ASEN5050 – STK Lab #3

DUE: Friday, 11/19/2015

Names: John Clouse

The purpose of this lab is to examine how to incorporate perturbations into STK's numerical integrators, and then evaluate those perturbations, including the J2 effect, drag, and interplanetary dynamics.

Note: This lab has been built for STK 9 – there may be small changes for STK 10.

Set Properties

Open STK and create a new scenario with the following epoch information:

Start Time: 1 Jul 2007 00:00:00.00 UTCG
Stop Time: 2 Jul 2007 00:00:00.00 UTCG

Set the *Epoch* to agree with the start time.

Click on the 2-D map and then the properties icon (on the toolbar). Click the *Details* tab. Deselect all the Items, and turn the Lat/Lon lines off. Look at what this does to your map display.

Propagate using J2 Perturbation

Create a satellite in the browser window called 'J2' that has the following properties:

a = 7000 km	Ω = 10 deg	orbit epoch = 01 Jul 2007 00:00:00
e = 0.05	ω = 20 deg	step size = 60 sec
i = 45 deg	ν = 90 deg	

Select *J2Perturbation* from the propagator menu. Select *OK* to begin the propagation.

1. Use this space to calculate dω/dt and dΩ/dt (in deg/day) for linear perturbations from Earth J2-oblateness:

$$\dot{\omega}_{sec} = \frac{3nR_{\oplus}^2J_2}{4p^2}\left\{4 - 5\sin^2(i)\right\} \qquad \dot{\Omega}_{sec} = -\frac{3nR_{\oplus}^2J_2}{2p^2}\cos(i)$$

```
>> n = sqrt(3.986e5/7000^3)*24*3600*180/pi
n =     5.336517796206832e+03 deg/day

>> p = 7000*(1-0.05^2)
p =     6.982500000000000e+03 km
```

```
>> dw_dt = 3*n*6378^2*0.0010826267/4/p^2*(4-5*
(sind(45)^2))
dw_dt =   5.422959476237151 deg/day

>> dRAAN_dt
= -3*n*6378^2*0.0010826267/2/p^2*cosd(45)
dRAAN_dt =  -5.112815226329515 deg/day
```

2. Highlight the satellite and select *Analysis -> Report & Graph Manager-> Classical Orbit Elements (Plot)* then click on the *Duplicate* button. Choose RAAN and Argument of Perigee for the Y-axis. Be sure to plot the elements in J2000. Click *OK* then click *Create*. Determine the actual change over one day for:

a) ω 5.425 ______
b) Ω -5.117 ______

Note: This can be accomplished by generating a report and reading off the starting and ending values. Return to *Report & Graph Manager* and duplicate the *Classical Orbit Elements (Report)*. Once again customize the desired parameters as was done for the graph.

Re-propagate the satellite with the Two Body Propagator and look at the resulting graph for the orbital elements to see how the elements are changed.

3. How does the argument of perigee behave and what does this mean?
The argument of perigee does not move. It is not perturbed.

4. What is the inclination (to the nearest tenth) at which the satellite argument of perigee does not experience any J2 effects? (Use the equation for $d\omega/dt$ from Step 1 to calculate this). Change the inclination of the satellite to your calculated value and create a new graph of the classical orbital elements. Validate with a report that your answer is correct.

```
>> i = asind(sqrt(4/5))
i =  63.4
```

Sun Synchronous Satellite

5. Using the inclination calculated in Step 4, change the scenario *Stop Time* to 11 Jul 2007 00:00:00.00 UTCG so that the satellite will propagate for a full 10 days with the J2 propagator. Create a Graph and a Report of the *Beta Angle*. (Beta Angle is the angle between the satellite-sun vector and the orbital plane.) What is the initial and final beta angle? What is the average rate of change (deg/day) in beta angle? Use the report to get exact numbers.

a.) initial Beta Angle -40.245 deg
b.) final Beta Angle -26.472 deg
c.) average rate of change 1.377 deg/day

6. Use this space to calculate the inclination needed for a sun synchronous satellite (due to J2 perturbation) with a semi-major axis of 7000 km and an eccentricity of 0.001:

```
>> p = 7000*(1-0.001^2)
p =     6.999993000000000e+03 km

>> i = acosd(360/365.2421897*2*p^2/(-3*n*6378^2*0.0010826267))
i =  97.874 deg
```

7. Change the properties of your satellite so that it has this inclination and eccentricity, and again propagate the satellite for 10 days. Create a new report. What is the initial and final beta angle? What is the average rate of change (deg/day) in beta angle? How does this compare to the rate computed in step 5? (Note: you won't be able to completely reduce the beta angle rate to zero due to the elliptical orbit of Earth around the sun, and the tilt of Earth's equatorial plane with respect to its orbital plane. However, the beta angle rate of a sun synchronous satellite will be much smaller than that of a non sun synchronous satellite at a similar altitude).

a.) initial Beta Angle -74.714 deg
b.) final Beta Angle -75.681 deg
c.) average rate of change -0.0967 deg/day

8. Use this space to recalculate the inclination for the same elements if you wanted a sun synchronous orbit around Mars instead of Earth. You can get the necessary constants for Mars (i.e. J2, radius, μ, etc.) from STK. Click on *Utilities* -> *Component Browser* -> *Central Bodies*, double-click *Mars*. Under *Gravity Model*, click *Details*. Any other information can be obtained from your textbook.

```
>> n = sqrt(42828.4/7000^3)*24*3600*180/pi
n =     1.749263522822176e+03 deg/day

>> i = acosd(360/686.9150*2*p^2/(-3*n*3396.19^2*0.00195545))
i =     115.717 deg
```

Propagate with Atmospheric Drag/Solar Radiation Pressure/Gravity Model

Create a new satellite in the browser, and call it "Radiation." Set the orbital altitude (perigee and apogee) to 300 km, with an inclination of 28.5 degrees (all other orbital elements zero). Set the orbit epoch to 1 Jul 2007 00:00:00.00 UTCG and the stop time to 6 Jul 2007 00:00:00.00 UTCG (five full days). Using the High Precision Orbit Propagator (*HPOP*), under the *Force Models* button, change the *Area to Mass Ratio* for both the solar radiation model and atmospheric drag model to 0.03 m^2/kg. Change the *Coefficient of Drag (Cd)* to 2.0 and the *Coefficient of Reflectivity (Cr)* to 1.0. Click O.K. Create a report of the classical orbital elements as well as a plot of the semi-major axis. Use the report data to answer the following question.

9. How much does the semi-major axis change in 5 days? -54.74 km

Go back to *Satellite Properties -> Force Models*. Change the *Maximum Degree* and *Maximum Order* of the gravity model to 0 and re-propagate the orbit. This removes the gravity perturbations from the calculations. Graph the semi-major axis.

10. What effect does the gravity perturbations have on **short scale** semi-major axis changes?
They produce sinusoidal motion with the same period of the orbit.

Simulating Mars Pathfinder

Define a new scenario and set the *Epoch* and *Start Times* in *Scenario -> Basic Properties* to 1 Mar 1997 00:00:00.00, and the *Stop Time* to 1 Mar 1998 00:00:00.00. Under *Scenario -> Properties -> Basic -> Animation*, set the *Step Size* to 3600 seconds. Set the following settings under the *Scenario -> Properties Browser -> 2D Graphics -> Global Attributes*:

General: Show Labels - OFF
Vehicles: Show Orbits - ON, Show Orbit Markers - ON, Others - OFF
Planets: Show Orbits - ON, Show Inertial Positions - ON, Others – OFF

Now add two planets using *Insert -> Object Catalog*, Mars and Earth. Under *Basic Properties* for each planet, set the ephemeris source to *DE 421* and the appropriate central body. Create a satellite and name it 'Pathfinder,' under *2D Graphics -> Pass*, set *Orbit Track Lead Type* to *All*. Make sure the time information matches the information above.

Now change the 2-D and 3-D maps so they are *Heliocentric*. Click the 2-D map and then click the *Graphic's Window Central Body* button (globe button on the toolbar) and select the *Sun*. Click on the 2-D map and click the *Properties* icon. Select *Orthographic* as the *Projection Type*, and *CBI* (central body inertial) as the *Displayed Coordinate Frame*. Make the *Display Height* 600 000 000 km and the *Center - Lat* to 84.6 deg (not quite polar). Animate the scenario to watch Earth and Mars orbit the Sun.

To create another map, this one Mars-centered, click *View*, *Duplicate 2-D Graphics – Sun*. Follow the process above to change the central body to Mars, turn off the Lat/Lon lines, and make the projection *Orthographic* and *CBI*, choosing an appropriate *Display Height* (40 000 km is good).

Normally in interplanetary modeling, you would define the initial state as an Earth orbit, such as LEO. But in this example, we are going to start the probe at mid-mission. Use the following information for the initial state in your Mission Control Sequence (remember, this is available after you change to the ***Astrogator*** propagator).

Set up the *Initial State* as follows:

Coord. System – Sun J2000
Element Type – Keplerian
Orbit Epoch – 1 Mar 1997 00:00:00.00 UTCG

a = 193216365.381 km	Ω = 0.258 deg
e = 0.236386	ω = 71.347 deg
i = 23.455 deg	ν = 85.152 deg

Now change the *Propagate* sequence, making the propagator *Heliocentric*. Click the *Advanced* button and turn OFF the *Maximum Time Propagation*. Make *Periapsis* the *Stopping Condition*, and in the *Central Body* field, substitute *Sun* for *Earth*. Run the sequence using the green arrow at the top left of the window.

11. On what day was the probe's closest approach to Mars? (Use both 2-D Graphics windows) 4 Jul 1997

To stop near Mars instead of swinging right by it, return to the Propagate segment and, in the Central Body field, substitute Mars for the Sun. Set the *Coordinate System* under *Segment Properties* to *Mars Inertial*. Run the scenario again. Use the result button and the summary icon (near the *Run Sequence* button) to obtain the distance from Mars.

12. At the end of the transfer what are the following values (wrt Mars):

Altitude 3713.414 km
Radius 7105.128 km
Date 4 Jul 1997 17:29:18.125 UTC