MAE 341: Spaceflight
Prof. Edgar Choueiri
Princeton University
Fall 2025
Homework Project # 2

Due on **Wednesday, November 12, 2025.**

Part 1: FIRST EXPLORATIONS WITH STK

1 Introduction

In this problem set you will carry out a number of exercises using the STK software and report your findings. The exercises will allow you to apply many of the concepts you have learned so far using a sophisticated tool. You will get to visualize many of the features of various space missions the same way astronautical engineers do in real situations.

If you are unfamiliar with STK, it is recommended that you explore the Beginner Tutorials, which guide you through the basics of using the software. The first five beginner tutorials should equip you with adequate skills to complete the homework. Some of these skills will also be reviewed during the precept.

Many of the terms used here were defined and discussed in detail in the lecture notes. A few of them are defined again below in the context of the STK software. A glossary of orbital terms is given in the appendix .

1.1 Some Relevant Remarks

Here are some comments on the method used in STK to evolve (propagate) an orbit subject to perturbations.

An integrator uses numerical integration of the forces on the satellite to compute the satellite trajectory. It is not possible to compute a single position and velocity at an arbitrary time using numerical integration. The position and velocity at a given time are obtained by "propagating" the orbit starting from initial conditions. You can think of the Gaussian-Lagrange equations (discussed in Section 5 of the third set of lecture notes) as one type of propagator where the orbital parameters are calculated at every step of the orbit as a fucntion of force models and the osculating orbital elements. In this fashion the orbit is "propagated". STK uses a more sophisticated propagator than the one we discussed in class.

A propagator only yields accurate results when used with the initial conditions generated for that propagator. HPOP is a High Precision Orbit Propagator that you will be using in STK. It can accept input in any of the coordinate types and systems available in STK; values entered in a system different from the J2000 Cartesian State Vector are converted. HPOP includes modern, high-fidelity models for all of the major perturbations affecting an Earth satellite:

- Joint Gravity Model (JGM) 2
- Lunar/solar point-mass gravitational effects
- Atmospheric drag
- Solar radiation pressure

HPOP also takes into account all of the major predictable motions of the Earth that affect the apparent position of the satellite:

- Precession of the equinoxes
- Nutation
- Diurnal rotation
- Barycentric displacement

In addition, the HPOP accounts for the differences among the three primary astronomical time systems:

- Universal Time Coordinated (UTC)
- International Atomic Time (TM)
- Terrestrial Dynamic Time (TDT)

All input and output are expressed in terms of UTC; TM and TDT are used internally to achieve increased accuracy.

2 Orbits

2.1 General Orbital Terms

Note: The orbit starting point is at the True Anomaly, which is measured relative to the Argument of Perigee. The Argument of Perigee is measured relative to the Ascending Node. Therefore, if the Longitude of the Ascending Node is -50.0°, the Argument of Perigee is 0.0°, and the True Anomaly is 180.00°, the first point of the propagated orbit is right at the descending node.

2.1.1 Orbit Size and Shape (includes the first 2 fields)

The first two fields are linked. If you choose one, the other element also appears. Choose one of the following pairs:

- **Semimajor Axis/Eccentricity (default):** Semimajor Axis is half the distance between the two points in the orbit that are farthest apart. Eccentricity describes the shape of the ellipse (where 0= a perfectly circular orbit and 1 = a parabolic path).

- **Apogee Radius/Perigee Radius:** Measured from the center of the Earth to the points of maximum and minimum radius in the orbit.

- **Apogee Altitude/Perigee Altitude:** Measured from the "surface" of the Earth (a theoretical sphere where the radius equals the equatorial radius of the Earth) to the points of maximum and minimum radius in the orbit.

- **Period/Eccentricity:** A Period is the duration of one orbit, based on assumed two-body motion.

- **Mean Motion/Eccentricity:** Mean Motion identifies the number of orbits per solar day (86400 sec/24 hour), based on assumed two-body motion.

2.1.2 Orbit Orientation (Includes the next 3 fields)

- **Inclination:** The angle between the angular momentum vector (perpendicular to the plane of the orbit) and the inertial Z axis.

- **Argument of Perigee:** The angle from the ascending node to the eccentricity vector (lowest point of orbit) measured in the direction of the satellite's motion. The eccentricity vector points from the center of the Earth to perigee with a magnitude equal to the eccentricity of the orbit. The last of three elements describing orientation allows you to specify the orientation of the intersection of the orbital and equatorial planes. Choose either:
  - **Right Ascension of the Ascending Node (default):** Specifies the angle from the inertial X axis to the ascending node measured in a right-handed sense about the inertial Z axis in the equatorial plane.
  - **Longitude of the Ascending Node:** The Earth-fixed longitude where your vehicle crosses the inertial equator (the intersection of the ground track and the inertial equator) from south to north. The specified ascending node crossing is assumed to be at, or prior to, the initial condition of the orbit.

2.1.3 Vehicle Location

The pop-up menu for this element allows you to specify a vehicle's location within its orbit at epoch:

- **True Anomaly (default):** Specifies the angle from the eccentricity vector (Argument of Perigee) to the satellite position vector, measured in the direction of satellite motion.

- **Mean Anomaly:** The angle from the eccentricity vector to a position vector where the satellite would be if it were always moving at its average angular rate.

- **Eccentric Anomaly:** An angle measured with an origin at the center of an ellipse from the direction of perigee to a point on a circumscribing circle from which a line perpendicular to the Semimajor Axis intersects the position of the satellite on the ellipse.

- **Argument of Latitude:** The sum of the True Anomaly and the Argument of Perigee. Used for nonequatorial orbits only.

- **Time Past Ascending Node:** The elapsed time since the last ascending node crossing.

- **Time Past Perigee:** The elapsed time since the last perigee passage

2.2 Exercise 1: Creating Low Earth Orbit (LEO) Vehicle

2.2.1 Exploration with STK

1. Create a new scenario. Set the start time to noon UTC on the 1st of November 2019, and the end time to noon on the 5th of November 2019.

2. In the Insert STK Objects window select a satellite object (or from the menus go to Insert → New...), and under Select a Method choose the Orbit Wizard. Click Insert.

3. In the pop up window, choose a circular orbit with an inclination of 45 degrees and a 300 km altitude. Leave RAAN (Right Angle of Ascending Node) zero. Leave the Analysis Time Period at that of the entire Scenario (default setting). Click OK.

4. The ground trace and 3-D image of the new satellite is now displayed on the map and the Earth viewer, respectively. Animate the displays using the blue ‘play’ button on the toolbar. Here you can also increase or decrease the speed of the animation, and choose the time you are looking at using the slider.

5. Right-click on your satellite in the Objects Browser (to the left of the screen) and choose Properties Browser. The Propagator shown in the first pane is J4Perturbation.

6. Without closing the Properties Browser, in the menu bar, go to Analysis → Report & Graph Manager. This manager allows you to graph many quantities. Click on the Create New Graph Style icon and name your new graph Orbital Radius. The Graph Style window should pop up. Call the time axis Time. On the right, scroll down to Spherical Elements, expand it, and then expand Fixed. Select Radius, and add it to the Y-axis box. Now hit OK.

7. Hit Generate. Notice how the radius of the orbit stays constant throughout the satellite's lifetime. A good circular orbit. Note that if you want to change the colour of your graph, or any other property, you can right click on your Orbital Radius graph style under the Report & Graph Manager and select Properties to modify.

8. Back in the satellite Properties Browser, change the Propagator to HPOP. Choose Force Models, and notice that now atmospheric drag, solar pressure, and a gravity model for the Earth, Sun and Moon are now included. Leave the settings as they are. In the Properties Browser, hit Apply.

9. Now return to your Orbital Radius plot (or recreate it). Notice the differences. Use the Force Models dialog to turn drag and solar pressure on and off, incease the coefficient of drag C_D to 10.0. Explore a few different scenarios and copy plots into your homework report, be sure to note which force models are contributing to which behavior in the Orbital Radius plot.

2.2.2 Additional Comments on LEO orbits

We supplement the discussion we had in the class notes on LEO orbits with the following comments.

Low Earth Orbit (LEO) LEOs are either elliptical or (more usual) circular orbits at a height of less than 2,000 km above the surface of the earth. The orbit period at these altitudes varies between ninety minutes and two hours. The radius of the footprint of a communications satellite in LEO varies from 3000 to 4000 km. The maximum time during which a satellite in LEO orbit is above the local horizon for an observer on the earth is up to 20 minutes. A global communications system using this type of orbit, requires a large number of satellites, in a number of different, inclined, orbits. When a satellite serving a particular user moves below the local horizon, it needs to be able to hand over the service to a succeeding one in the same or adjacent orbit. Due to the relatively large movement of a satellite in LEO with respect to an observer on the earth, satellite systems using this type of orbit need to be able to cope with large Doppler shifts. Satellites in LEO are also affected by atmospheric drag which causes the orbit to gradually deteriorate.

Many LEO satellites are used for remote sensing as discussed in Section 3 of the third set of lecture notes. They are also used in constellations of a large number of communication satellites. Examples of "Big LEO" systems are GlobalstarTM (48+8 satellites in 8 orbital planes at 1400 km), and IridiumR (66+6 satellites in 6 orbital planes at 780 km). The new SpaceX Starlink satellite constellation is leading a paradigm shift in LEO constellations, with thousands of small satellites being places in orbit to provide continuous internet coverage to the globe. Such mega-constellations will provide challenges for reducing orbital debris in the future.

2.3 Exercise 2: Creating a Medium Earth Orbit (MEO) Vehicle

2.3.1 Exploration with STK

1. In the same scenario as before, create a new satellite (Insert → New...). Now you will have to keep up with what the satellites are named. You can rename them by right-clicking them in the Object Browser.

2. In the Orbit Wizard, again create a circular orbit, this time with a 15 degree inclination and an altitude of 10,000 km. Again use the default time period.

3. You will now see the MEO satellite on both ground-track and 3-D displays. (You can zoom in and out of the 3-D display by scrolling or holding the right mouse button, and you can rotate the view by holding the left mouse button.)

4. Enter the Properties Browser of your MEO satellite, and change the propagator from J4Perturbation to HPOP. Notice the relative lack of orbit degradation—atmospheric drag becomes much less important way up here.

5. Now, having selected your MEO satellite in the Object Browser, choose Analysis → Access... from the menu. Under the list of objects on the left, choose your LEO satellite, and hit Compute. Now animate your scenario (play button), and watch as the two satellites can "see" each other, as the animations indicate with a line joining the two. During these periods, the satellites can directly communicate. Notice that there are periods when this is not possible.

6. Analysis → Report & Graph Manager, select your MEO satellite on the left, and on the right, under Installed Styles, select Lightning Times. Click Generate. This will show you when the satellite is in the sun and when it is not—that is, when your solar panels will be charging the satellite . This information becomes important when sizing the batteries that the spacecraft must carry to make it through the night. Same this plot and put it in your homework report.

2.3.2 Additional Comments on MEO orbits

Intermediate Circular Orbits (ICO), or Medium Earth Orbits (MEO) ICOs are circular orbits at an altitude of around 10,000 km. Their orbit period measures about 6 hours. The maximum time during which a satellite in LEO orbit is above the local horizon for an observer on the earth is in the order of a few hours. A global communications system using this type of orbit, requires a modest number of satellites in 2 to 3 orbital planes to achieve global coverage. ICO satellite are operated in a similar way to LEO systems. However, compared to a LEO system, hand-over is less frequent, and propagation delay and free space loss are greater. Examples of ICO systems are Inmarsat-P (10+2 satellites in 2 inclined planes at 10355 km), and Odyssey (12 + 3 satellites in 3 inclined planes, also at 10355 km).

2.4 Exercise 3: Creating a Highly Elliptical Orbit (HEO) Vehicle

2.4.1 Exploration with STK

1. In the same scenario, create a new satellite. In the orbit wizard, choose Molniya. Leave the Apogee Longitude and Perigee Altitude as they are. Note those are the only two parameters you need to define; the definition of Molniya sets the rest. Pull up the Properties Browser for your Molniya satellite and examine the orbital elements.

2. Observe the ground tracks. Specifically, convince yourself of why this orbit would have been useful to the USSR for spy satellites during the Cold War—especially since the USSR owned territory at the high latitudes necessary to launch a Molniya, while the US did not.

3. Compute the access for your Molniya satellite with the other two satellites. Notice now that there are times when the Molniya satellite can be used as a bridge between the other two when there is no line-of-sight communication path.

4. Change the propagator to HPOP and apply the change. Under 2D Graphics → Pass, change the Ground Track: Lead Type to All. This shows you the entire ground tracks for the total simulation time and you can graphically observe the precession of the line of nodes (to declutter your ground tracks plot you might want to clear the Access calculations from the Access panel).

5. Back under Basic → Orbit (on the left hand menu) we can modify the orbital parameters. Select the first parameter, Semimajor Axis, and change it to Apogee Altitude. Set the Apogee Altitude to 40,000 km and the Perigee Altitude to 1,000 km. Set the Inclination to 40 degrees. Use the rest of the defaults and click on Apply to propagate and observe the Ground Track.

6. Look at the ground track, comparing it to that of the former Molniya orbit and write a few sentences in your homework report explaining the features you observe.

2.4.2 Additional Comments on HEO orbits

HEOs were discussed in Section 6 of the third set of the lecture notes. They typically have a perigee at about 500 km above the surface of the earth and an apogee as high as 50,000 km. The orbits are inclined at 63.4 degrees in order to provide communications services to locations at high northern latitudes. This particular inclination value is selected in order to avoid rotation of the lines of apsides, i.e. the intersection of a line from Earth center to apogee and the Earth surface will always occur at a latitude of 63.4 degrees North as discussed in the lecture notes. Orbit period varies from eight to 24 hours. Owing to the high eccentricity of the orbit, a satellite will spend about two thirds of the orbital period near apogee, and during that time it appears to be almost stationary for an observer on the earth (this is referred to as apogee dwell). After this period a switch-over needs to occur to another satellite in the same orbit in order to avoid loss of communications. Free space loss and propagation delay for this type of orbit is comparable to that of geostationary satellites. However, due to the relatively large movement of a satellite in HEO with respect to an observer on the earth, satellite systems using this type of orbit need to be able to cope with large Doppler shifts (frequency shifts due to the speed of the satellite). Examples of HEO systems are:

- The Russian Molniya system, which employs 3 satellites in three 12 hour orbits separated by 120 degrees around the earth, with apogee distance at 39,354 km and perigee at 1000 km;
- The Russian Tundra system, which employs 2 satellites in two 24 hour orbits separated by 180 degrees around the earth, with apogee distance at 53,622 km and perigee at 17,951 km;
- The proposed Loopus system, which would employ 3 satellites in three 8 hour orbits separated by 120 degrees around the earth, with apogee distance at 39,117 km and perigee at 1,238 km;
- Ehe European Space Agency's (ESA's) proposed Archimedes system. Archimedes would employ a so called "M-HEO" 8 hour orbit. This produces three apogees spaced at 120 degrees. Each apogee corresponds to a service area which could cover a major population center, for example the full European continent, the Far East and North America.

Another example of HEO (and sometimes LEO) orbits is the Earth-synchronous orbit discussed in Section 5 of the third set of lecture notes. A geosynchronous is any type of orbit which produces a repeating ground track. This is achieved with an orbit period approximately an integer multiple or sub-multiple of a sidereal day (NOTE: The word "approximately" is used because there is a need to correct for the node regression due to the precession of the satellite orbit).

2.5 Exercise 4: Creating a Geosynchronous Orbit (GEO) Vehicle

2.5.1 Exploration with STK

1. In the same scenario, create a new geosynchronous satellite. Make the longitude of the subsatellite point -75 degrees.

2. Make sure the focus is on the 3-D Graphics window. Now, in the pane just to the right of the Object Browser, hit the icon that looks like an eye with an arrow pointing to it (Title: View From/To). In View From, choose your GEO satellite; in View To, choose Earth. Animate the scenario, and watch the other satellites fly through the field of view, while the GEO stays focused on South America.

3. Change the propagator to HPOP. Now animate the scenario, and watch (intently) the ground track. Near the end of the five days of simulation, you can see a small North-South wobble. This wobble didn't appear when using the default propagator.

4. Now change the orbit's inclination to 4 degrees. Animate the scenario again, and watch as the Earth does a sickening swaying motion in the 3-D display. You can also now see the figure-8 of the orbit on the ground track. (If you can't see it, it will become more distinct if you change the inclination to a higher value.)

5. Create a new Facility (in the same window as creating a new satellite) by selecting Facility and Define Properties (on the right). Place it at -117 deg 08 min longitude, 32 deg 49 min latitude (in San Diego). Calculate the access that each satellite has with the Facility. These are the times when data can be downloaded, commands can be uploaded, etc. Make a note of how easy communications is for some satellites, and how infrequent it is for others.

2.5.2 Additional Comments on GEO orbits

The geostationary or geosynchronous orbit (GEO) was discussed in Section 5 of the third set of lecture notes. It is a circular prograde orbit in the equatorial plane with an orbital period equal to that of the Earth, which is achieved with an orbital radius of 6.6107 (Equatorial) Earth Radii, or an orbital height of 35786 km. A satellite in a geostationary orbit will appear fixed above the surface of the Earth. In practice, the orbit has small non-zero values for inclination and eccentricity, causing the satellite to trace out a small figure of eight in the sky. The footprint, or service area of a geostationary satellite covers almost 1/3 of the Earth's surface (from about 75 degrees South to about 75 degrees North latitude), so that near-global coverage can be achieved with a minimum of three satellites in orbit. The disadvantage of a geostationary satellite in a voice communication system is the round-trip delay of approximately 250 milliseconds. However GEOs are very useful for one-way communication and data transfer.

2.6 Exercise 5: Illustrating The Drag Paradox

As a fun illustration of the drag paradox, do the following:

1. Create a new scenario (you'll probably want to save the last one you were using). Choose a time frame which will give you a day of propagation.

2. Add two identical satellites. Give them circular orbits, 300 km altitude, and leave the rest of the settings as defaults. When you animate the scenario, the two satellites will of course be right on top of each other.

3. Now change the propagator of one of the two satellites to HPOP, and under Force Models, change the Area/Mass ratio to 1 m^2/kg. Click Apply.

4. Animate the scenario again, and watch as the satellite that experiences drag pulls head of the other, moving faster as its orbital radius decreases. It eventually crashes South-East of Africa.

5. Use the Radius graph style that you created long ago to observe the manner in which the orbit decays. Also compare the Classical Orbital Elements graphs of the two satellites (under Installed Styles). Include these plots in your report.

PART 2: Further Explorations with STK

3 Orbit Transfers

In the last homework assignment you simulated the orbits of various satellites and experimented with force models to see how perturbations affected the orbits. What you could not do was move between orbits or calculate ΔV for such maneuvers.

The STK propagator known as Astrogator gives the user the capability of doing just that. It allows you to first dictate a beginning orbit; then to execute a maneuver; and see what orbit that maneuver takes you to. In executing a maneuver, you can either calculate the ΔV to be applied first by hand, and feed it into the software, or have the software iteratively calculate it for you.

Start this section by completing the STK Astrogator tutorial on the Hohmann Transfer. This will walk you through planning a Hohmann transfer, in which you use known (or calculated) ΔV's to model the maneouver.

Continue by completing the tutorial on the Hohmann Transfer Targeter, which will walk you through how to get STK to perform the Δv calculations for you. This will be critical to accomplishing the following excercises.

3.1 Hohmann Transfer

Plan a Hohmann transfer for a satellite from LEO at 500 km altitude to a final altitude of 96,792 km (far above GEO). [The Targeter tutorial directs you to do just such a transfer to an orbit near GEO, so to get the hang of it you can just follow the steps while inserting your own numbers. When in doubt about what values to use for particular parameters, you may use the values from the tutorial.]

As you build your mission using Astrogator, you can look at a summary for each step of the sequence by choosing the "Summary" button in the Basic Orbit window for your satellite. In particular, for each maneuver of the mission you can look up the ΔV that Astrogator has calculated for you, and the date/time when that maneuver was accomplished.

1. What is the total ΔV for the Hohmann transfer of your simulation?

2. How long did the Hohmann transfer take?

3.2 Bielliptic Transfer

Along the same lines, create another satellite in your scenario and now plan a bielliptic transfer to get the satellite between the same two orbits. Use as the apogee radius of the intermediate transfer ellipse twice the radius of the final orbit. [Hint: You will need to think carefully about how you set the constraints on your second δv maneuver.]

1. What is the total ΔV for the bielliptic transfer of your simulation?
2. How long did the bielliptic transfer take?

Comment on the tradeoffs between the Hohmann and bielliptic transfers. As a mission planner, which is more appealing to you?

3.3 Inclination Change

Now plan a mission in which a satellite in a 500 km equatorial orbit executes a maneuver to enter a 500 km polar orbit [Hint: Carefully think about which directions of thrust are required for this maneuver.].

1. What is the ΔV of this maneuver? Comment on the size in comparison to typical ΔV magnitudes that you are familiar with.

3.4 Sphere of Influence

This problems explores the validity of the assumption that the radius of the sphere of influence for a given planet is

$$r_{SOI} = \left(\frac{M_{planet}}{M_{sun}}\right)^{2/5} r_{planet} \tag{1}$$

where r_{planet} is the mean distance of the planet from the sun. In order to investigate this limit, let us set up STK in the following way.

1. Create a new scenario titled **SOI**. Set the Start Time to 15 Nov 2019 00:00.000 and End Time to 15 Nov 2021 00:00.000.
2. Insert a new planet in the scenario and rename it **Earth** (Insert → Default Object...). Open the properties of **Earth** and make sure the central body is set to Earth.

3. In the 3D viewer, change the 3D Graphics Window's Central Body to the **Sun** (smalle blue globe with an arrow pointing down).

4. Open the properties of the 3D viewer (top left). Under the Grid property, change tick Show under Ecliptic Coordinates. Under the Advanced property, change the Max Visible Distance to 1e+009 km.

5. Reorient the view so the Sun and Earth are visible.

6. Open the Scenario properties. Under2D Graphics click on Global Attributes check Show Orbits in the Planets section and deselect Show Subplanet Points and Show Subplanet Labels. Under vehicles, deselect Show Ground Tracks. When you apply these changes, you should be able to see the Earth's orbit in the 3D viewer.

7. Insert two satellites into the scenario called **SAT1** and **SAT2**.

8. Open the properties of **SAT1** and change the propagator to Astrogator.

9. Click on Initial State and set the Coordinate type to Target Vector Outgoing Asymptote. Set the radius of periapses to 7000 km, C3 to 60 km^2/sec^2, RA of Outgoing Asymptote to 0 deg, Velocity Azimuth at Periapsis to 0 deg, and the True Anomaly to 0 deg.

10. Under Propagate stage, set the Propagator to Heliocentric and set the trip time to 3 years. Under the Advanced tab, uncheck Maximum Propagation Time. Run the simulation by selecting the Green Arrow and observe the resulting orbital track.

11. For the second satellite, **SAT2**, follow the same steps as above from 8 to 10. However, change the propagator in the Propagate state to Earth Point Mass. Set the Stop Condition to R Magnitude. Set the value to 50000 km. Deselect or remove the other Stop Condition.

12. Create a second Propagate stage.

13. Under the Utilities menu (at the top) select Component Browser. Under Propagators, select Earth Point Mass and click duplicate. Rename the new propagator to Sun and set the Central Body to the Sun (double click on the line in the Description column).

14. Return to the Second Propagate stage of **SAT2**, set the Propagator to sun. Set the propagation time to three years and deselect the Maximum Propagation Time (under advanced).

15. Change the color of the propagating orbits to a different color from **SAT1**. Run the scenario by pressing the green arrow and observe the result. Include an image of the resulting orbits in your homework.

What we have done is to display the propagation of the satellite under two different scenarios. In the first case, **SAT1**, all gravitational effects in the solar system are taken into account by using the Heliocentric propagator. In the second case, the satellite trajectory is numerically calculated out to a certain radius from the earth (50000 km) assuming the earth is a point mass. At this point, the trajectory is calculated assuming the sun is a point mass and the only object. It should be clear that the orbits do not coincide.

Questions

1. Generate a graph for Radius from sun versus time for **SAT1** and **SAT2** and comment on the discrepancies in orbit.

2. Increase the value of R Magnitude in the Stop Condition for **SAT2** to 200,000 km and 10^7 km respectively and generate the SAT1, SAT2 Radius plots. Comment on the discrepancies.

3. Calculate r_{SOI} and substitute this value for R Magnitude in the Stop Condition and generate the Radius plot. Comment on the relative discrepancy of the orbits.

4. 10^7 km is well-outside the SOI of earth. Why is the discrepancy larger for this value than for r_{SOI}?

The Astrogator is a powerful tool for planning much more complicated missions than the ones in this assignment. Explore the possibilities further by looking into some of the other tutorials and/or playing around with the things you've learned so far.

4 APPENDIX: Glossary

Useful Glossary of Orbital Terms

Many of the terms in this glossary were discussed in more details in the lecture notes.

Apoapsis The point of an elliptical orbit that is farthest away from the gravitational center of the system consisting of the primary body and the satellite (In Earth-based systems, the apoapsis is called apogee).

Apogee The point in the satellite orbit that is farthest from the gravitational center of the earth.

Apogee Altitude, or Apogee Height The altitude of Apogee above a specified reference point serving to represent the surface of the earth.

Apogee Kick Motor (AKM) Motor, used once during the lifetime of a geostationary satellite to provide the large delta-v required to turn a highly elliptical orbit with apogee at the geostationary altitude into a circular, geostationary orbit. Apogee Kick Motors are needed because many launchers are not able deliver a satellite into geostationary orbit (the Russian Proton launcher is an exception to this rule).

Apsis One of the extreme end-points of the major axis of an elliptical orbit (apogee and perigee are apses).

Argument of the Perigee (ARGP) Angle in the plane of the satellite orbit between the ascending node and the perigee, measured in the direction of the satellite motion.

Ascending Node Point in the equatorial plane where the satellite crosses through the equatorial plane in a northerly direction.

Atmospheric Drag Slowing down force acting on a satellite due to the Earth's atmosphere. Below 160 km height, the atmosphere causes a satellite orbit to decay (spiral down) within a few revolutions. Above 700 km, drag has hardly any influence. Angle in a meridian, measured northward from the ecliptic to a particular direction (direction defined as the line from the earth center to a particular celestial object).

Delta-v Speed change needed for a particular change in orbit parameters. The direction and size of the delta-v determines which orbit parameters are most affected, and by how much. For instance, a delta-v, orthogonal to the orbit plane at the time of ascending node or descending node crossing, results in an inclination change (this is a maneuver which requires a relatively large amount of propellant. Tight inclination control therefore limits the lifetime of a satellite in orbit considerably).

Descending Node Point in the equatorial plane where the satellite crosses through the equatorial plane in a southerly direction.

Drift Orbit A new geostationary satellite is usually delivered in an orbit which is slightly higher or lower than its final orbit. It then appears to drift slowly towards its final location. The satellite may be halted temporarily (using a Hohmann Transfer) at a different location to allow it to be tested without causing interference, after which it is drifted again to its final location.

Eccentric Anomaly ϵ Auxiliary angle which is used in the integration of Newton's equations for elliptical motion. E is the angle between the main axis and the line from the center of the ellipse to a point Q on the circle which has been circumscribed about the ellipse (See figure 7 of the third set of lecture notes). The point Q is a projection of the satellite along a line which is parallel to the minor axis of the ellipse. The angle ϵ appears in the famous Kepler's Equation (Equation 65 of the third set of notes).

Eccentricity e Constant defining the shape of the orbit. Discussed in details in the lecture notes.

Eclipse Passage of a satellite through the Earth's shadow. For a geostationary satellite there are two periods in the year of about 40 days duration, centered around vernal and autumnal equinox, when a satellite passes through the shadow of the Earth once every day. A communications satellite needs to be equipped with batteries in order to avoid interruptions of traffic during eclipse.

Ecliptic The plane of the earth's revolution around the sun.

Ephemeris Time (ET) Measurements with highly accurate atomic clocks show that the rotation period of the Earth is slightly irregular. Ephemeris Time is introduced to remove the dependence on the Earth rotation, and is calculated from the observed motion of the Moon. In practice, differences between the rates of ET and Universal Time (UT) may be neglected. The absolute difference has increased over the last 100 years to about 60 seconds.

Epoch Date and time chosen as the reference date/time from which time is measured. A set of orbital elements is valid for a specified epoch.

Equation of the Center Relation between True and Mean Anomaly, used as a first approximation to Kepler's Equation.

Equation of Time Difference between the Mean Solar Time and the real solar time. This varies between a minimum of -15 and a maximum of

+15 minutes during the year.

Equinox Moment at which the Sun as viewed from the Earth appears to cross the celestial equator. This occurs at about 21 March - the vernal equinox - and at about 22 September - the autumnal equinox.

Figure of the earth The shape of the Earth can be approximated by a spheroid of revolution, i.e. a geometrical shape in which any cross-section parallel to the equator is a circle, and any cross-section through the north-south axis is an ellipse of which the minor axis coincides with the Earth's axis.

Footprint of a Satellite Point on the surface of the Earth, directly below a satellite. The footprint is the intersection of the Earth's surface and the line connecting the center of the Earth and the satellite.

Footprint That portion of the earth's surface from which the elevation angle towards a satellite exceeds a specified value (usually 0, 5, or 10 degrees).

Free-Space Loss Signal attenuation that would occur on a link between an isotropic antenna on the surface of the earth and an isotropic antenna on-board a satellite in the absence of any propagation effects such as atmospheric absorption, diffraction and obstruction.

Geosynchronous Orbit Circular prograde orbit in the equatorial plane of the earth with an orbital period of exactly one sidereal day. The radius of a geostationary orbit is 6.6107 (equatorial) earth radii. Greenwich Mean Time (GMT) Mean Solar Time at the meridian of Greenwich, England, formerly used as a basis for standard time. GMT is replaced by Universal Time (UT).

Hohmann Transfer Transfer between two circular coplanar orbits via an intermediate elliptical orbit of which the perigee is tangent to the smaller circle and the apogee is tangent to the larger circle. A Hohmann Transfer is the most economical transfer from the standpoint of $\Delta - v$, i.e. amount of propellant required. It also takes longer than any other possible transfer orbit. Note that this type of transfer requires two burns, a start and a stop burn.

Hour Angle Angle measured westward from the observer's meridian to the meridian that contains the direction to a celestial object.

Inclination of an Orbit, i The angle between the plane of the Earth's equator and the plane containing the orbit (counted positively in the direction towards the North Pole).

Inclined Orbit Any non-equatorial orbit of a satellite.

Julian Date (JD) The sequential day count reckoned consecutively beginning on 1 January 4713 BC (The Julian Date for 1 January 1990 was

2,446,892).

Keplerian Elements A set of six parameters which together describe shape and orientation of an elliptical orbit around the earth, as well as the position of a satellite in that orbit at a given epoch. The usual elements are: Right Ascension of the Ascending Node, Argument of the Perigee, Mean Anomaly, Semi-Major Axis, Inclination and Eccentricity.

Line of Nodes Line of intersection of the equator and the orbital plane. This line goes through the ascending and descending nodes.

Major Axis of an ellipse a the longest diameter of the ellipse which goes through the center of both focal points.

Mean Anomaly, M Angle, measured from the periapsis in the direction of the satellite's motion, which a satellite would sweep out if it moved at a constant angular speed, i.e. $M = 2\pi t/\tau$ (radians), where τ is the orbital period.

Mean Motion n The average angular velocity of a satellite in an elliptical orbit, i.e. $n = 2\pi/\tau$ (radians/second), where τ is the orbital period.

Mean Solar Time or Universal Time (UT) time measured with respect to the motion of a fictitious body called the Mean Sun which moves at a constant rate (another way to state this assumption is that the earth moves in a circular orbit around the sun, and that the axis of rotation is perpendicular to the orbital plane (ecliptic). The time interval between two meridian crossings of the Mean Sun is exactly one solar day. Due to the combined effects of the eccentricity of the Earth's orbit and the tilt of the Earth rotation axis, the real sun arrives at our local Meridian a little early at certain times of the year, and a little late at other times. The difference between real solar time and mean solar time is called the Equation of Time.

Measurements with highly accurate atomic clocks show that the rotation period of the Earth is slightlirregularar. Ephemeris Time (ET), introduced to remove the dependence on the Earth rotation, is calculated from the observed motion of the Moon. In practice, differences between the rates of ET and UT may be neglected. The absolute difference has increased over the last 100 years to about 60 seconds.

Node Rotation The non-spherical shape (oblateness) of the Earth causes a rotation of the orbital plane. This precessional notion is similar to that of a simple top: the normal to the orbital plane sweeps out a cone shaped urface in space with a semi-vertex angle equal to the inclination i. As the orbit precesses, the line of intersection of the equator and the orbital plane (the line of nodes) rotates westward for a prograde orbit and eastwards for a retrograde orbit. This effect is known as node rotation.

Osculating Orbit Orbit along which a satellite would move if all perturbing accelerations would be removed at a particular ime. At that time, or Epoch, the osculating and true orbits are in contact. (Orbit parameters are always given for the osculating orbit, because the true, perturbed, orbit cannot be described this way).

Periapsis The point of an elliptical orbit that is closest to the gravitational center of the system consisting of the primary body and the satellite (In Earth-based systems, the periapsis is called perigee).

Perigee The point in the orbit of a satellite orbiting the earth which is closest to the gravitational center of the earth.

Perigee Altitude or Height The altitude of Perigee above a specified reference point serving to represent the surface of the earth.

Period τ Time for a satellite to complete one revolution around the center of gravity.

Perturbation Deviation from true elliptical motion of a satellite caused by disturbing accelerations due to the non-spherical shape of the Earth , influence of sun and moon, drag and sun radiation pressure.

Precession Rotation of the orbital plane caused by the non-spherical shape (oblateness) of the Earth. Precessional motion is similar to that of a simple top: the normal to the equatorial plane sweeps out a cone shaped surface in space with a semi-vertex angle equal to the inclination i. As the orbit precesses, the line of intersection of the equator and the orbital plane (the line of nodes) rotates westward for a prograde orbit and eastwards for a retrograde orbit.

Prograde Orbit Orbit of a satellite orbiting the earth, in which the projection of the satellite's position on the (Earth's) equatorial plane revolves in the direction of the rotation of the Earth. (the orbit inclination of a satellite a prograde orbit is less than 90 degrees).

Retrograde Orbit Orbit of a satellite orbiting the earth, in which the projection of the satellite's position on the (Earth's) equatorial plane revolves in the direction opposite to that of the rotation of the Earth. (the orbit inclination of a satellite in a retrograde orbit is in excess of 90 degrees).

Right Ascension Angle in the plane of the equator, measured eastward from the vernal equinox direction to a particular meridian, e.g. the meridian which contains the vector pointing in the direction of the satellite.

Right Ascension of the Ascending Node (RAAN) Angle in the equatorial plane between the direction to the vernal equinox and the direction to the ascending node, measured counter-clockwise when viewed from the north side of the equatorial plane.

Round-trip Delay Time The time required for a signal to travel from an earth station via a satellite to another earth station (the round-trip delay for a geostationary satellite is approximately 250 milliseconds).

Semimajor Axis of an Ellipse The longest diameter of the ellipse which goes through the center and both focal points, is called the major axis. The portion from the center of the ellipse in either direction is the Semimajor Axis.

Sidereal Time Time required for the Earth to rotate once on its axis relative to the stars. This occurs in 23h 56m 4s of ordinary mean solar time. A sidereal day consists of 24 sidereal hours. The sidereal day starts when the Vernal Equinox crosses the Greenwich meridian. Sidereal time is therefore equal to the Hour Angle of the Vernal Equinox.

True Anomaly, θ The angle in the plane of the satellite orbit between the periapsis and the satellite position, measured in the direction of the satellite's motion.

Vernal Equinox Direction Direction towards a point in the constellation of Aries. On the first day of spring, a line joining the center of the Earth and the center of the sun points in this direction. This line is the intersection of the earth's equatorial plane and the ecliptic plane, which is the plane of the earth's revolution around the sun. The vernal equinox direction is used as the x-axis for an astronomical reference system (if extreme precision is needed, it would be necessary to specify that the reference frame is based on the vernal equinox for a particular epoch).

Universal Time (UT) Local mean solar time on the Greenwich meridian, also called Greenwich Mean Time (GMT), or Zulu Time (Z).