Space Flight Operations
Assignment #3

AE 4361
Prof. G. Lightsey

1 February 2022

Due: 8 February 2022, 5:00 pm, submit as a pdf file on Canvas web site.

Late Submittal: Before 10 February 2022, 5:00 pm for 50% credit.

Office Hours: Professor Lightsey: Wednesday, 3:00-4:00 pm, BlueJeans (access from Canvas)
TAs: Monday, 5:00-6:00, BlueJeans (access from Canvas)
Or contact email for an appointment (see syllabus for email addresses)

Reading: Basics of Space Flight web site:
Chapter 3 (https://solarsystem.nasa.gov/basics/chapter3-1, all pages),
Chapter 4 (https://solarsystem.nasa.gov/basics/chapter4-1),
and Chapter 5 (https://solarsystem.nasa.gov/basics/chapter5-1).

Complete the reading and then complete the Multiple Choice Quiz on Canvas, which is due by February 8 at 5:00 pm.

This assignment also includes a Matlab portion. Your code must be your own work. Attach a copy of your Matlab code at the end of your assignment submission. Your code must include useful comments, your name, and the date of creation. Also, when necessary, include a screenshot of the code output.

Some of the problems will require you to solve Kepler's Equation, which can be done with the Newton-Raphson root finding method. For your convenience, two Matlab scripts have been included with the assignment: ***NewtRaph.m*** and ***Kepler_Solver.m***. Feel free to use both scripts to compute your answers for the problems that follow. You will need to copy both files into the same folder. Note that the ***Kepler_Solver.m*** file calls the ***NewtRaph.m*** script, so you will only need to call ***Kepler_Solver.m*** to compute your answers. Refer to the code comments for more information on how to run the script.

Problem 1.

An Earth-orbiting satellite is in an elliptical orbit with a period of 12 hours and a perigee radius of 12,000 km. Using this information answer the following questions:

(a) What is the semi-major axis of the orbit?

We are interested in the position of the vehicle at a time t = 9 hours after perigee passage. Answer the following for the current vehicle location at this time:

(b) What is the radius of the orbit?

(c) What is the velocity magnitude?

(d) What is the radial component of the velocity?

Problem 2.

A satellite in orbit about the Earth has instantaneous position given by $\boldsymbol{r}$ = 6045$\boldsymbol{i}$ + 3490$\boldsymbol{j}$ + 0$\boldsymbol{k}$ km and a velocity specified by $\boldsymbol{v}$ = −2.457$\boldsymbol{i}$ + 6.618$\boldsymbol{j}$ + 2.533$\boldsymbol{k}$ km/s. Determine the orbital elements for the satellite's orbit. Fill in the provided table with your results. Angles should be expressed in degrees and distances in kilometers. You may use Matlab to perform vector algebra. If you do, turn in a screenshot of your printed Matlab script and output.

Semi-major Axis (a)	
Eccentricity (e)	
Inclination (i)	
Right Ascension of Ascending Node (Ω)	
Argument of Perigee (ω)	
True Anomaly (ν)	

Problem 3.

This problem will focus on creating the Matlab function that will serve as the foundation for the ground track and satellite visibility problems in the next assignment. You will be required to build on this function, so it is important to make sure that everything is working as intended. In addition to the answers to the questions below, **you must also** turn in the following:

- **i.** Screenshots of your function outputs (one screenshot for each part of this problem).
- **ii.** A screenshot of the code you wrote to solve the problem. This must be appended at the end of the assignment and must include useful comments, your name, and the date of creation.

As mentioned earlier, you are allowed to use the ***NewtRaph.m*** and ***Kepler_Solver.m*** scripts that have been included with the assignment.

Your task for this problem is to create a Matlab function that takes in five orbital elements (*a, e, i,* Ω, and ω), as well as the time since perigee crossing (t), and then computes the respective position in ECI coordinates. The function output should be the *x, y*, and *z* components of the position (expressed in meters). You can use the following test case to verify that your function is working:

Input (ISS Orbit):

$a = 6.796620707 \times 10^{6}\ m$; $i = 51.6439°$; $e = 2.404 \times 10^{-4}$

$\Omega = 86.8571°$; $\omega = 1.8404°$; $t = 100\ s$

Output:

$$\underline{\mathbf{r}}_{ISS} = \begin{bmatrix} -0.239086091 \\ 6.747093009 \\ 0.769129531 \end{bmatrix} \times 10^{6}\ m$$

Make sure that your code outputs the correct results before moving forward. Using the function you created, find the ECI position coordinates (in meters) for each of the cases described below. As discussed earlier, provide a screenshot of your function output for each case.

(a) ***Vanguard-1 – launched in 1958, the oldest human satellite still in orbit!***

a = 8.612 × 10^6 m; i = 34.2396°; e = 0.1843301

Ω = 209.6300°; ω = 129.0719°; t = 300 s

(b) ***Hubble Space Telescope***

a = 6.907 x 10^6 m; i = 28.4714°; e = 0.0002445

Ω = 10.7658°; ω = 340.0906°; t = 3600 s

(c) ***Molniya Communication Satellite***

a = 2.323698972 × 10^7 m; i = 64.0370°; e = 0.680478

Ω = 343.6936°; ω = 288.0884°; t = 10,000 s

(d) ***Starlink 3327***

a = 6.722 × 10^6 m; i = 53.2164°; e = 0.0001762

Ω = 88.7877°; ω = 11.5745°; t = 900 s