In Orbit Performance of LEO Satellite Electrical Power Subsystem - SW Package for Modelling and Simulation Based on MatLab.7 GUI

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Abstract: This paper discusses a development software tool for modelling and simulation of satellite electrical power subsystem (SEPSMS) in orbit operation analysis. The development of the components database to facilitate the modelling and simulation of the satellite EPS for remote sensing low earth orbit (LEO) application are introduced and some of them are deduced. In particular, the model of orbit estimation and solar array illumination are introduced, the specific output power of solar array based on GaAs solar cells single junction is presented also. The orbit perturbations and satellite separation from launch vehicle errors are included in the analysis of satellite EPS operation during the lifetime period of satellite. The model of off-nominal operation of the solar array is included in the EPS analysis during the lifetime period. Also the degradation parameters of the different EPS components are considered in the components models. The present work attempts to overcome certain inadequacies of contemporary simulation applications with respect to SEPSMS, by developing mathematical or empirical system modelling schemes for EPS components. These schemes are then integrated within a state-of-the-art simulation environment, so that they can be employed in practice [1]. For the purposes of the present work, it was decided to work with a state-of-the-art simulation program based on MatLab.7 Graphical User Interface (GUI). The EPS system parameters is presented in three figures screen grouped as; window of orbit and solar illumination, orbit shadowing period and local time of descending (LTD) , window of solar array output current, load profile and storage battery capacity all of them as a function of time. The third window shows the satellite bus voltage and static pattern of storage battery temperature. The result of the introduced tool is verified by another one for a similar satellite issued by an experienced agency in this field [2], the convergence was very high.

Key Words: Satellite power subsystem, LEO, Modelling and Simulation, Sunsynchronous orbit Dynamics, MatLab.7 GUI, Real-Time monitoring.

1. Introduction

The EPS power characteristics are determined by both the parameters of the system itself and by the satellite orbit. The EPS includes solar arrays, storage batteries and a management and control device that estimates the appropriate mode of system operation. [3,4]. One of the most significant challenges facing a designer engineer during the development of subsystem destined for space is the difficulty in testing prototypes under space environment conditions [5]. Modelling and simulation tools are more usefully in these cases. Most satellite simulators can be divided into three groups according to their usage; development, integration and verification, and operations. A subsystem simulator, such as a solar array simulator or a battery simulator, is a typical development simulator. This kind of simulator is used to analyze the characteristics of the subsystem or to evaluate the design parameters. Integration and verification simulators are used for hardware-hardware or hardware-software interface validation and integration and test procedures

validation. Satellite operations simulators are used for ground systems checkout, command and telemetry validation, and satellite operation team training. Satellite operations simulators may be divided further into static and dynamic simulators. The former is designed to process stored events and the latter, for a simulator that models the spacecraft's dynamics as realistically as possible. Some dynamic simulators include hardware in the simulator loop. The software models running on such computing devices can simulate the functions of most hardware subsystems with relatively high precision. A simulator using software models has more flexibility in developing simulation models and requires less expense than one that involving hardware [6].

2. Outline of the Previous Work

Some EPS components especially storage battery performance in a particular device is still determined by testing. Advances in computer simulation of battery performance now allow good estimates of battery performance to be made, so devices can be designed in software. The simulation allows a rapid assessment of

how the system will perform and can reduce or eliminate the need for testing [7,8]. A Spacecraft Simulator (SPASIM) to simulate the functions and resources of a spacecraft to quickly perform conceptual design (Phase A) trade-off and sensitivity analyses and uncover any operational bottlenecks during any part of the mission is introduced by [9]. The primary function of the SPASIM software is to create a virtual environment to simulate a spacecraft. By reviewing the similar work in the field of modeling and simulation of the satellite EPS, the advantages, time saving and economical solution could be found, so this tool is introduced.

3. Simulation Method

Unlike [10, 11], SEPSMS will perform a real-time series simulations of power system performance for time steps typically between 150ms to 10 minutes or 1 hour to 10 hours (two selectable ranges). The model will use an energy balance approach within each time step such that the sum of the outputs of all the energy sources (solar array output) must equal the sum delivered to all the energy sinks (typically storage input, the load, and losses). A deterministic method will be used within each time step to account for fluctuations in the solar energy and in the load which can have a significant impact on performance. Within each time step of a simulation, system performance will be calculated.

4. Modelling of Satellite EPS

To estimate a solar array illumination, unlike ballistic calculations of satellite orbits and routes, a simplified model of satellite orbital motion is used, which operates with six orbital elements [12, 13]. For Satellite operational in a circular orbits, the orbital illumination parameters, and hence power subsystem output energy is characterizing by an angular position of its plane about the Sun (the angle between a normal to an orbital plane and Sun direction). In the following subsections, this angle matrix will be deduced.

4.1 Keplerian Orbits Parameters in Space

For earth-orbiting spacecraft, it is common to define an inertial coordinate system (X,Y &Z). The Z axis is the axis of rotation of the earth in a positive direction, which intersects the celestial sphere at the celestial pole. The X-Y plane of this coordinate system is taken as the equatorial plane of the earth, which is perpendicular to the earth's axis of rotation Z. In the geocentric coordinate system, the earth's equatorial plane is inclined to the ecliptic plane, which is the plane of the earth orbit around the sun, by an angle of 23.5^0 (Fig. 1). Both planes intersect along a line that

is quasi-inertial in space with respect to the stars The X axis of our inertial system coincides with this line, which is called the vernal equinox vector. The third axis Y completes an orthogonal right-handed system.
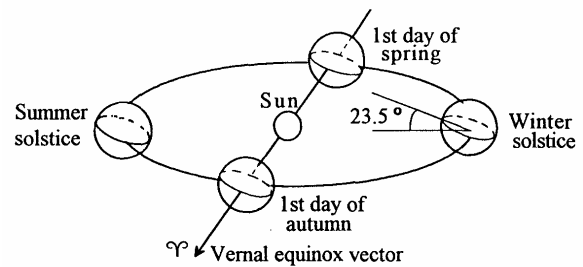

Fig. 1, Plane of the earth orbit around the Sun

Any Keplerian orbit can be completely described by six orbital elements, two to describe the size and shape, three to describe the orientation and one to describe the satellite location [12, 14]. In this model, the classical orbital elements will be used and it will be assumed that we are referring to an earth-orbiting satellite. These elements are; semimajor axis (a), eccentricity (e), mean anomaly (M). Another three additional parameters necessary to position an orbit in space, these parameters are [14]:

i The inclination angle of the orbit,

Ω The angle in the equator plane that separates the node line from the X_r axis (vernal equinox direction) is called the right ascension of ascending node, and

ω The argument of perigee, angle between r_p and the node line.

4.2 Model of Inclination Angle Calculation and Optimization

The sun synchronous orbit is the orbit that precesses very nearly 360° per year. Such orbit pass over the same part of the earth at roughly the same time each day, making communication and various forms of data collection convenient. For circular orbits the precession is given by the following formula:

$$\overset{o}{\Omega} = -9.95 \left(\frac{R_{eq}}{r} \right)^{3.5} * \cos i \ (^o/day). \qquad (1)$$

Where:

$\overset{o}{\Omega}$ is the orbital precession in $(^o/day)$

R_{eq} is mean equatorial radius of Earth, 6378 km,

r is the satellite-geocenter distance in km

i is the orbit inclination,

To obtain an orbital precession of 360° (sun synchronous orbit requirements), a precession rate of 0.986 ^{o}/day (360/365.25) is needed. Substituting this value in the precession formula (1)and by knowing the orbit altitude, the sun synchronous orbit

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inclination that is can be calculated from the following formula:

$$i = \arccos\left[-0.09910 * \left(\frac{r}{R_{eq}}\right)^{3.5}\right] \text{ (deg).} \quad (2)$$

The major perturbation force which gradually destroys the sun-synchronous property is the deep resonance with the sun's motion. The resulting drift in local time at ascending node may be as large as 35 minutes after three years. This undesirable perturbation can be minimized by introducing a small bias in the initial inclination without having to perform station keeping maneuvers [15].

4.3 Modeling of Solar Array Illumination
The six parameters are convenient for defining an orbit in space in the inertial system defined by its three axes X_Υ, Y, and Z, as shown in Fig. **2**. However, it can, also be useful to express the location of a moving body in other parameters, such as Cartesian or polar.

Many times of the three-dimensional transformation can be performed as necessary to achieve a desired overall transformation in space around different axes. The subsequent transformations from inertial coordinates X, Y, and Z to orbit coordinates P, Q, and W as defined in Fig. **2**. The orbit plan system is transformed by the three angles; $(\omega, i \ \& \ \Omega)$. It should be noted that; (P is a unit vector directed from the center of the orbit to the perigee, W is a unit vector along the momentum axis of the orbit (normal to orbit plan), h = r X v, and Q = P x W).

$$\begin{pmatrix} P \\ Q \\ W \end{pmatrix} = [A_z(\omega)][A_x(i)][A_z(\Omega)] \begin{pmatrix} X \\ Y \\ Z \end{pmatrix} \quad (3)$$

Where:

$[A_z(\omega)]$ Transformation matrix about axis z by "ω".

$[A_x(i)]$ Transformation matrix about axis x by "i".

$[A_z(\Omega)]$ Transformation matrix about axis z by "Ω".

Unlike P, Q and W transformation, another effective transformation based on the three angles; $(\omega+\theta)$, $(i \ \& \ \Omega)$ can be processed, transformation from inertial plan X, Y and Z to local coordinate system (orbital) R, S and W, where R is a unit vector along radius vector r, W the same in PQW (normal to orbit plan or a unit vector along the momentum vector h) and S completes the right orthogonal system. This transformation can be proceeds as follows:

$$\begin{pmatrix} R \\ S \\ W \end{pmatrix} = \begin{pmatrix} C(\omega+\theta) & S(\omega+\theta) & 0 \\ -S(\omega+\theta) & C(\omega+\theta) & 0 \\ 0 & 0 & 1 \end{pmatrix}\begin{pmatrix} 1 & 0 & 0 \\ 0 & Ci & Si \\ 0 & -Si & Ci \end{pmatrix}\begin{pmatrix} C\Omega & S\Omega & 0 \\ -S\Omega & C\Omega & 0 \\ 0 & 0 & 1 \end{pmatrix}\begin{pmatrix} X \\ Y \\ Z \end{pmatrix} \quad (4)$$

Now we have a transformation from inertial to local coordinate systems, in our case, LEO satellite, we are looking for the angle between the normal to orbit plane (vector W) and sun direction that is the major parameter in the estimation of the solar array illumination coefficient. From equation 4, the matrix of the vector W can be deduced as follows:

$$(W) = [0 \ 0 \ 1]\begin{pmatrix} 1 & 0 & 0 \\ 0 & Ci & Si \\ 0 & -Si & Ci \end{pmatrix}\begin{pmatrix} C\Omega & S\Omega & 0 \\ -S\Omega & C\Omega & 0 \\ 0 & 0 & 1 \end{pmatrix}\begin{pmatrix} X \\ Y \\ Z \end{pmatrix} \quad (5)$$

Then,

$$(W) = \begin{bmatrix} Si \ S\Omega \\ -Si \ C\Omega \\ Ci \end{bmatrix}\begin{pmatrix} X \\ Y \\ Z \end{pmatrix} \quad (6)$$

Fig. 3 shows the rotation of the ecliptic plane on the equatorial plane by the angle 23.5°. The angle (γ) of the vector of Sun $(\dot S)$ represents the solar longitude from vernal equinox.

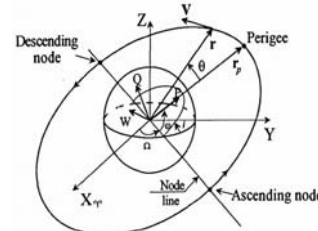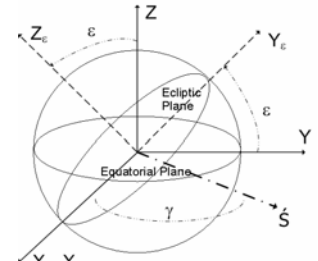
Fig. 2, Coordinate Systems Vectors.

Fig. 3, Ecliptic plane and Equatorial plane

The transformation from equatorial plan (X,Y and Z of inertial coordinate) to ecliptic plan $(X_\varepsilon, Y_\varepsilon, \& Z_\varepsilon)$ assuming the rotation around X axis, can be expressed as:

$$\begin{bmatrix} X\varepsilon \\ Y\varepsilon \\ Z\varepsilon \end{bmatrix} = \begin{pmatrix} 1 & 0 & 0 \\ 0 & C\varepsilon & S\varepsilon \\ 0 & -S\varepsilon & C\varepsilon \end{pmatrix}\begin{bmatrix} X \\ Y \\ Z \end{bmatrix} \quad (7)$$

The vector of Sun in ecliptic plan can expressed as:

$$[\dot S_\varepsilon] = \begin{pmatrix} C\gamma \\ S\gamma \\ 0 \end{pmatrix}\begin{bmatrix} X_\varepsilon \\ Y_\varepsilon \\ Z_\varepsilon \end{bmatrix} \quad (8)$$

From (3), (4), the vector of Sun in inertial coordinate can be expressed as:

$$[\dot S] = \begin{pmatrix} 1 & 0 & 0 \\ 0 & C\varepsilon & S\varepsilon \\ 0 & -S\varepsilon & C\varepsilon \end{pmatrix}\begin{pmatrix} C\gamma \\ S\gamma \\ 0 \end{pmatrix}\begin{bmatrix} X \\ Y \\ Z \end{bmatrix}, \quad (9),$$

Then,

$$[\dot S] = \begin{pmatrix} C\gamma \\ C\varepsilon \ S\gamma \\ -S\varepsilon \ S\gamma \end{pmatrix}\begin{bmatrix} X \\ Y \\ Z \end{bmatrix} \quad (10)$$

Fig. 4 shows the angle between the normal to orbit plane (Y_o) and the Sun direction $(\dot S)$ named (ν).

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The vector of normal to orbit plane is defined in equation 6 referred to inertial coordinate and the vector of sun is defined in equation 10 also referred to inertial coordinate, from these two vectors, (ν) angle can be expressed as follows:

$$\cos\nu = \frac{\overline{W}\,\overline{S}}{\|W\|.\|S\|} \qquad (11),$$

Then,

$$\cos\nu = \begin{bmatrix} Sin(i)\,Sin(\Omega)\,Cos(\gamma) \\ -Sin(i)\,Cos(\Omega)\,Cos(\varepsilon)\,Sin(\gamma) \\ +Cos(i)\,Sin(\varepsilon)\,Sin(\gamma) \end{bmatrix} \qquad (12)$$

4.4 Modeling of SA Specific Output Power

As applied to the earth small-sized satellites, on which as a rule, the SA are used stationary with respect to its body with an earth orientation and stabilized in the orbital coordinate system, the illumination change law of SA working surface is described by the following equation:

$$E_{SA} = E_s * \cos\alpha(\tau) \qquad (13)$$

Where:

$$\cos\alpha_j(\tau) = \begin{Bmatrix} \cos\nu(t)*\sin\theta_j*\sin\varphi_j \\ +\sin\nu(t)*\cos\theta_j*\cos\beta(\tau) \\ -\sin\nu(t)*\sin\theta_j*\cos\varphi_j*\sin\beta(\tau) \end{Bmatrix}, \qquad (14)$$

- E_s AM0 solar radiation intensity, 1360 W/m²;
- θ_j, φ_j Fixed setting angles of SA in altitude and in azimuth in the related coordinate system,
- $\beta(\tau)$ Orbital angle counted-off from an orbital zenith point to Satellite orbital radius vector.
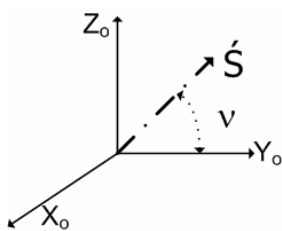- $\beta(\tau) = 2*\pi*\tau/T_o$, T_o is a period of one complete orbit shown in Fig. 5.

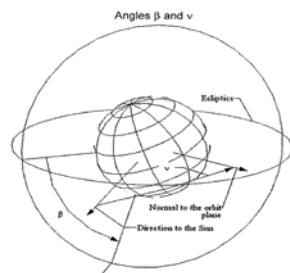Fig. 4, Angle between normal to orbit plane and Sun direction

Fig. 5, Orbit and SA illumination angles ν & β.

When $\cos\alpha_j(\tau) \le 0$, a solar radiation flux is not incident on SA working surface (earth shadowing or SA self-shadowing). For sun-synchronous orbit with fixed solar array panels at 90° with altitude, by means ($\theta = 0$) and the flight angle φ takes any value:

$$E_{SA}(t) = E_s * \sin\nu(t) * \cos\beta(t) \text{ W/m}^2 \qquad (15)$$

4.5 Modeling of Load Profile

The load profile is a graphical representation of the power requirements of the satellite subsystems with time. In fact the load profile is not a periodic function since the peak session can be different from cycle to cycle based on the payload planning, but in our case a periodic load profile is used to check the EPS operation in worst case, by means an imaging session is assumed each orbit. Load profile consists of two parts; duty power and peak power:

Duty Power, it is the power consumed by the satellite OBE that are powered on all the time like OBC, ADCS.

Peak (Session) Power, it is the power consumption during the imaging session.

4.6 Modeling of Storage Battery

The battery is a sealed NiCd type, 22 series cells, and 8Ah each. The model of the storage battery presents the main parameters of the storage battery like; voltage, capacity and temperature as a function of orbital time. The electrical model with dynamics of voltage changes during charging/discharging cycles is used in this paper. A detailed model includes the electrical and thermal properties of NiCd Battery for Low Earth Orbit Satellite's is well described in special article.

5. SEPSMS SW Package User Manual

The SW window consists of three graphs and one control panel. Fig. 6 shows the GUI window of the proposed SW package. The 1st graph named orbit parameters graph presents the; (ν) angle in degree, local time of descending node in hours and orbit shadowing period in minutes. The 2nd graph named SA Output Power presents three parameters; SA output current I_{SA}, load current I_L and NiCd battery capacity in Ah. The 3rd graph shows the battery voltage and fixed pattern of off-line calculated average temperature of the battery during charge/discharge operation. The control panel consists of three sub-panels.

The 1st panel for the nominal orbital parameters, the 2nd panel for orbit perturbation (mainly altitude) and injection inaccuracies in altitude and inclination, SA off nominal situation is represented as percentage of its output while the 3rd panel for simulation control. It includes three push-buttons; simulate, clear and close, function of each described in its name. Additional checkbox named time step (TS-1(min)), checking of this box means that time will be in minutes and clears means that time will be in day for long term simulation.

Two sliders are stated in the right and left sides of control panel. The right one is used for easy

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change of orbit altitude while the left side one is used for simulation step control.

5.1 Features of the presented package

Unlike [10], from the beginning, one of primary goals was that SEPSMS should be a user-friendly model. Several features of the model are aimed at achieving that goal:

- Windows-based Graphical User Interface (GUI) specifically Software Package,
- The user will be able to make a choice of design data (variables) within individual windows for entering his proposed system data.

5.2 EPS Simulation in nominal & ideal orbit

In this case, only the nominal orbit parameters should be filled in its sub panel by the user data. Orbit altitude and inclination should be estimated by the user if the orbit is non-sun synchronous type. In case of Sunsynchronous orbit, just define the altitude, check the Sun_Syn checkbox, the inclination angle will be calculated and appears in its window automatically. The also kindly needs to estimate the life time (period of simulation he needs) and define the application LTD. Set point of Time step has a value of "1" of the time unit; minute or day. The user should be careful when he check the checkbox of time step, if the lifetime set by years, the program needs some minutes to out the simulation results, 1 minute 55 seconds for one year simulation with a 1 minute time step using a P4, 2.4 GHz and 512 MB RAM PC machine. The results for one year simulation with one minute and one day simulation are shown in Fig. 7 and Fig. 8. In Fig. 8, it could be seen that the output data in the graph of SA output and Battery charge/discharge is very condensate since the user could see the maximum and minimum peak for each parameter. But it is available for the user at any moment or instant to monitor any of EPS parameters for any certain period of time by using the figure toolbar in GUI proposed SW program.

5.3 EPS Simulation in nominal mode and perturbed orbit

In this case, the effect of orbit perturbation on the satellite EPS performance will be studied. Three parameters could affect the EPS performance are studied; perturbation in the orbit altitude due to gravity of the earth, injection inaccuracies in inclination and altitude. The injection inaccuracy of the altitude is assumed to be ±6km while the injection inaccuracy in the inclination is assumed to be ±0.045°. The results are shown in Fig. 9 to 12.

In Fig. 9, it could be seen that LTD is extremely affected by the orbital perturbation (mainly in altitude injection inaccuracy), since its values

changes from ~9:55–11:15. While in this diagram Fig. 10, it could be seen that LTD appears in its worst case since its peak to peak value changes from ~9:30 -11:30 minimum with negative inclination injection inaccuracy and maximum at positive inclination injection inaccuracy.

In Fig. 12, it could be seen in the graph of SA output power left top one that the load profile is switched to minimum duty power consumption (red colored curve). This decision is taken by the on board computer (OBC) to switch the satellite to standby mode because the battery voltage reaches the minimum level (24V). After five successful orbits, the battery capacity reach the nominal value and voltages rises up to ~33V, the OBC takes decision to operate the satellite pay load in nominal mode again.
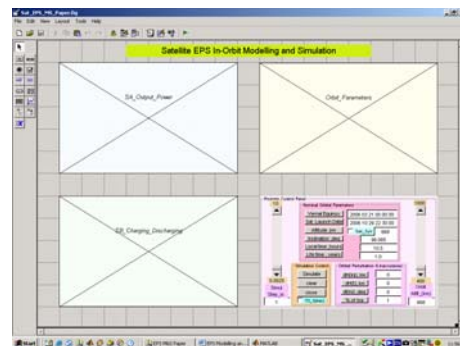

Fig. 6, the GUI window of the proposed SW package

6. Conclusion

The LEO EPS properties are studied and detailed described in this paper. A software tool for modelling and simulation of satellite electrical power subsystem (SEPSMS) in orbit operation is developed. The orbit dynamics matrices are deduced and the solar array illumination coefficient is estimated as a function of the satellite operation lifetime. The model of most of the EPS components is presented. The developed SW tool SEPSMS is a user friendly and could analyze the EPS operation at different operation modes; nominal and off-nominal. The results are introduced in many graphs; each one is illustrated in each caption or with a followed section when needed.

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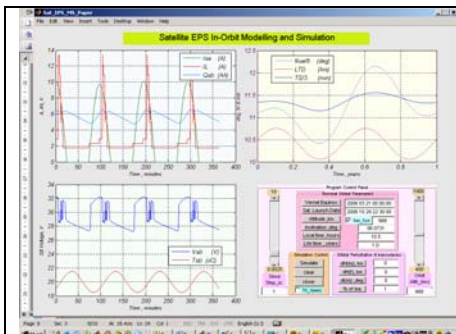
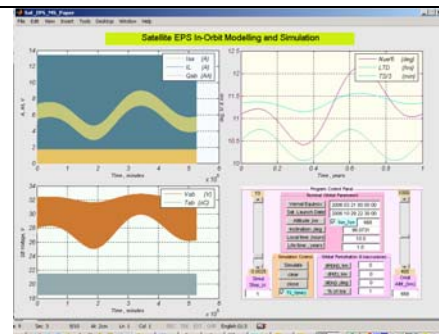Fig. 7, EPS 1-year simulation results at 1 day step.
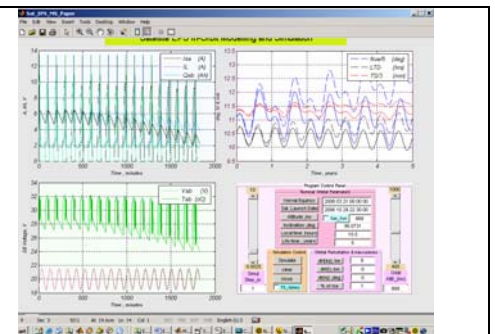

Fig. 8, year simulation results at 1-min step.

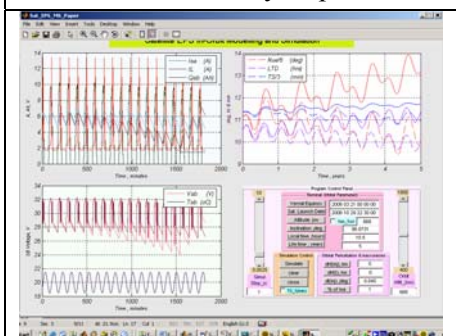Fig. 9, Altitude injection inaccuracy during lifetime, dash for -6km.

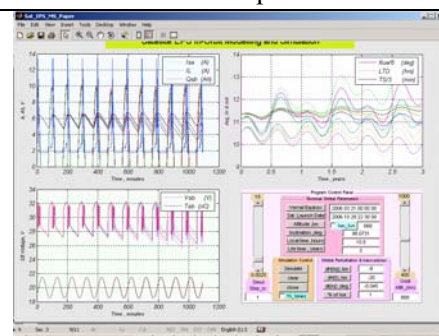Fig. 10, inclination injection inaccuracy, dash lines for -0.045^o and solid lines for +0.045^o

Fig. 11, A combined perturbation [+6km : +0.045^o, +6km : -0.045^o, -6km : +0.045^o and -6km :-0.045^o
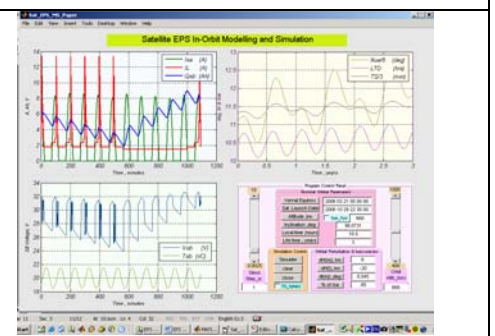

Fig. 12, shows the effect of SA off nominal operation (I_{SA} 85% of nominal)