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# Flight dynamic model for low earth orbit satellites

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**Abstract.** Low Earth Orbit satellite(LEO) missions have potentially attractive features such as low launch cost, and wide remote sensing applications.

There are three vital missions during the satellites life time including orbit transfer from the parking to the operating orbit, maintaining the operating orbit and de-orbiting mission to a disposal orbit at the end of the satellite lifetime according to the standard disposal orbit law. The mathematical model has been implemented to study various LEO mission orbits including the three vital missions.

The mathematical model evaluates the required performance parameters, velocity budget, number of manoeuvres, duration and consumed propellant mass for each mission.

The mathematical algorithm is carried out using Matlab/Simulink, the complete mathematical flight dynamic model has been verified through comparisons with the generated mission scenarios from Satellite Tool kit (STK), the graphical user interface is designed to display the model.

The carried out algorithm can be applied in satellite mission design to help predicting the parameters for each mission to minimize the risks and intensive tests. it can be considered as powerful tool to design and analysis LEO satellite missions

*Keywords:* Satellite dynamic, flight dynamic model, space mission, deorbiting, orbit transfer.

## 1. Introduction

Modeling and simulation with precise accuracy have become important research topics in the field of Satellite design. Matching the mathematical and simulation models with the real action behavior is highly essential for determined the total velocity budget required for satellite missions, since these missions are so expensive, operated far from the user and not possible to conduct testing in an actual environment. Then, very accurate and authentic simulation tool with high reliability simulating the actual space environment is highly required for space craft (SC) designers. Simulation tools are also essential in orbit control design; the available tools are too expensive and sometime cannot be used directly because it is not easy to customize these commercial simulators to suit specific needs. These simulation tools are not only useful for the development of control algorithms but they also provide a testing bed to conduct various kinds of testing. These tools also provide a simulation environment for the hardware in the loop testing to test the algorithms including physical hardware that has to be used in the real flight.

Various researches were published to study the methods of carrying out different types of orbit determination for different types of satellites for small and large scale size satellites using different



propagators and data types such as Doppler Shifts, GPS, GNSS, Magnetometer and Sun Sensor Data based on Least square patch method or Kalman filter estimator [1]–[6].

The main objective of this paper is to develop a powerful satellite simulation tool that can be used as a baseline to develop, implement and analyze the satellite dynamics, space missions, fuel budget, propulsion system (PS) selection and orbit characteristics during satellite primary design phase.

Further flexible MATLAB based structure of this simulator facilitates customized design which could suit a number of satellite missions. For a spacecraft engineer, numerous satellite simulation packages are currently available in the market Satellite Tool Kit (STK). Cost of these simulation environments is very high and also it is not suit to the specific requirements of a satellite mission [7].

The current paper is organized to brief describe Flight Dynamic Model (FDM) solving techniques, mathematical algorithm for each missions (orbit transfer, maintenance and deorbiting), mathematical model verification and results analysis [8].

## 2. FDM mathematical modelling

FDM model comprises three vital satellite transfer missions, the 1<sup>st</sup> mission is the transfer from the parking orbit (according LV insertion performance) to the working orbit (LEO), the 2<sup>nd</sup> mission is the maintaining of the working orbit during the satellite lifetime, while the 3<sup>rd</sup> mission is at the end of the satellite lifetime called de-orbiting mission to the standard disposal orbit based on space law issue [9]. Construction of the FDM requires implementing of each stated model separately for the precise analysis of each satellite missions. After completion of the separate models, they are connected to form the overall structure of FDM that can be used to obtain the preliminary satellite scenario data required during the execution of each mission. The detailed block diagram of FDM [10] is presented in Fig. (1). A mathematical model has been implemented to solve the LEO satellite missions at various options of altitudes, eccentricities, inclinations and propulsion systems used, to evaluate performance parameters for each mission as required velocity budget, number of manoeuvres, duration and consumed propellant mass. Each satellite mission model is carried out over three phases, built the mathematical model and executed by the Matlab/Simulink package, establishing the mission scenario by the STK software package and comparison between the built mathematical model in Matlab/Simulink and the STK scenario for mathematical modal evaluation and variation.

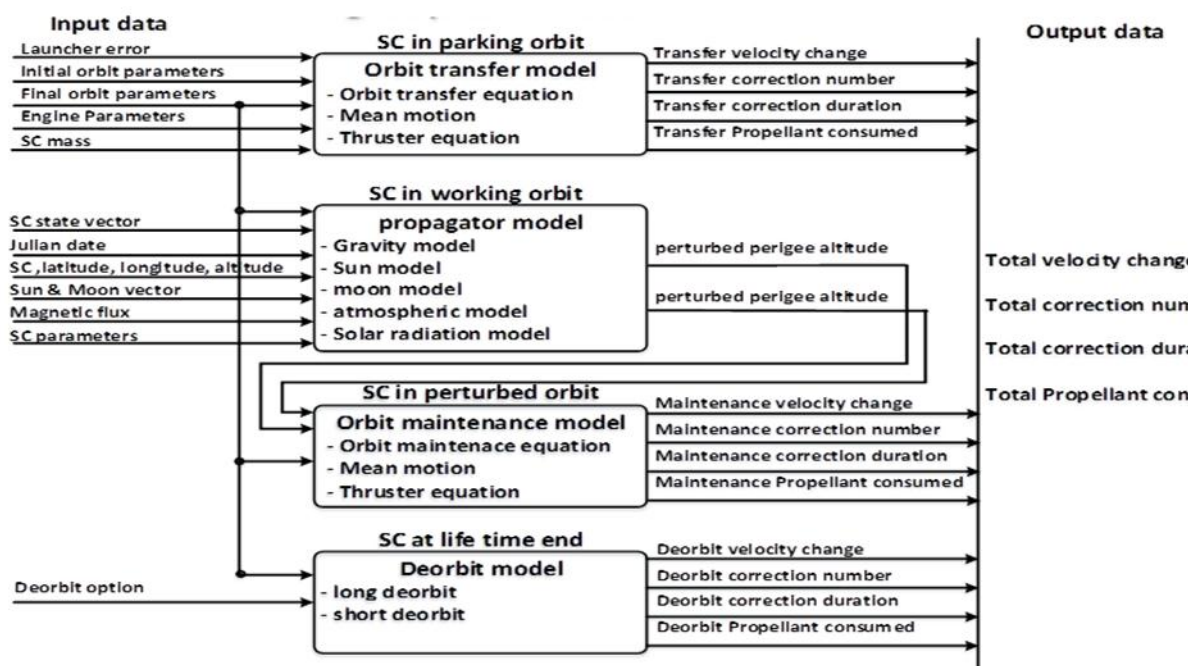


Figure 1. Flight dynamic model block diagram [10]

The mathematical FDM has been integrated and verified by the STK package at different altitudes (500 to 800 km). Finally, to easily clarify the different outputs for each satellite missions, the graphical user interface (GUI) is designed and connected with the program. The GUI front panel includes the input and output parameters, the output data are displayed by a 2 or 3-D visualization graph to helps the user to easily clarify the different outputs and better understanding of the mission scenario.

The FDM is written in Matlab\Simulink package, which is based on the understanding various fundamental equations related to satellite propagation, space dynamics, astrodynamics and satellite propulsion[11].

### 2.1. Mathematical model assumptions

The assumptions considered in the FDM are as follows[12]:

- Satellite is modeled as a point of mass object without change in orientation,
- Satellite has a constant drag coefficient for different satellite orientation,
- Simplified Earth gravitational model used is the WGS84 [13],
- The Sun and Moon are the only third body effect considered affecting the satellite,
- No forces act on the satellite except for gravitational and centrifugal forces acting along the line of centers,
- Constant thrust performance and propellant mass flow rate during manoeuver execution,
- Instantaneous velocity changes of the satellite under the effect of propulsive force,
- Thruster operation time is smaller than half the orbit period to minimize the gravitational losses.

### 2.2. Orbit transfer model

Orbit transfer model is based on the Hohmann transfer technique [11], the manoeuvres are executed at the perigee and apogee of the orbit. Execution of manoeuver at apogee leads to an increase of altitude of perigee and vice versa.

The mathematical algorithm considered the launch Vehicle (LV) insertion error (altitude, inclination), the manoeuver execution location and manoeuver durations.

The parking orbit classical orbital elements (COE) includes the perigee radius (rpp) and apogee radius (rpa) are defined in the following equations respectively[14] [15].

$$r_{pp(-)} = R_E + h_{pp} - h_{le} \quad (1)$$

$$r_{pa(-)} = R_E + h_{pa} - h_{le} \quad \text{or} \quad r_{pa(+)} = R_E + h_{pa} + h_{le} \quad (2)$$

where the subscript (+) corresponds to the increase altitude error of (LV) and vice versa for the (-) subscript,  $R_E$  is the Earth's radius (6378.14 km),  $h_{pp}$ ,  $h_{pa}$  are the perigee and apogee altitudes of parking orbit respectively and  $h_{le}$  is the altitude error of LV.

Final working orbit COE includes the perigee radius (r<sub>fp</sub>), apogee radius (r<sub>fa</sub>) as

$$r_{fp} = R_E + h_{fp} \quad (3)$$

$$r_{fa} = R_E + h_{fa} \quad (4)$$

where  $h_{fp}$ ,  $h_{fa}$  are perigee and apogee altitude of final orbit.

The semi major-axis,  $a$ , of the parking, transfer and final orbits (ap, at and af) respectively are defined as.

$$a_{p(-)} = \frac{r_{pp(-)} + r_{pa(-)}}{2} \quad \text{and} \quad a_{p(+)} = \frac{r_{pp(-)} + r_{pa(+)}}{2} \quad (5)$$

$$a_t = \frac{r_{pp(-)} + r_{fa}}{2} \quad (6)$$

$$a_f = \frac{r_{fp} + r_{fa}}{2} \quad (7)$$

The velocity at perigee and apogee of different satellite orbit phases (parking(p), transfer(t) and final (f)) are expressed as

$$v_{pp(-)} = \sqrt{\frac{\mu}{a_{p(-)}} * \frac{r_{pa(-)}}{r_{pp(-)}}} \quad \text{and} \quad v_{pp(+)} = \sqrt{\frac{\mu}{a_{p(+)}} * \frac{r_{pa(+)}}{r_{pp(-)}}} \quad (8)$$

$$v_{tp} = \sqrt{\frac{\mu}{a_t} * \frac{r_{fa}}{r_{pp(-)}}} \quad \text{and} \quad v_{ta} = \sqrt{\frac{\mu}{a_t} * \frac{r_{pp(-)}}{r_{fa}}} \quad (9)$$

$$v_{fa} = \sqrt{\frac{\mu}{a_f} * \frac{r_{fb}}{r_{fa}}} \quad (10)$$

where  $\mu$  is the Earth's gravitational constant (398600km<sup>3</sup>/s<sup>2</sup>) and the value of vpp is selected to be the smallest of vpp(-) and vpp(+). The selection of perigee minimum value will lead to obtain the maximum anticipated velocity change, which represents the severe value of velocity and consequently the safer velocity budget calculation.

The total required velocity change  $\Delta v_{total}$  for altitude and inclination correction expresses as

$$\Delta v_1 = v_{tp} - v_{pp} \quad \text{and} \quad \Delta v_2 = v_{fa} - v_{ta} \quad (11)$$

$$\Delta v_{alt} = \Delta v_1 + \Delta v_2 \quad (12)$$

$$\Delta v_i = v_{fa} * i_e \quad (13)$$

$$\Delta v_{total} = \Delta v_{alt} + \Delta v_i \quad (14)$$

where  $\Delta v_{alt}$  and  $\Delta v_i$  is the velocity change required to eliminate altitude and inclination of LV error.

The satellite mean motion (n) is representing the number of revolutions per day the satellite performs around the Earth, it can be calculated as follows:

$$n = \left[ \sqrt{\frac{\mu}{a_f^3}} \right] * \left[ 1 + (\sqrt{(1 - e^2)}) * \frac{(1.5 * R_E^2 * J_2)}{(P^2)} * 1 - (1.5 * \sin^2 i) \right] \quad (15)$$

where eccentricity  $e = \frac{(r_{fa} - r_{fp})}{(r_{fa} + r_{fp})}$ , semi-latus rectum  $P = a_f * (1 - e^2)$  of the orbit and  $J_2$  is Second Jeffery's constant of gravity.

The consumed propellant  $m_{pr}$  and delta velocity  $\Delta v_{eng}$  by thruster operation are calculated as:

$$m_{pr} = \left( \frac{F}{I_{sp}} \right) * t_{op} \quad (16)$$

$$m_e = m_{sc} - m_{pr} \quad (17)$$

$$\Delta v_{eng} = I_{sp} * \ln \left( 1 - \frac{m_{sc}}{m_{pr}} \right) \quad (18)$$

where  $m_{pr}$ ,  $m_{sc}$ ,  $m_e$  are the mass of propellant, satellite and satellite empty mass respectively,  $I_{sp}$  is the propellant specific impulse,  $F$  is the thrust value and  $t_{op}$  is the maximum duration of thruster operation.

After knowing the total required velocity of transfer ( $\Delta v_{tot}$ ) and the amount of velocity change per each manoeuvre ( $\Delta v_{eng}$ ), it can be easily calculated the total number of manoeuvres required (N) and though the overall orbit transfer duration ( $T_{dur}$ ) in days as follows:

$$N = \frac{\Delta v_{tot}}{\Delta v_{eng}} \quad (19)$$

$$T_{dur} = \frac{N}{n} \quad (20)$$

The total consumed propellant mass during the orbit transfer mission  $m_{pr\_tot}$ , it can be calculated as follows:

$$m_{pr\_tot} = m_{sc} * (1 - e^{-\frac{\Delta v_{tot}}{I_{sp}}}) \quad (21)$$

### 2.3. Orbit transfer model block diagram

The mathematical model implementation is based on the sub-routine structure, where the debugging of each routine is easily carried out. The block diagram of the orbit transfer model is presented in Fig. (2).

The block diagram consists of four main blocks, the input data, active manoeuvre, correction data calculations and the output data. The input block includes the main specifications data of the satellite (mass and propulsion system specifications), the required insertion orbital parameters (altitude and inclination) and the launch vehicle insertion error. The second block, which is named “Active manoeuvre”, contains the main equations of Hohmann transfer using the stated propulsion system of the used satellite and the propagation equations representing the passive motion of satellite in orbit. The third block “correction data calculations” is an intermediate block that calculates the raw data needed for the output block, finally the fourth output block, it represents the main required data from the whole orbit transfer model, which are used in the mission analysis stage of orbit design.

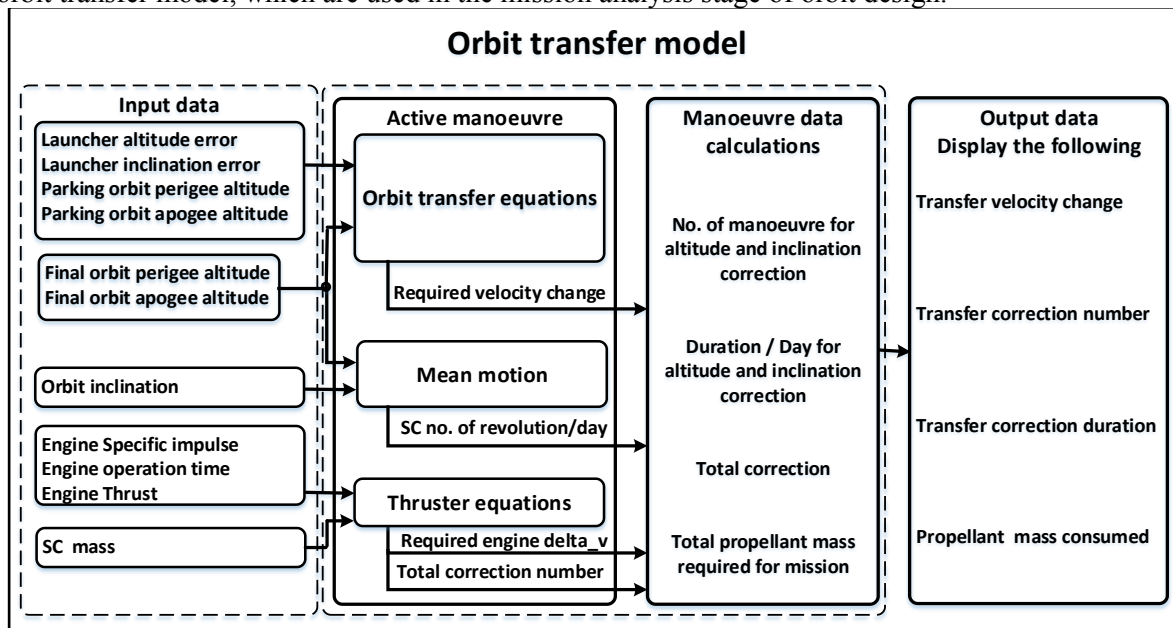


Figure 2. Orbit transfer model Block Diagram

### 2.4. Orbit maintenance model

The orbit maintenance mathematical model is built on the same concept of orbit transfer but excluding the equations representing the LV insertion error. The orbit transfer is referred to the SC manoeuvres after the LV separation where the required altitude raise is in terms of tens and hundreds of km [16]. For the orbit maintenance, it is carried out during the lifetime of SC in its working orbit. After a certain period of operation, the SC degraded in altitude in terms of hundreds of meters to about 2 or 3 km’s annually for a SC orbit altitude above 600 km in the LEO. The orbit maintenance manoeuvres work on correcting this small change in altitude.



### 2.4.1. Deorbiting Model

By the end of the designed lifetime of the SC. The SC should be removed from the working orbit to give space for another operating satellite and not to harm the already operating satellites that may collide with them as artificial debris.

The de-orbiting model is built on the same concept of orbit transfer, based on the reports issued by the “Committee on The Peaceful Uses of Outer Space”[17] and the “Inter-Agency Space Debris Coordination Committee”[18]. The two committees stated that deorbit of SC with perigee below 2000 km should remain in orbit for no longer than 25 years after completion of the operational mission. The regulations of SC deorbit by the end of its lifetime can be achieved by fulfilling one of the following two requirements.

To decay the attitude of SC to a lower altitude where it takes 25 years to decay passively to the Earth’s atmosphere (Long period deorbiting).

To actively decay the SC altitude directly to the Earth’s atmosphere ( $\approx 100$  km) where it takes less than one orbit to reach the Earth’s ground (Short period deorbiting).

## 3. FDM graphical user interface

To obtain a user friendly utilization of the constructed FDM, the (GUI) application is connected with the program syntax. The GUI includes the input parameters of each satellite transfer phase, control tabs (e.g. run tab), output parameters data with different (2D, 3D) charts of the dynamic responses of satellite. This GUI also helps the user to easily clarify the different outputs for each satellite transfer phase. The graphical user interface of the flight dynamic model is given in Fig. (3).

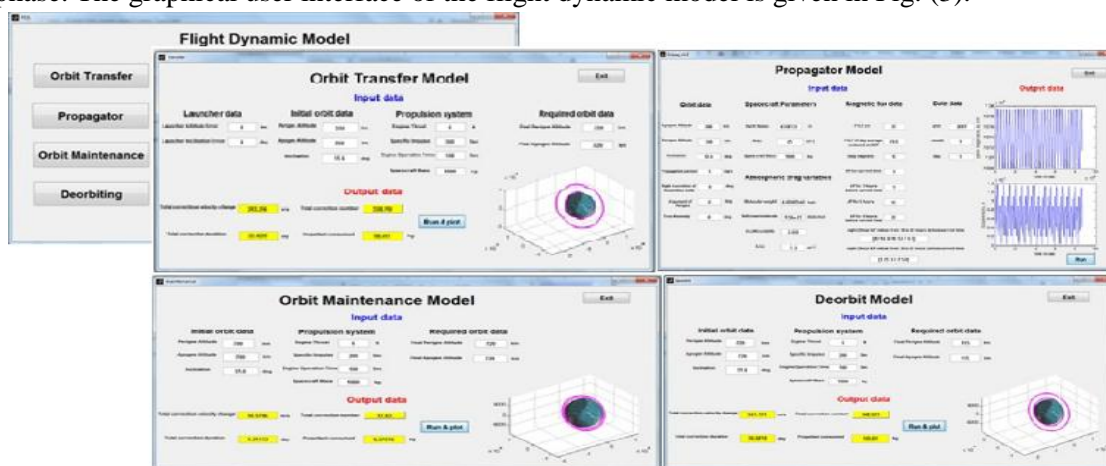


Figure 3. GUI windows of FDM

## 4. Matlab/Simulink Simulation

The mathematical model implementation is based on the sub-routine structure, where the debugging of each routine is easily carried out. The model block diagram is implemented in the Simulink and its layout scheme is printed-out in Fig. (4).

The model is built on the general case of orbit transfer, which is transferring from an elliptical orbit (parking orbit) to an elliptical orbit (working orbit) by an ellipse orbit (transfer orbit) with consideration of launcher error in altitude and inclination. The general case comprises also when transferring from circular to elliptical and vice versa.

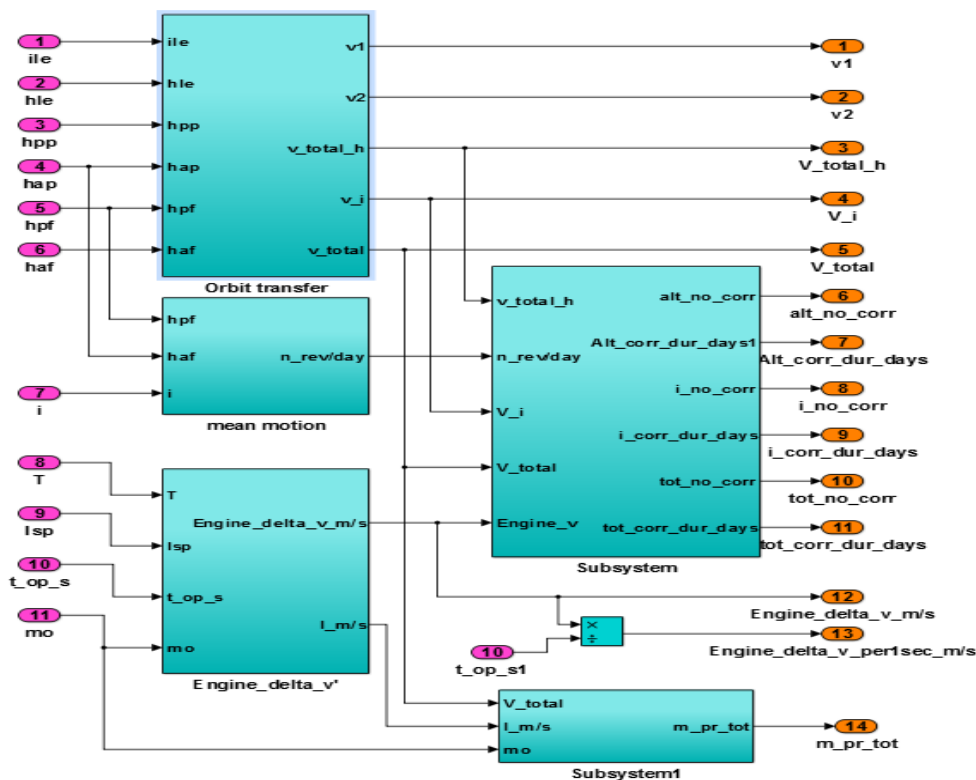


Figure 4. model block diagram

## 5. Model results and verification

Verification of the mathematical model output is carried out using the Astrogator module in STK application. This module enables adding a user defined systems of the SC with all its specifications, which is used to examine the performance of such systems after their operation under the specified operating conditions. Mainly in the Astrogator, SC specifications are tailored to resemble the designed model (SC dry mass, propellant mass, drag coefficient, effective area exposed to drag/solar radiation and the insertion state vector classical orbital elements) as shown in Fig. (5). Orbit transfer phase depends on the propulsion system, that's why its detailed specifications (specific impulse, thrust and operation conditions) are entered to examine its effect. To identify the starting and stopping conditions of the thrusters to perform the required manoeuvres, a criterion is chosen to select the timing of starting the manoeuver to be before reaching the apogee/perigee by half the designed duration of each manoeuver as illustrated in Fig. (6).

After running the built STK/Astrogator scenario, the SC performs the orbit transfer manoeuvres as entered from designed model. Output from the scenario is the final reached altitude, which includes the apogee/perigee altitudes and the corresponding consumed propellant mass ( $\Delta V$  budget). Finally, STK/Astrogator output data is compared with the MATLAB/Simulink designed model to measure its consistency and accuracy. The orbit transfer designed model is capable of executing manoeuvres from a circular to elliptical or circular working orbits.

After the scenario build sequence is completed, it comes to appoint where it is ready to be "Run". After running the scenario, the 2D and 3D in Fig. (7). view windows of the globe show the total number of revolutions the SC performed during the orbit transfer process. There are two colors on the transfer revolutions; the red color illustrates the passive section (propagate segment) and the red color is for the active manoeuvres section (Sequence segment) as shown in Fig. (7).



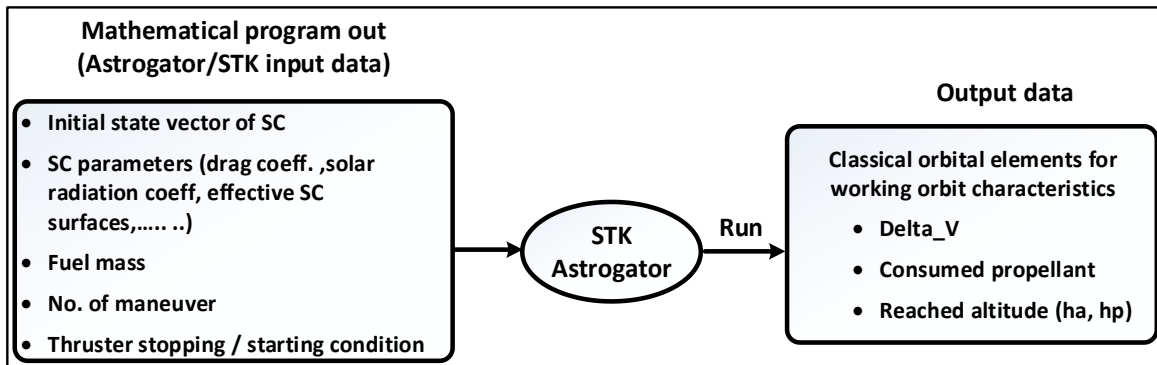


Figure 5. STK Astrogator

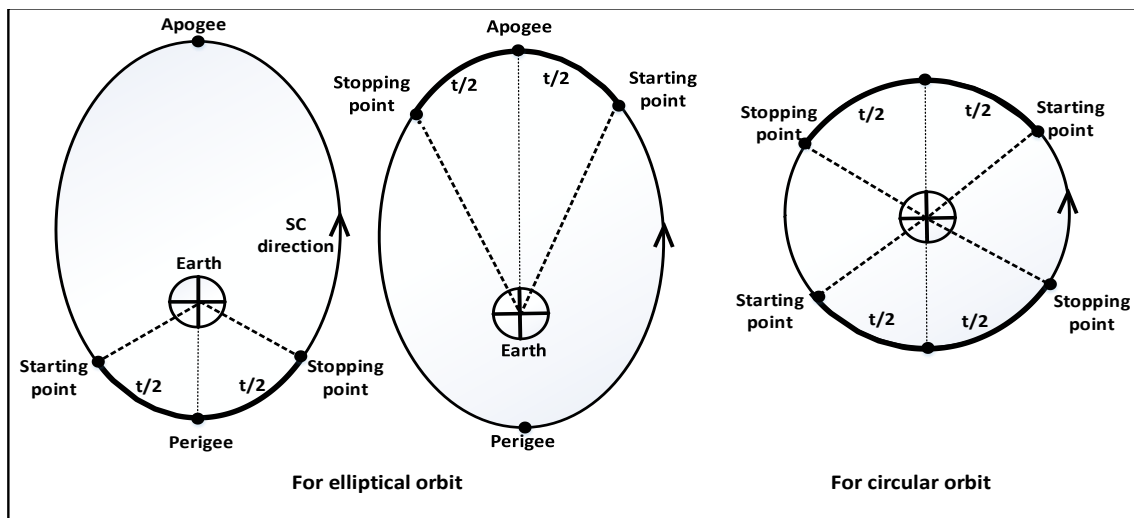


Figure 6. Selection criterion for thruster s

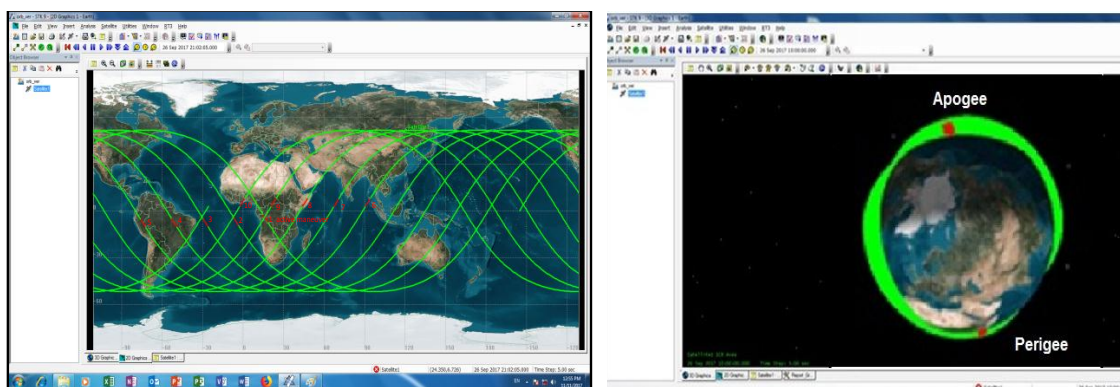


Figure 7. View of globe with tra

The verification of orbit transfer model is carried out by orbit transfer from a circular parking orbit at 350 km to different circular altitudes from 500 to 800km, another verification is performed with increasing the base circular orbit to an elliptical orbit with a perigee altitude 500 km and different apogee altitudes from 600 to 800 km. Proposed orbit transfer scenarios are executed on the implemented GUI model (deduced mathematical model) and is compared with the same scenario result obtained from the STK application. Three output parameters are compared, which are the apogee altitude, perigee altitude and the consumed propellant mass. Verification results are shown in Table (1).

Table 1. Orbit transfer mathematical model results and verification.

Manoeuvre Parameters		Final apogee altitude $h_a$ , km		Final apogee altitude $h_p$ , km		Consumed propellant mass, kg		
		STK package	Error %	STK package	Error %	Model	STK package	Error %
$(h_a \times h_p)$ Km	$(n_a, n_p)$							
Circular (500)	(115,117)	495.964	0.81	494.687	1.07	33.824	33.917	-0.27
Circular (600)	(159*162)	597.156	0.48	595.166	0.81	55.148	55.193	-0.08
Circular (700)	(202*206)	698.264	0.25	694.702	0.76	75.559	75.549	0.01
Circular (800)	(244*250)	794.116	0.74	793.581	0.81	95.109	95.800	-0.72
Elliptical (600*500)	(69*117)	594.771	0.88	485.87	2.91	44.563	44.497	0.15
Elliptical (700*500)	(69*162)	693.688	0.91	483.717	3.37	54.992	54.962	0.05
Elliptical (800*500)	(69*206)	792.466	0.95	483.331	3.45	65.124	65.086	0.06

- Orbit transfer GUI mathematical model verification shows a common error with the STK model due to the existence of atmospheric drag effect and solar radiation pressure in the STK application, which are not included in the mathematical model.
- Final apogee altitude verification shows a maximum error less than ~1% for circular and elliptical transfers.
- For the final perigee altitude verification, circular transfer reached a maximum error that is ~1%. For the perigee altitude elliptical transfer, a greater error is experienced with a maximum value of ~3.5%, the increase in error is due to the lower altitude of perigee, which has a greater atmospheric drag effect, a matter of fact that increases the error value between the GUI math model and the STK application.
- Propellant consumed verification using the chemical PS shows a mild error with a maximum value ~0.7%, as the amount of consumed propellant mass is very small for the given altitude errors discovered for proposed scenarios.

## 6. Model results analysis

A further analysis is carried out to see the difference between different propulsion systems (Chemical and electrical), orbit transfer manoeuvres are calculated for different altitude changes from 500 km to 800 km as shown in Table (2). For a constant velocity budget manoeuvres ( $\Delta V$ ), the total number and duration of manoeuvres required by electrical PS are ~9.5 times the number required by using the chemical PS. The mass of consumed propellant for the chemical PS is ~5 times the required mass of propellant for electrical PS. From the mentioned analysis concerning the total number & duration of manoeuvres and the propellant mass for different PS's, it is concluded that:

- Chemical PS is used for fast response manoeuvres with a disadvantage of higher consumed mass of propellant.
- Electrical PS can be used for missions that has no constraint on execution duration, but, the mass of propellant is of higher importance to be low.

- Chemical PS is of lower complexity in design and implementation when compared to electrical PS.

**Table 2.** Transfer mission using chemical or electrical PS

Manoeuvre Parameters	500 km		600 km		700 km		800 km	
	Chem. PS	Ele. PS	Chem. PS	Ele. PS	Chem. PS	Ele. PS	Chem. PS	Ele. PS
Total manoeuvre velocity, m/s	84.39		139.13		192.68		245.10	
Total manoeuvre number	141	1340	232	2208	321	3058	409	3891
Total manoeuvre duration, day	9.09	86.57	15.15	144.30	12.21	202.00	27.28	259.80
Propellant consumed, kg	33.83	6.86	55.15	11.28	75.56	15.59	95.10	19.79

The SC de-orbiting to an orbit that will stay for less than one revolution means that SC should be degraded in altitude to reach the atmospheric layer. The calculated altitude of the final orbit (to reach the atmospheric layer) is equal to 115 Km, de-orbit transfer manoeuvres is calculated from different initial altitudes starting from 800km to 500km and executed by chemical propulsion system (mono-propellant system with thrust=6 N, Isp = 250 sec) as shown in Table (3).

**Table 3.** Fast deorbiting mathematical model results

Manoeuvre Parameters	500 km	600 km	700 km	800 km
Total manoeuvre velocity change, m/s	222.4	277.1	330.6	383.0
Total manoeuvre number	353	432	506	574
Total manoeuvre duration, day	22.21	27.52	32.54	37.34
Propellant consumed, Kg	82.535	99.996	115.685	129.955

SC long de-orbit reflects the regulation, which states that SC can be degraded in altitude to an orbit with lifetime of 25 years. The altitude of the final orbit for the SC used in study is equal to 567 Km. The manoeuvre scenario is calculated for initial SC orbit starting from 800k to 600 km as shown in Table (4).

**Table 4.** Long deorbiting mathematical model results

Manoeuvre Parameters	600km	700	800
Total manoeuvre velocity change, m/s	17.9	71.5	124.0
Total manoeuvre number	28	110	186
Total manoeuvre duration, day	1.87	7.39	12.69
Propellant consumed, Kg	6.819	26.360	44.304

## 7. Conclusion

The comprehensive of a mathematical flight dynamic model, the verification and analysis results are concerned with the vital parameters (velocity budget, number of manoeuvres, duration and consumed propellant mass) during the various transfer missions (orbit transfer, maintaining and de-orbiting). The following conclusions are:

### 7.1. For orbit transfer model

- the final altitude verification shows a maximum error less than ~3.5% for circular and elliptical transfers, the increase in error is due to the lower altitude of perigee and a greater atmospheric drag effect,
- Propellant consumed verification using the chemical PS shows a mild error with a maximum value ~ 0.7% due to altitude errors.
- For a constant velocity budget manoeuvres ( $\Delta V$ ),

- the number and duration of manoeuvres required by electrical PS are  $\sim 9.5$  times required by using the chemical PS,
- the consumed propellant for the chemical PS is  $\sim 5$  times the required mass of propellant for electrical PS,
- The chemical PS performance is faster in its response with respect to electrical PS, recommendation goes to the chemical PS to shorten the orbit transfer period.

### 7.2. For orbit maintenance model

- The SC degraded in altitude for one year is  $\sim 0.1$  to 2 or 3 km at above 600 km,
- The degraded amount at altitude 500 km is higher than at the altitude 800 km, that's why, the amount of velocity budget  $\Delta V$  required is higher at lower altitude, as well as the required duration and consumed propellant.

### 7.3. For de-orbit model

- Fast de-orbiting required placing the SC at the altitude of 114.5 km (1st variant), the required  $\Delta V$  budget  $\approx 168.84$  m/s, period for electrical PS  $\approx 10$  times greater than chemical PS, unfortunately, propellant consumed for chemical PS  $\approx 6$  times greater than electrical PS.
- Long de-orbiting required placing the SC at the altitude of 567 km (2nd variant), SC will remain in the orbit  $\approx 24.3$  years, the  $\Delta V$  budget requires  $\approx 41$  m/s
- The second variant will be preferred for de-orbit pattern for more economical and mass budget point of view.

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