In this paper, we present the development of the flight dynamics system used for the operation of the Thaichote satellite. The system comprises the orbit determination, orbit propagation, event prediction and station-keeping maneouvre modules. The ground-based orbit determination uses the spacecraft's navigation data retrieved from the onboard GPS receiver on a daily basis. Using the estimated orbital state vector from the orbit determination module is used as an initial condition, the equations of motion are 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 where the satellite is operating. Using a highly accurate numerical integrator based on the Burlish-Stoer algorithm the ephemeris data can be generated for long-term predictions. Some events occurring during the prediction course that are related to the mission operations are detected and reported. This information is necessary for the station-keeping maneouvre planning module. The automated flight dynamics system is proposed to replace the traditional operation, where a number of dedicated personnel are required for handling the tasks. Via the user-friendly graphical user interface, the spacecraft operations require only initial parameter setup.

*Keywords*— Automated Satellite Operations, Flight Dynamics System, Thailand's Earth Observation Satellite

## 1. 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 1.

The flight dynamics system 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. The vital routine operations comprise the orbit determination (OD), orbit prediction (OP), event prediction (EP), and station-keeping maneouvre (SK).

In this paper, we describe an in-house development of the Thaichote's flight dynamics system. An automatic flight dynamics operations system is also proposed. The man-hour resource, human error and operational cost are expected to be reduced from this proposed automation system.

## 2. FLIGHT DYNAMICS SYSTEM

Flight Dynamics System (FDS) is a computer based software system. The FDS for the Thaichote satellite mission, as shown in figure 1, includes satellite OD, OP, EP and SK. The FDS is operated by the system management and database management. The FDS is normally supported by graphical user interface for easy operation, regardless of automatic flight dynamics operations.

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 |

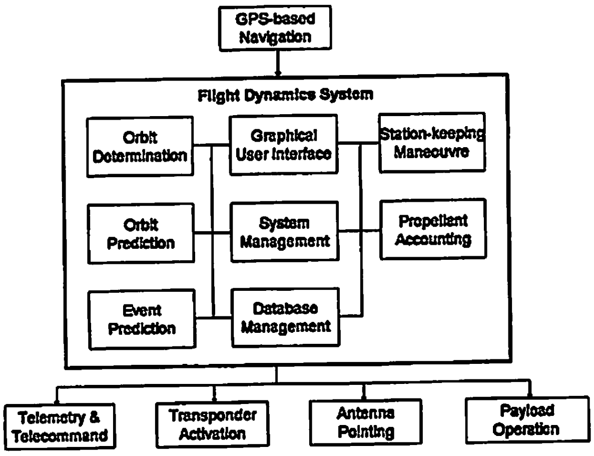

Fig. 1 The Thaichote Flight Dynamics System.

### 2.1 Obit Determination

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 [3], 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

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estimated state vector for the evaluation of the orbital decay rate. Figure 2 shows a typical residual profile of the estimated position vectors at each observation epochs.

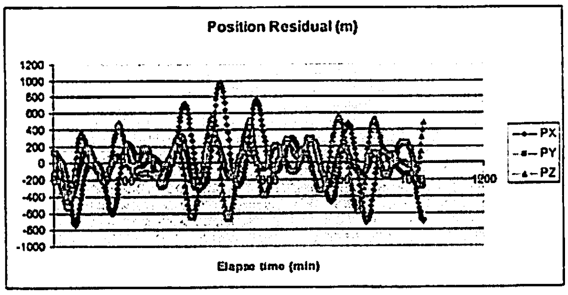

Fig. 2 Position residual of the estimated trajectory

### 2.2 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 [4] 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 822 km, the dominant force acting on the spacecraft is contributed by the non-spherical Earth. The geopotential function, $U$, can be described using spherical harmonics function as

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

where $\mu$ 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 $r$, $\phi_{sat}$, $\lambda_{sat}$ are radius, latitude and longitude of the satellite's position vector, respectively.

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 .

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:

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

where $\ddot{\bar{r}}_{\oplus sat}$ 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 $\bar{r}_{\oplus 3}$ and $\bar{r}_{sat3}$, respectively.

The third-body object's 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 [6] 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 [7] has been adopted for the modeling of such effect. It computes the atmosphere density ($\rho$) from data on solar activity index, $F_{10.7}$, and from the geomagnetic index, $K_p$. The perturbing force is calculated from

$$a_d = -\frac{1}{2}\rho C_D \frac{A_D}{m} v_r \bar{v}_r \tag{3}$$

where, $\bar{v}_r$ 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

$$a_{rp} = \nu P_S AU^2 C_R \frac{A_R}{m}\frac{\bar{r} - \bar{r}_S}{\left\|\bar{r} - \bar{r}_S\right\|^3} \tag{4}$$

where PS 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 v, that measures the degree of Sun's occultation by a body like the Earth or the Moon. The combined effects from all perturbations can be integrated both for short-term and long-term applications.

### 2.3 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.

### 2.4 Station-Keeping Maneouvre Planning

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. 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 is shown in Fig.5.

## 3. AUTOMATED FLIGHT DYNAMIC OPERATIONS

Automation of the satellite mission control is important to modern satellite operation environment in two reasons. One is the saving of the man power in the satellite mission control and the other is to prevent satellite operator from mistakes in manual operation process. Normally, automation of the satellite mission operation starts from daily routine operations. The automation makes the satellite operator focus on more important and urgent mission operation. The Flight Dynamics Automation (FDA) is the driver of the related functions in FDS. Fig. 5 illustrates a high level software functional architecture of the FDA and FDS.

### 3.1 Automation of Orbit Determination

OD provides the estimated position and velocity of the satellite based on the measurement data. A batch weighted least square estimation is normally performed everyday using the GPS data. A priori orbit state from orbit stack data file is used for starting of the orbit estimation. Automation of OD focuses on the input and output file handling and execution of the OD program. Automation scenario of OD is as follows.

- At predefined time, check if there is a new tracking data file in a designated directory (including data transmission failure check)
- Generate a new input data file for orbit determination program
- Execute orbit determination program
- Evaluate the orbit determination results (including failure check)
- Generate a report file for orbit determination
- Update orbit stack data
- Move the old track data file to the specified directory according to the OD execution result (Success/Fail/Update Success/ Update Fail)

### 3.2 Automation of Orbit Propagation

OP provides the future position and velocity of the satellite based on the orbit determination results. The OP is performed using the numerical integration of the satellite orbit equation. Automation scenario of OP is as follows.

- At predefined time, check if there is a new orbit stack data in a designated directory
- Generate a new input data file for orbit prediction program
- Execute orbit prediction program
- Evaluate the orbit prediction results (including failure check)
- Generate a report file for orbit prediction

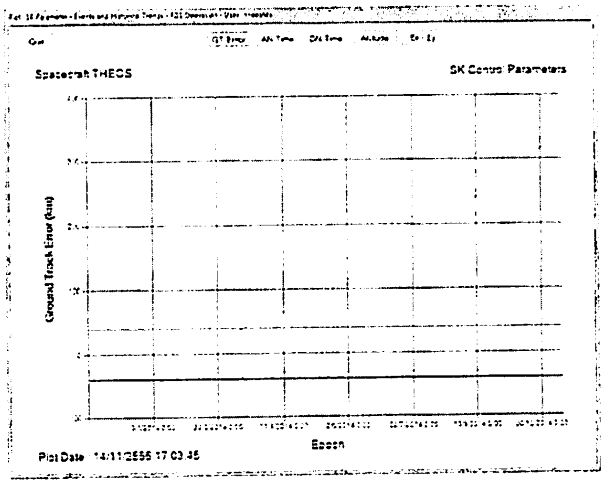

Fig. 3 Groundtrack error prediction.

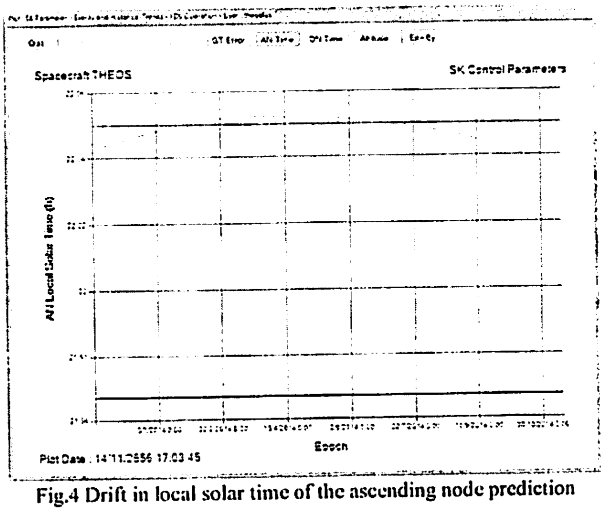

Fig.4 Drift in local solar time of the ascending node prediction

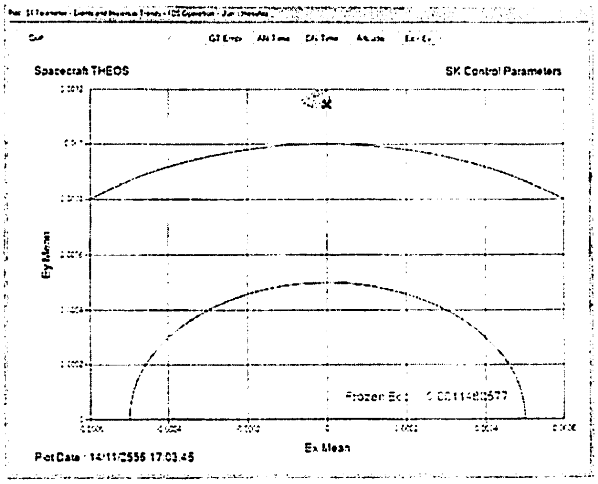

Fig. 5 Evolution of eccentricity vector around the frozen condition

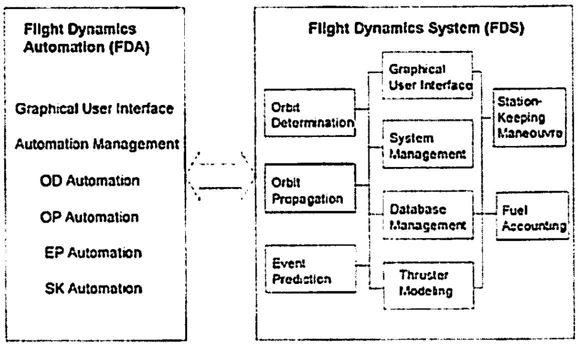

Fig. 6 Functional architecture of the FDA and FDS

### 3.3 Automation of Event Prediction

EP provides Sun eclipse time due to the Earth and the Moon, satellite sensor intrusion time, and station-keeping box boundary of the spacecraft. The event prediction also provides the various orbital events such as nodal crossing time and apsidal passing time. Automation scenario of EP is as follows.

- At predefined time, check periodically if there is a new event prediction data in a designated directory
- Generate a new input data file for event prediction
- Execute event prediction program if necessary
- Evaluate the event prediction results (including failure check)
- Generate a report file for event prediction
- File transfer to related subsystems for satellite operations

### 3.4 Automation of Fuel Accounting

Fuel Accounting (FA) provides the capabilities for analysis of remained fuel mass inside the spacecraft fuel tank by telemetry data. Fuel accounting consists of two programs. Pressure, Volume, and Temperature method using temperature and pressure of the fuel tank transmitted from telemetry data and Thruster-On-Time method that calculates remaining fuel accounts using pulse of the thruster firing from telemetry data. Automation scenario of FA is as follows.

- At predefined time, check if there is a new telemetry data in a designated directory (including failure check)
- Generate a new input data file for fuel accounting program
- Execute fuel accounting program
- Evaluate the fuel accounting results (including failure check)
- Generate a report file for fuel accounting
- Update the spacecraft database

## 4. CONCLUSIONS

We have presented the development of the automatic flight dynamics operations for the Thaichote satellite. The automated flight dynamics system is proposed to replace the traditional operation, where a number of dedicated personnel are required for handling the tasks. Via the user-friendly graphical user interface, the spacecraft operations will require only initial parameter setup. The proposed system can perform algorithm executions, flight dynamics data archiving, file formatting and file distribution to different centers without any intervention from operators. The man-hour resource, human error and operational cost are expected to be reduced from this proposed automation system.

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