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Orbital Motion Modelling for Spacecraft Mission Analysis and Design

S. Chouraqui and M. Benyettou
Department of Data Processing, University of Sciences and Technology of Oran,
USTO, BP 1505 EL M'Noouar, Oran, Algeria

Abstract: Visualization of the orbit motion of a satellite in space is an important step to carryout mission planning and design. In an environment of learning such visualization tools help the students to understand and appreciate the dynamical motion. The purpose of this study was to present, first, the simulation of the satellite orbit analysis based on data collected from NASA/Norad two line element files. The analysis includes orbit determination and prediction of satellite position/velocity, orbit eclipse and ground trace and so on. Second the application of this simulation to design the Low Earth Orbit (LEO) satellite power subsystem. The intention of this simulation was to create a Graphical User Interface (GUI) which consists of menus, push buttons, text boxes, plots and other interface devices that allow the GUI user to immediately see the impact of various changes.

Key words: Orbit analysis, orbit determination, satellite position, satellite tracking, eclipse, solar panels, power system

INTRODUCTION

To track satellites through space, it is needed to know where they are now and where they'll be later so that sensor coverage can be predicted and are pointed antennas at them to gather data. Although we can easily predict this motion when the orbit is a circle, the problem becomes more complicated when the orbit is an ellipse and most orbits are at least slightly elliptical.

An orbit propagator is a mathematical algorithm for predicting the future position and velocity (or orbital elements) of an orbit given some initial conditions and assumptions[1].

The simulation of the orbit of a satellite runs in MATLAB environment by returning the position and velocity vectors at corresponding times. The input parameters to the function developed are defined in a vector containing the Kepler elements as well as the Start and End simulation time in Julian day. The orbits are propagated using the standard SGP4 module for near earth satellites and SDP4 module for deep space satellites[2].

The orbit of a satellite is highly non-linear trajectory due to not only classical Kepler's motion theory but also several perturbation factors such as irregular potential field of earth, lunisolar attraction, atmospheric drag and solar radiation pressure.

SGP4/SDP4 employs general perturbation theory to provide highly accurate prediction of orbital positions. SGP4/SDP4 is also useful if Keplerian elements are old because it more accurately accounts for long-term effects of small gravitational and drag force.

After computing the orbit, the data can be used to generate a best-estimate history of where the satellite was (definitive orbit) and where it will (predicted orbit), along with the associated speed and direction (velocity) at each point. The definitive orbit, being the most accurate available, is used by the experimenters in processing and interpreting scientific data. The predicted orbit is used to produce scheduling aids. Scheduling aids indicate spacecraft environmental conditions (such as sun or shadow, interference regions and altitude) as well as all potential station-to-station contact times (view periods). Using this list of environmental conditions, the experimenter can plan scientific data collection and knowing the station view periods, can select the times needed to meet mission communications requirements.

The power subsystem is a direct energy transfer system consisting of solar array, batteries and power system electronics[3]. The power subsystem provides the major electrical functions of power control, energy storage and distribution of power to the observatory loads. Power which is generated in the solar array is supplied directly to the observatory loads. The orbital period and the fraction of time that spacecraft is in sunlight (and eclipse) are of fundamental importance to the design of the power subsystems. The most importance thing in the power subsystems the solar array analysis, which the

Corresponding Author: S. Chouraqui, Department of Data Processing, University of Sciences and Technology of Oran, USTO, BP 1505 EL M'Noouar, Oran, Algeria

fundamental is to determine the angle between a solar panel vector and the sun vector and give the amount of power available from each solar panel.

The described orbit model is used to design the power available from the solar panel. To point the solar panels located on the surfaces of the cubic satellite toward the sun as accurately and continuously as possible for the purpose of recharging the on board batteries, we will suppose a stabilised satellite attitude[4].

Coordinate system and time: Time and coordinate systems are an inherent component of the orbit determination problem. As it will be described, measurements of some aspect of a satellite's position or motion must be indexed in time. The time index of the measurements must be related to the time used in the equations of motion, but different time systems are used, all of which attempt to represent the concept of Newtonian time. The relation between the various time systems must be known[1].

To describe the position of an object in the earth or in the space, it is need to define some coordinate systems. This study explain how to transfer a vector from one coordinate system to another coordinate system. This is very important, because sometimes we describe the position of a spacecraft in the space, but later a relationship must be find between this spacecraft and a specific target the ground of the earth. Therefore those transformations between each coordinate system are needed.

Transformation between time systems: Usually calculation is performed the program must interface with external input-output data sets which are referenced to different time system, such as Universal Time (UT) for computing Green wich sidereal time and Universal Time Coordinate (UTC) for input-output epochs and tracking data. Therefore a brief description of the relevant time systems and their interrelationships is needed[1,5].

The time systems which are used in this work are mainly Coordinated Universal Time, Sidereal Time and Julian Day[1].

Gregorian calendar to JD: If UTC or some the other time system is expressed as a Gregorian Calendar, i.e., Year-Month-day-Hour-Minute-Second[YMDHmS], JD corresponding to this time is computed as follow[6]:

$$JD = 367\times Y - \text{int}\Big(7\big(Y+\text{int}\big((M+9)/12\big)\big)/4\Big)$$
$$-\text{int}\Big(3\big(\text{int}\big((Y+(M-9)/7)/100\big)+1\big)/4\Big)$$
$$+\text{int}(275\times M/9)+D+172028.5+\text{frac}(JD)$$

Where, frac (JD) is the fractional part of JD and given by:

$$\text{frac}(JD) = \frac{H}{24}+\frac{m}{24\times 60}+\frac{S}{24\times 60\times 60}$$

NORAD defined to JD: The NORAD defined format is yyddd.dddd, where yy is the year and ddd.dddd is the elapsed day number from the 0:00 UTC 1st of January.

If the NORAD defined format yyddd.dddd is given, the JD corresponding to this time is computed as following:

$$JD = JD0+ddd.ddd$$

JD0 the JD of 0:00 UTC 1st of January of the year yy.

Sidereal time: The sidereal time is simply time measured relative to the stars; time relative to the position of the sun is measured. This time scale is referred to as mean solar time. As with any time system, time is defined as the angle between the observer and some reference direction. With mean solar time, the reference direction is the direction of the mean sun. With sidereal time, the direction is the vernal equinox-just the direction we need for our calculation[1,5].

Coordinate transformation: In an orbit simulation/orbit determination process a lot of information will be needed in different reference frames. For example, the satellites equations of motion are described in an inertial frame, while the coordinates of a tracking station or user will be given in an earth-centred-earth-fixed frame. Therefore, it will be necessary to transform force, velocity and position vectors from one frame to another. This is done using rotation matrices.

Transformation between ECI and Orbit plane: In an Earth Centred Inertial (ECI) coordinate system, the unit vectors i, j, k (Fig. 1), are usually introduced, with i taken along the x axis and j, k associated with *y* and z-axes, respectively.

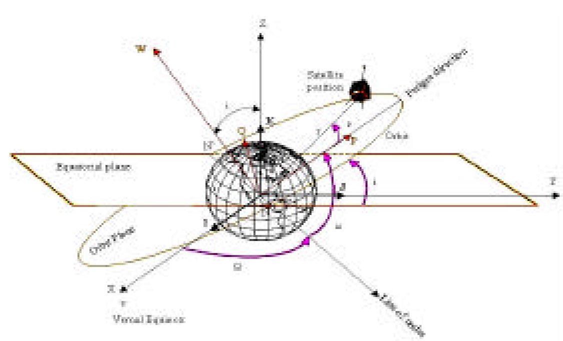


Fig. 1: Unit vector in ECI coordinate system

The angles in this Fig. 1, i, Ω, ω are the classical orientation angles used to define the position of the orbit in space. Their definition is described above. Following we will describe how to transfer a vector from I, J, K to orbital plane P, Q, W.
The transformation is proceeded as follow:
Rotate through the angle Ω about z as follows:

$$I' = \cos\Omega\, I + \sin\Omega\, J$$
$$J' = -\sin\Omega\, I + \cos\Omega\, J$$
$$K' = K$$

Or more compactly,

$$\begin{bmatrix} I' \\ J' \\ K' \end{bmatrix} = \begin{bmatrix} \cos\Omega & \sin\Omega & 0 \\ -\sin\Omega & \cos\Omega & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} I \\ J \\ K \end{bmatrix}$$

the primed variables denoting a first rotation.

Rotate the primed axes through the angle i about I' as follows:

$$\begin{bmatrix} I'' \\ J'' \\ K'' \end{bmatrix} = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos i & \sin i \\ 0 & -\sin i & \cos i \end{bmatrix} \begin{bmatrix} I' \\ J' \\ K' \end{bmatrix}$$

The double primes denoting a second rotation.

Finally, rotate the double primed axes through the angle ω about K'' to obtain:

$$\begin{bmatrix} P \\ Q \\ W \end{bmatrix} = \begin{bmatrix} \cos\omega & \sin\omega & 0 \\ -\sin\omega & \cos\omega & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} I'' \\ J'' \\ K'' \end{bmatrix}$$

Therefore,

$$\begin{bmatrix} P \\ Q \\ W \end{bmatrix} = \begin{bmatrix} P_x & P_y & P_z \\ Q_x & Q_y & Q_z \\ W_x & W_y & W_z \end{bmatrix} \begin{bmatrix} I \\ J \\ K \end{bmatrix}$$

Where:
$P_x = \cos\omega \cos\Omega - \sin\omega \sin\Omega \cos i$
$P_y = \cos\omega \sin\Omega + \sin\omega \cos\Omega \cos i$
$P_z = \sin\omega \sin i$
$Q_x = -\sin\omega \cos\Omega - \cos\omega \sin\Omega \cos i$
$Q_y = -\sin\omega \sin\Omega + \cos\omega \cos\Omega \cos i$
$Q_z = \cos\omega \sin i$
$W_x = \sin\omega \sin i$
$W_y = -\cos\Omega \sin i$
$W_z = \cos i$

Transformation between ECEF and ECI: The Earth Cantered Earth Fixed (ECEF) coordinate system is a right handed Cartesian coordinate system with the origin at the centre of mass of the reference ellipsoid and a z-axis pointing toward the north pole, a x-axis pointing toward the intersection of the prime meridian and the equator and y-axis completes a right handed of the ECEF orthogonal coordinate System. It is customary to locate an object relative to the earth by two angular coordinates (latitude-longitude) and altitude above (below) the adopted reference ellipsoid.

The explicit form for the transformation matrix from ECI to ECEF is:

$$\begin{bmatrix} \cos\Theta & \sin\Theta & 0 \\ -\sin\Theta & \cos\Theta & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

With Θ is the angle hour.

Orbit computation: Previously, we introduced definitions of time and coordinate systems and their transformations. The next step, we need to know how a spacecraft moves in the space, how to describe its movements?

The classical Kepler orbit is described by six parameters:

a: semi major axis
e: numerical eccentricity
i: inclination of the orbital plane
Ω: right ascension of the ascending node
ω: argument of perigee.
T_o: time of perigee crossing

The three Keplerian law are associated with the following equations:

1. Keplerian law (orbit energy)

$$\frac{v^2}{2} - \frac{GM}{r} = -\frac{GM}{2a}$$

2. Keplerian law (rotational impact)

$$h = r \cdot v \cdot \cos\gamma$$

with γ angle between the normal on the radius vector and the velocity vector.

3. Keplerian law (orbit period)

$$T^2 = \frac{4\pi^2}{GM} a^3$$

Together with the geometrical equations for the ellipse, the movement of a satellite in his orbital plane can be described. In the following the equations are given only with a brief description[1].

Radius: $$r = \frac{P}{1+\varepsilon\cos\varphi}$$

Ellipse parameter: $$P = a\left(1-\varepsilon^2\right)$$

Time of flight: $$t-T_o = \frac{T}{2\pi}\cdot\left(M-\varepsilon\sin M\right)$$

Eccentric anomaly: $$E = \arccos\left(\frac{\varepsilon+\cos\varphi}{1+\varepsilon\cdot\cos\varphi}\right)$$

Mean anomaly: $$M = E-\varepsilon\cdot\sin E$$

Mean motion: $$n = \frac{360}{T} = \frac{2\pi}{T}$$

Flight path angle: $$\tan\gamma = \frac{\varepsilon\sin\varphi}{1+\varepsilon\sin\varphi}$$

With:
φ: True anomaly
M: Mean anomaly
T: Orbital period

To obtain three dimensional Cartesian co-coordinates, the ellipse parameters have to be transformed to Cartesian vector using the following expression:

$$\overline{x}_{OP} = \begin{Bmatrix} r\cos\varphi \\ r\sin\varphi \\ 0 \end{Bmatrix}$$

The index OP indicates a reference frame lying in the orbital plane with the x-axis coinciding with the line of apsis, the z-axis normal to the orbit plane and the origin being the focus of the ellipse.

The transformation from the orbital plane frame to an inertial fixed frame is done applying the following vector-matrix operation.

$$\overline{x}_I = \begin{bmatrix} P_x & Q_x & 0 \\ P_y & Q_y & 0 \\ P_z & Q_z & 0 \end{bmatrix}\cdot\overline{x}_{OP}$$

NORAD two line element definition: In this study, to predict satellite position we have used NORAD two-line element setting to determine the satellite's orbital elements[2].

Data for each satellite consists of three lines in the following format (Fig. 2).

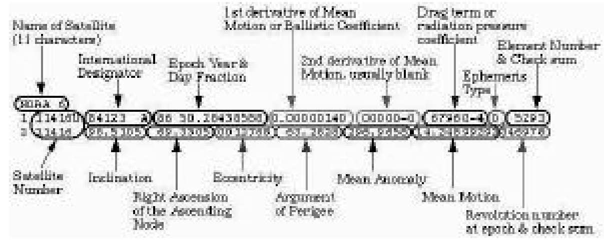


Fig. 2: NASA Two Lines Elements [TLE]

Satellite ground track: The ground track of an orbiting satellite is the trace of its path across the surface of the earth. The path on the surface of the earth is the trace of the satellite's nadir, or the point directly below the satellite where the orbital plane intersect the earth's surface. All ground track drawings use the latitude-longitude coordinate system. Any point along the ground track is then described by two values: the latitude which measures how far North or South a point lies from the equator and the longitude, which measures how far East or West a point lies from the Prime Meridian[7].

As a satellite moves through its orbit, the earth rotates eastward and the ascending node precesses due to the earth gravitational field perturbation. Beginning at an ascending node, after one complete revolution, the satellite passes over a different point on the earth at the next ascending node (Fig. 3).

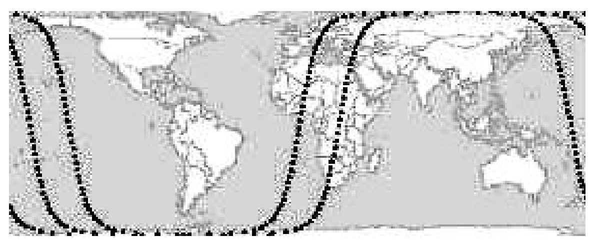


Fig. 3: Ground track drift

This shift in the ground track from one pass to the next is determined by Seller[7]:

Where:
ω_e: Represents the rotation rate of the earth
P: Represents the nodal period of the satellite
Ω: Represents the precession rate of the ascending node.

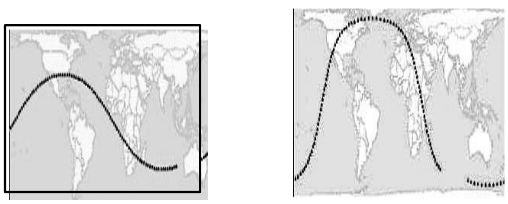
$$S = P\left(\omega_e-\dot{\Omega}\right)$$

The value of S can be interpreted as the shift of the ground track per orbit revolution (rad/rev). The drift rate of the actual ground track relative to the ideal ground track can then be described by:

$$\delta S = S - S_{ideal}$$

Simulation test: The results obtained by the visibility program developed, by changing the satellite's parameters to visualise how the orbit was predicted and how it is affected by the Keplerian Orbital Elements.

Effect of inclination: Figure 4 shows the ground traces of two different low earth orbits that vary only in inclination, all other parameters of the orbital element set remain unchanged. The inclination of orbit in Fig. 4a is 30° and orbit in Fig. 4b is 60°. The latitude of the highest point of the ground trace indicates the orbital inclination of the satellite. The same principle holds true concerning the latitude of the lowest point. The degree of latitude equals the inclination from 0° to 90°.



a) Inclination 30° b) Inclination 60°

Fig. 4: Effect of inclination

Effect of perigee: This ground trace in Fig. 5 is for an eccentric orbit with an inclination of 60° and a period of 12 h. The perigee is over the Southern Hemisphere and the apogee is over the Northern Hemisphere. At perigee, the satellite is travelling faster than it is when at apogee. The satellite would actually spend 2 h below the equator and 10 h above the equator during each revolution around the earth.

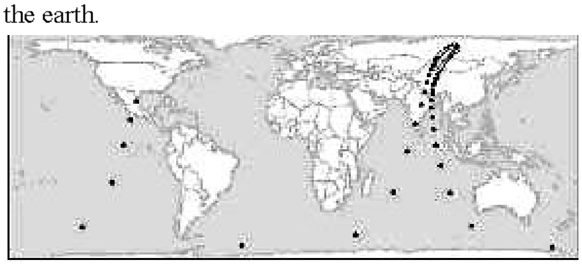


Fig. 5: Effect of perigee

Polar orbit: Polar orbits have inclination at or near 90°. A satellite in a polar orbit passes over or close the North and South Pole on each orbit. A satellite in a low earth polar orbit will eventually pass over most, if not all, of the earth's surface in a single day. For this reason, the low earth polar orbit is popular for earth resources, weather and surveillance satellites (Fig. 6).

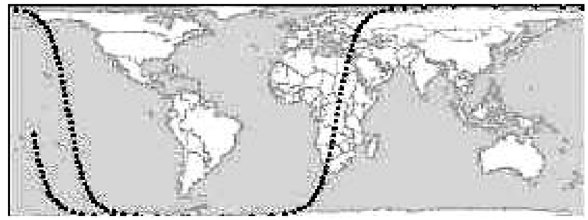
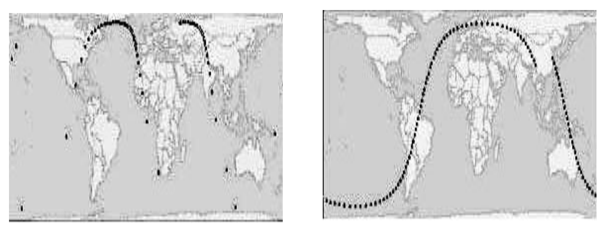


Fig. 6: Polar orbit

Effect of period: As the period of an orbit increase, the satellite's altitude increases covering less ground in a given amount of time. There are trade-offs made between the altitude of different orbits. A lower altitude orbit provides for improved resolution and global coverage in a single day (inclination dependent). A higher altitude orbit provides for longer continuous access to a single point on the surface of the earth (Fig. 7a and b).



a) Increased period b) Decreased period

Fig. 7: Effect of period

Geosynchronous orbit: A geosynchronous orbit has a period equal to one day and an altitude of about 36000 km above the surface of the earth. Therefore, it takes as long for the satellite to revolve around the earth as it takes for the earth to rotate once on its axis. If the orbit is circular and equatorial (inclination = 0), the satellite is geostationary and appears motionless in the sky to a ground observer. If the orbit is slightly eccentric, but still equatorial, the satellite will appear to drift to the left and right of an observer on the equator in twelve hour cycles. If the orbit has an inclination about a point on the equator, the satellite was shown in Fig. 8.

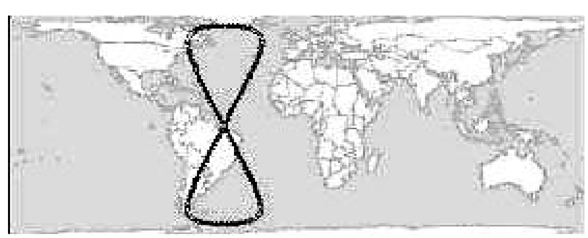


Fig. 8: Geosynchronous orbit

Molniya orbit: The Molniya orbit is used principally by Russian communications satellites. The orbit is characterised by a long orbital period (12 h), a high eccentricity (0.7) and an inclination of about 63.4°. The combination of these three orbital parameters establish an orbit whose apogee is far above the northern hemisphere, at a distance from ground stations in excess of 40,000 km. Perigee is at relatively low altitude (1000 km) where the satellite is travelling at high velocity . As the satellite climbs in its orbit toward apogee over the northern hemisphere, it slows down. As a result, the satellite spends a large portion of its 12 h orbit (in excess of 8 h) high above the northern hemisphere (Fig. 9).

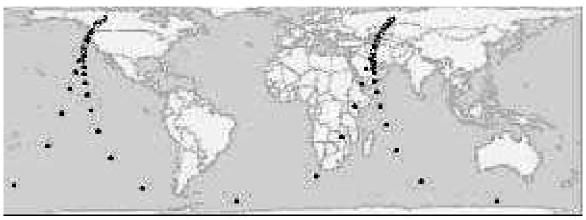


Fig. 9: Molniya orbit

The orbital period of Molniya is precisely 0.5 of sidereal day which enables the satellite to follow the same ground track on each orbit. It thus appear in the same relative position in the sky during each pass. Since the orbital period is half a day, apogee occurs twice per day, typically once over Russia and once over Canada.

User interface implementation: The search for a universal solution of the equation of motion for a satellite orbiting an oblate planet is a subject that has merited great interest because of its theoretical and practical implications.

A question to answer is: Given the orbital elements a, e, i, Ω, ω and M at epoch t_0, how to know exactly where a satellite is at time t and where it is going without ever once touching it?

User Interface developed for the visibility prediction program, it should provide the users with a friendly interface to input different satellites and the Keplerian Elements as well as the time to see how the position ant the orbit of the satellite change (Fig. 10 and 11).

Satellite power subsystem design: Reliable, continuous operation of the power system is essential to the successful fulfilment of spacecraft mission, a failure even a brief interruption in the source of power can have catastrophic consequences for the spacecraft's attitude and temperature control as well as its electrical systems. Therefore, the power system and its components must be designed and fabricated with reliability as a primary requirement.

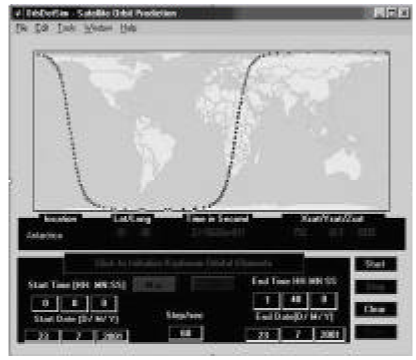


Fig. 10: Simulation GUI window

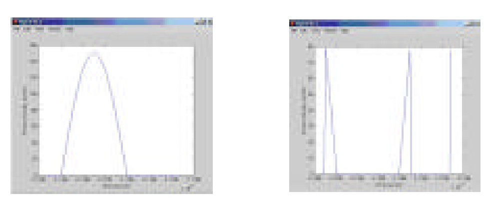
Fig. 11: Configuration GUI window

After doing the orbit simulation, which has a major role to design the satellite power subsystem, the aim of this part of work is to determine the amount of the power available from the solar panels for UoSat-2 satellite.

Given the NASA two lines elements for UoSat-2, the ground trace using the prediction visibility program presented before is performed in Fig. 10. The simulation results of the density power available for the satellite are given in figures as follows (Fig. 12a and b, 13a and b, 14a and b).

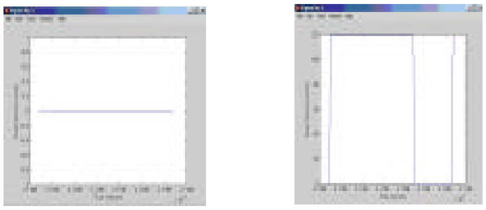
a: $+X_{sat}$ direction   b: $-X_{sat}$ direction

Fig. 12: Power density available from solar panels in $+X_{sat}$ and $-X_{sat}$ direction



a: $+Z_{sat}$ direction   b: $-Z_{sat}$ direction

Fig. 13: Power density available from solar panels in $+Z_{sat}$ and $-Z_{sat}$ direction



a: $+Y_{sat}$ direction b: $-Y_{sat}$ direction

Fig. 14: Power density available from solar panels in $+Z_{sat}$ and $-Z_{sat}$ direction

CONCLUSIONS

The aim of the study presented is to give a real time simulation, which provides support for model development and integration, simulation execution and analysis of results.

In order to achieve these missions it is therefore, essential to determine and predict an accurate orbit of a satellite.

The study given is the orbit simulation which is a starting point for the simulation of future, hopefully real, spacecraft.

The reason for doing this simulation was to give a visual insight into satellite orbits, by producing an animation of the satellite orbiting and also allowing a level of interaction, it lets the user gain a better understanding of satellite orbits. By changing different parameters the user is able to observe how it affect the shape and position of the satellite orbit.

Power generation and energy storage are the two main components of the power system. Consideration towards both of these systems should be done concurrently during the design phase. The advantages and disadvantages of both the power generation and energy storage methods must both be considered together in order to arrive at an optimal design.

The results out of the simulation were compared and tested with real tracking algorithm, it was shown that the compared program can then generate same track shape and the same eclipse time duration.

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