Orbit Propagation Assignment

Aaryav Mishra

July 2021

1 Introduction

The orbit propagator's main objective is to provide the current estimate position and velocity of our satellite. In this assignment, I have used GMAT to estimate an orbit and ground track of our satellite, which is at an altitude of 7000km; having a circular orbit with e=0 and comparing it with our results on MATLAB for the same plot. While accounting for J2 perturbations.

2 GMAT

In GMAT the values I entered were as follows: Under the spacecraft tab, I edited the defaultSC by defining the initial values in cartesian to be:

$$X = 7000$$
$$Y = 0000$$
$$Z = 0000$$
$$Vx = 000$$
$$Vy = -1.034737378012107$$
$$Vz = 7.474773493078592$$

Then I set the Epoch format to UTCGregorian, the Coordinate system to EarthMJ2000. Under the ballistic/mass section of GMAT, I set the following values as defined for our satellite:

$$DryMass = 4$$
$$Coefficientofdrag = 2.2$$
$$Coefficientofreflectivity = 1.8$$
$$DragArea = 0.01$$
$$SRPArea = 0.01$$

Rest of the values were left as default in GMAT. The required orbit was plotted along with the ground track while the report file was saved as .txt.

3 MATLAB

In MATLAB the initial conditions were defined same as that in GMAT. Then in a separate file the a function that executes the Runga Kutta 4 method on the acceleration function which also accounts for J2 was defined. The acceleration vector in the 3 directions being:

$$a_x = -\frac{\mu.x(1 + J_2\frac{3}{2}(1 - 5\frac{z^2}{r^2})}{r^3}$$

$$a_y = -\frac{\mu.y(1 + J_2\frac{3}{2}(1 - 5\frac{z^2}{r^2})}{r^3}$$

$$a_z = -\frac{\mu.z(1 + J_2\frac{3}{2}(3 - 5\frac{z^2}{r^2})}{r^3}$$

The initial state matrix involved 6 rows, containing the position vectors in the 3 directions and the velocity vectors in the 3 directions respectively. After applying RK-4 the resultant matrix consisted of 2 sets of 3 row vectors defining position and velocity respectively as in the case of the provided initial condition with an interval of 10 seconds spanning 10000 steps. The acquired position vectors were plotted in a 3D plot giving a circular orbit with some wobbliness in its path. The ground track for the same was plotted by converting the frame from ECI to ECEF frame Then finding the latitude and longitude of the corresponding points using the following formulae:

$$X_{ecef} = \frac{1}{2}(X_{eci}^2 + Y_{eci}^2)cos(\frac{i}{1000}) + tan^{-1}(\frac{X_{eci}}{Y_{eci}})$$

$$Y_{ecef} = \frac{1}{2}(X_{eci}^2 + Y_{eci}^2)sin(\frac{i}{1000}) + tan^{-1}(\frac{X_{eci}}{Y_{eci}})$$

$$Z_{ecef} = Z_{eci}$$

Since the Z axis would remain the same, along which rotation is taking place for frame conversion. At a given point,

$$latitude = sin^{-1}(\frac{Z_{ecef}}{R})$$

$$longitude = tan^{-1}(\frac{Y_{ecef}}{X_{ecef}})$$

Where R is the radius of the earth. These latitudes and longitudes are plotted as tiles in the 2D plot. In this plot the trajectory of the satellite as predicted by the orbit propagator is visualised as a ground track.

The results of both the GMAT orbit and ground track and the MATLAB orbit and ground track is attached along with the code.