



Case Study

Modeling Satellite Maneuvers

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Ansys Software Used

Ansys Systems Tool Kit (STK®), part of Ansys Digital Mission Engineering Software

Summary

This case study demonstrates some basic and advanced concepts in modeling various satellite maneuvers using the Ansys STK Astrogator tool. All maneuvers investigated are in the context of satellites orbiting the Earth as a central body.

All modeling and simulation in this case study is done using Ansys Systems Tool Kit (STK®), part of Ansys Digital Mission Engineering Software. Ansys STK software allows engineers to model complex systems inside a realistic and time-dynamic three-dimensional simulation. It is the industry standard for designing and developing satellite missions. Integration with other Ansys products, such as Ansys Electronics Desktop, provides an avenue to bring component and subsystem design and models into an integrated system-level simulation environment.

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1. Introduction to Satellite Maneuvers

Satellite, or orbital, maneuvers can be defined as the purposeful change of the orbital parameters of a spacecraft. This is in contrast to orbital perturbations, which are due to the effects of non-spherical Earth and other gravitational influences.

These changes in satellites' orbit can be due to multiple factors. For example, with the proliferation of Low Earth Orbit (LEO) mega-constellations and associated rise in number of rocket launches, the amount of space debris has also increased dramatically. As a result, satellites in the LEO region regularly need to perform collision avoidance maneuvers, which are not without inherent risk¹.

Another common reason is insertion into a final orbit. For example, telecommunications satellites often operate from a Geostationary Earth Orbit (GEO), situated roughly 36000 km away from the Earth's surface. However, due to their mass, they are often launched into an initial LEO stage, before maneuvering through a Geostationary Transfer Orbit (GTO) to their final destination. A recent example is the journey undertaken by the James Webb Space Telescope to the Sun-Earth L2 point.

Moving a spacecraft from one orbit to another involves the use of the satellite's propulsion subsystem, including thrusters and fuel tanks. The use of thrusters to perform an orbital maneuver is often called a "burn" or "firing the thrusters".

To perform most orbital maneuvers, the thrusters are activated for short periods of time, resulting in a change in the velocity vector of the satellite only. These are known as "impulsive" maneuvers. Some maneuvers, however, are executed over a longer period. These require consideration of both velocity vector change, position vector change, and changes in the mass of the satellite to be included in calculations. An example of such a maneuver is one using ion thruster engines.

In this Case Study, we will only consider a subset of impulsive maneuvers. We will go through the fundamental maths behind them, and we will then model them using the Ansys STK Astrogator tool.

As already mentioned, an impulsive maneuver results in a change in the velocity vector of a satellite. This change is conventionally denoted as ΔV , pronounced as Delta-Vee. Since this is a change in a vector quantity, we would need to specify both the magnitude and direction of this velocity change.

How much ΔV is necessary for a certain maneuver depends on the mass of the spacecraft, the orbital parameters of the initial and final orbit, where along its orbit the satellite is, i.e. its instantaneous velocity, and last but not least, the type of fuel used. Mathematically, ΔV is defined through the ideal rocket equation:

$$\Delta V = I_{sp} g_0 \ln \frac{m_w}{m_d}$$

Where:

- I_{sp} is the specific impulse, a property of the fuel used, measured in seconds.
- g_0 is the standard gravity, i.e. $\sim 9.81 \text{ m/s}^2$.
- m_w is the *wet mass* of the satellite, i.e. the sum of the spacecraft mass and the propellant mass.
- m_d is the *dry mass* of the satellite, i.e. the mass of the spacecraft after the thruster burn has finished.

¹ <https://www.space.com/satellites-collision-avoidance-maneuvers-increase-collision-risk>



2. The Hohmann Transfer

The Hohmann Transfer maneuver is commonly used by satellites transferring between two orbits in the same plane, around the same central body. It is the most fuel-efficient two-impulse maneuver, and as such is preferred when mission designers want to minimize overall fuel mass.

The downside of the Hohmann Transfer maneuver is that it is not the fastest possible way to maneuver a satellite between to coplanar orbits.

The concept of a Hohmann Transfer is illustrated in Figure 1. In this diagram, a satellite is in an initial circular orbit (Orbit 1) around a central body. Once it reaches point A, which is normally the periapsis, the satellite performs the first thruster burn. The direction of the ΔV is such that it adds to the satellite's instantaneous velocity.

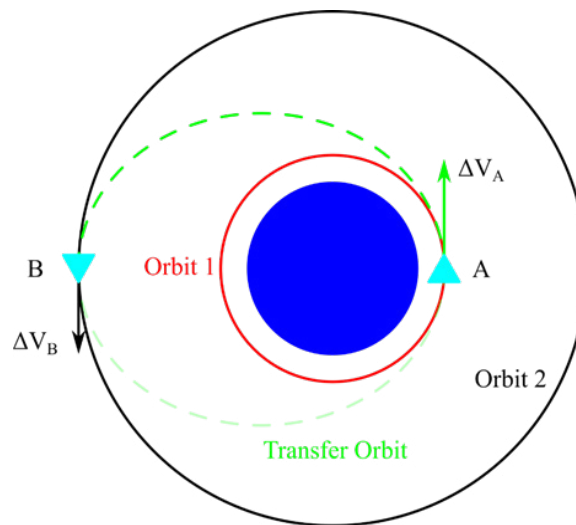


Figure 1. Hohmann Transfer

This ΔV is enough raise the apoapsis radius of the satellite's orbit and in effect places it in an elliptical transfer orbit, which has the same periapsis radius as Orbit 1, and an apoapsis radius equal to the apoapsis of the final Orbit 2.

Once the satellite reaches the apoapsis of the Transfer Orbit, i.e. point B, it then performs a second ΔV maneuver. Since this second maneuver is often used to place the spacecraft in a circular orbit, it is often referred to as "circularizing ΔV ". In essence, this maneuver raises the periapsis radius of the orbit.

The Hohmann Transfer can be used to maneuver a satellite from a higher to a lower orbit, for example when it is time to deorbit a particular mission. In this case, the direction of the ΔV , and by extension the thruster burn, is opposite to that of the instantaneous velocity vector.

Consider the following example to better illustrate the steps and calculations behind a Hohmann Transfer:

- A 200 kg satellite is in a circular LEO orbit with a height of 400 km.
- The target orbit is a circular MEO one with a height of 1200 km. This is a typical orbit for a OneWeb satellite².
- The satellite is using solid propellant in its propulsion system, with a *specific impulse* of 300 s.

² Additional information: Eutelsat OneWeb



The first step is to calculate the velocity of the satellite when it is at periapsis. Note that since the satellite is in a circular orbit, its velocity is the same regardless of its true anomaly. We can calculate it using the following formula:

$$V = \sqrt{\frac{\mu}{r}}$$

Where μ is the Earth's standard gravitational parameter, $3.986 \times 10^{14} \text{ m}^3/\text{s}^2$, and r is the orbit's radius. In this case, $r = 6378 + 400 = 6778 \text{ km}$. Applying this formula, we obtain the value of 7668.631 m/s for the satellite's velocity.

The next step is to calculate the velocity of the satellite at periapsis if it were in the elliptical transfer orbit. This time, we need to use general case formula for instantaneous velocity:

$$V = \sqrt{\mu \left(\frac{2}{r} - \frac{1}{a} \right)}$$

Where r is the distance between the focus of the orbit and the satellite, and a is the semi-major axis. For the elliptical transfer orbit, r will have the same value of 6778 km , since the satellite is still at periapsis. The semi-major axis will be half the sum of periapsis and apoapsis radii, or in this case $(6378 + 400 + 6378 + 1200) / 2 = 7178 \text{ km}$. The velocity of a spacecraft at periapsis in such an orbit will then be 7879.405 m/s .

The difference between these two values, i.e. $7879.405 - 7668.631 = 210.774 \text{ m/s}$, is the value of our first ΔV .

Using the same approach at point B, we find the value of the second ΔV to be 204.974 m/s .

An important note to make is that these calculations assume that all bodies are represented as ideal point masses, the ΔV thruster burn happens instantly, and the system is an ideal two-body one. In practice, this is not the case, and is a major reason for using the Ansys STK software and Astrogator to model such maneuvers.

Once the ΔV magnitudes are known, we can estimate how much fuel we would need for them. Applying the ideal rocket equation at point B, when the final dry mass of the satellite will be 200 kg , we find that we would need approximately 14.44 kg fuel. Working backwards to point A, where our final "dry" mass is now 214.44 kg . This is because the satellite needs to carry the fuel necessary for the second maneuver with itself. Given this, we find that we would need approximately 15.94 kg fuel for the first maneuver. Table 1 summarizes these findings.

Table 1. Hohmann Transfer Example Results

ΔV Maneuver	Required velocity change, [m/s]	Required fuel mass, [kg]
First, i.e. Point A	210.774	15.94
Second, i.e. Point B	209.974	14.44



3. Plane Change Maneuver

Another regularly executed maneuver is the plane change one. Instead of changing the shape of the orbit by changing the apoapsis and periapsis radii, this maneuver alters the inclination of the satellite's orbit. This type of maneuver is often combined with the second ΔV of a Hohmann Transfer.

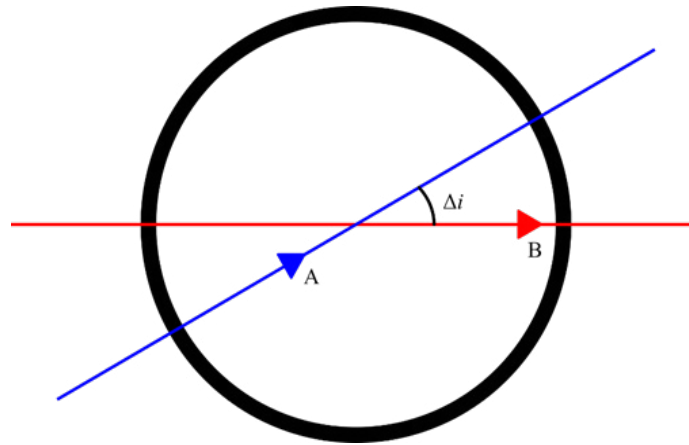


Figure 2. Inclination change maneuver

In the illustration in Figure 2, the inclination of the satellite's orbit is changed by an amount of Δi . However, other parameters of the orbit, such as the semi-major axis and eccentricity, will be preserved during an inclination-change only maneuver.

Since changing the inclination is essentially rotating the orbital plane around a line, it is often performed at either the ascending or descending node, rotating the plane around the line of nodes. This makes it possible to achieve a 0-degree inclination and is the usual practice for GEO satellites.

In contrast to the Hohmann Transfer, the direction of the ΔV vector is perpendicular to the instantaneous velocity vector and to the orbital plane.

For an inclination-only maneuver, the amount of ΔV can be calculated according to the following formula:

Due to its proportional dependence on the instantaneous velocity, inclination change maneuvers are very expensive in terms of required propellant mass. For example, for an inclination change of 60 degrees, the required ΔV will be equal to the spacecraft's current velocity. Because of this, these maneuvers are minimized by, for example, achieving the desired inclination during the launch phase.

4. Modeling with Astrogator

We will use a short scenario to showcase satellite maneuver modeling and optimization using the Ansys STK Astrogator tool. Since we will only be modeling the maneuver and a few orbits before and after it, the overall duration of the scenario can be set to 24 hours.

We will model two maneuvers – a Hohmann Transfer and an inclination change. We will use the same spacecraft and initial orbit parameters as those in the introductory sections.



4.1 Hohmann transfer – calculations by hand

We begin by creating a new scenario with Earth as a central body and adding a single satellite. In contrast to previous scenarios where we would use the TwoBody, J2, or HPOP orbital propagators, we are going to use the Astrogator one for this and subsequent scenarios. An initial set up of the Astrogator propagator is illustrated in Figure 3. The way the starting orbit of the satellite is specified is the same as with the other propagators. We can also readily select the Earth-center ICRF coordinate system.

The way Astrogator works is by composing a list of “Segments” into a “Mission Control Sequence”, or MCS for short. There is a wide range of options for these segments, however the ones we will be using in our scenarios are the “Propagate”, “Maneuver”, and “Target Sequence”.

New segments are added through the “New” icon underneath the green “Play” icon. We can also generate an report with a summary of the MCS by pressing the scroll-looking icon. In this particular example, we have also changed the colors of the individual segments to tell them apart more easily in the Graphics windows.

For our first scenario, we will use the values for ΔV which we calculated manually earlier. Therefore, our MCS consists of three Propagate segments and two Maneuver ones.

Once we have inserted all the segments, we can specify the spacecraft parameters and fuel tank parameters – these will be important later when we ask Astrogator to calculate the fuel mass necessary for the individual ΔV maneuvers.

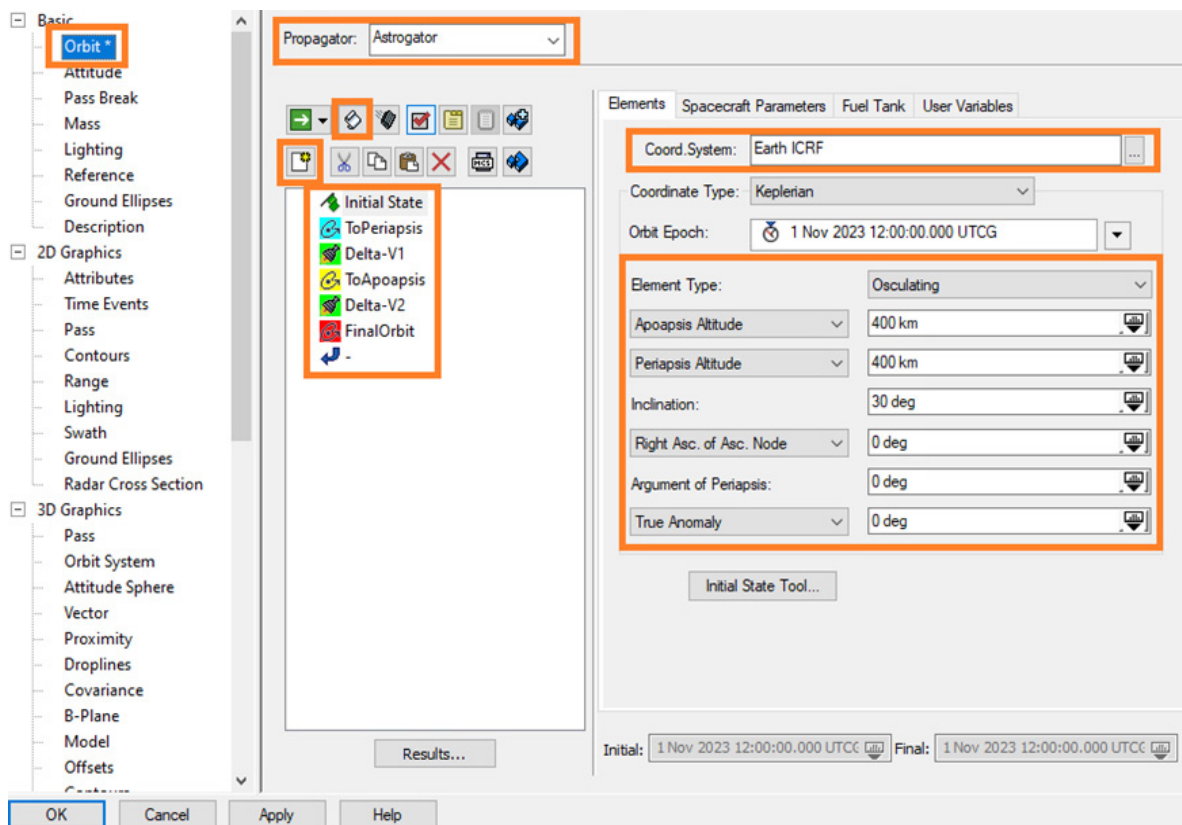


Figure 3. Initial Astrogator setup



The figure consists of two side-by-side screenshots of the STK (Systems Tool Kit) software interface, specifically the 'Spacecraft Parameters' and 'Fuel Tank' tabs.

Left Screenshot (Spacecraft Parameters):

- Drag:** Dry Mass: 200 kg (highlighted with an orange box), Coefficient (Cd): 2.2, Area: 20 m².
- Solar Radiation Pressure (Spherical):** Coefficient (Cr): 1, Area: 20 m².
- Radiation Pressure (Albedo/Thermal):** Coefficient (Ck): 1, Area: 20 m².
- GPS Solar Radiation Pressure:** K1: 1, K2: 1.
- Initial:** 1 Nov 2023 12:00:00.000 UTCC, **Final:** 1 Nov 2023 12:00:00.000 UTCC.

Right Screenshot (Fuel Tank):

- Tank Pressure:** 5000 Pa.
- Tank Volume:** 1.5 m³.
- Tank Temperature:** 293.15 K.
- Fuel Density:** 1000 kg/m³.
- Fuel Mass:** 35 kg (highlighted with an orange box).
- Maximum Fuel Mass:** 100 kg (highlighted with an orange box).
- Initial:** 1 Nov 2023 12:00:00.000 UTCC, **Final:** 1 Nov 2023 12:00:00.000 UTCC.

Figure 4. Spacecraft and fuel tank parameters

The values that we will use are shown in Figure 4. Note that we can specify other values which affect a spacecraft's orbit, such as drag coefficients and area, etc. For this particular case, we will use the default values for these parameters.

When configuring the Propagate segments we need to specify the correct stopping conditions. In accordance with the theory behind Hohmann Transfers, our satellite needs to apply its first ΔV burn at the periapsis, or perigee, of its orbit. The second burn is applied when the satellite has reached the apoapsis, or apogee, of the transfer orbit. Lastly, we have a Propagate segment for half a day, just so we can visualize the final orbit. Figure 5 illustrates how to set up the first such segment. Putting "3" in the "Repeat Count" field means that the orbit of the satellite will be propagated until it has visited the periapsis point three times, at which point Astrogator will move on to the next segment in the MCS.

We configure the "ToApoapsis" segment in a similar way, while we can configure the "FinalOrbit" to propagate for as long as we want. This is how we will configure all such segments, not just for this satellite, but for all subsequent ones in this Case Study.

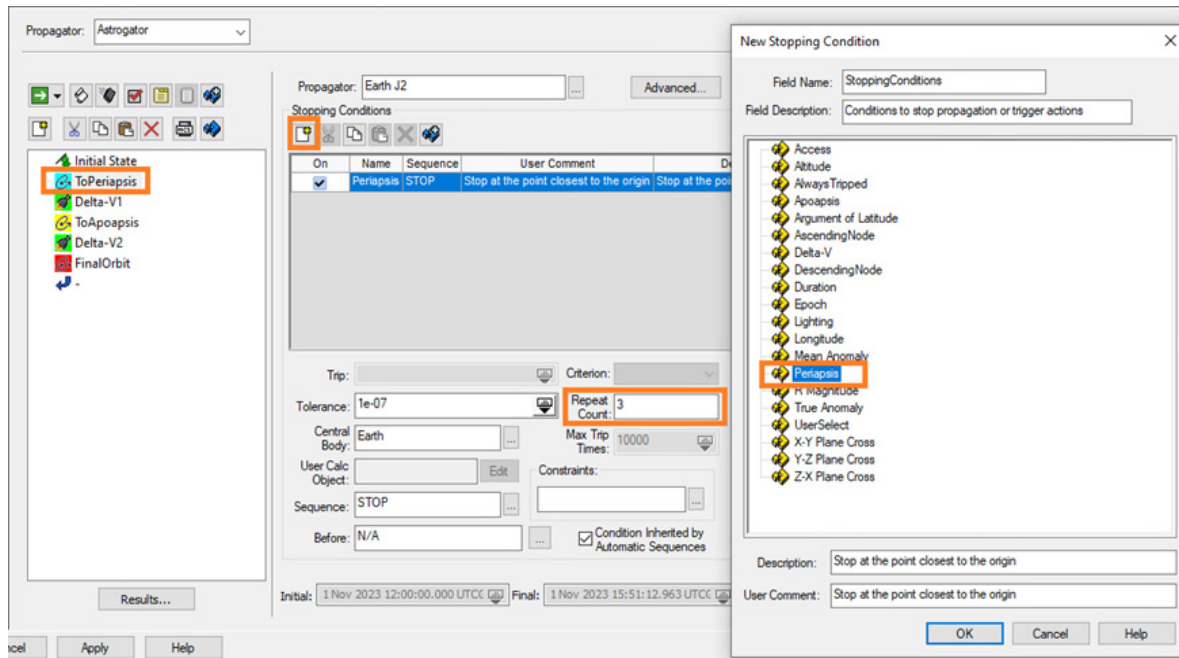


Figure 5. Setting up a Propagate segment

When configuring the Maneuver segments we need to specify the following parameters – force model, ΔV magnitude, and engine model. Because this is an orbit raising maneuver, we configure the direction of the ΔV change to be along the satellite's velocity vector. We also use the default constant thrust engine model, which uses fuel with an Isp of 300 s, the same one we used in our calculations.

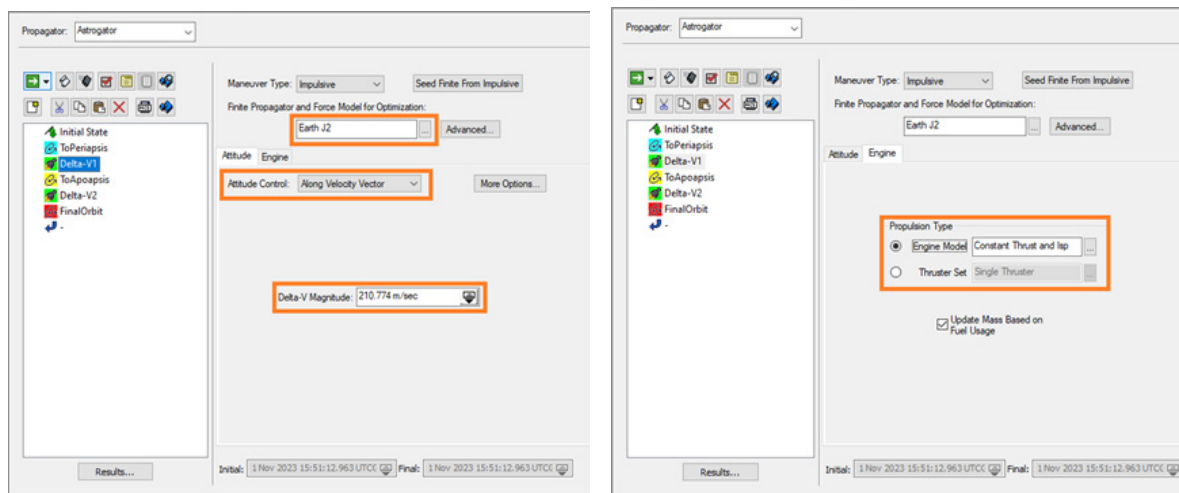


Figure 6. Maneuver segment settings

These settings are illustrated in Figure 5 for the first ΔV maneuver. The second one is configured in a similar fashion, using the same numerical values as those in Table 1. Once we have configured all segments, we can then press the green arrow button to execute all of them. The 2D and 3D Graphics windows are automatically with the results of the entire MCS.

For example, Figure 6 shows the initial orbit of the satellite, before the ΔV burns. Figure 7 then shows the satellite during the transfer orbit (left picture) and when it has reached the final orbit (right picture).

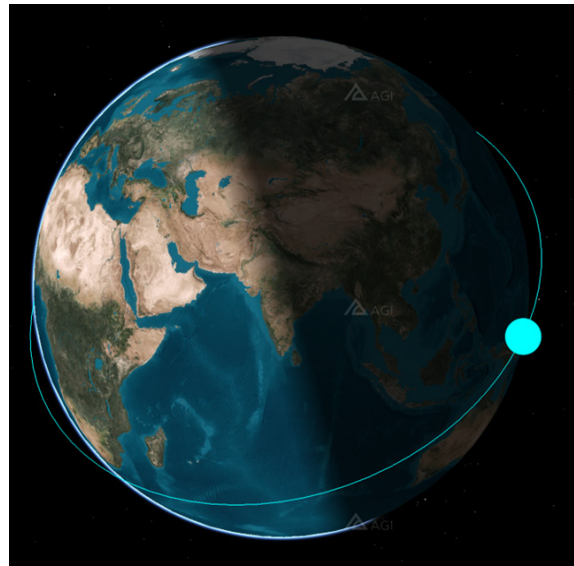


Figure 7. Satellite at start of the scenario

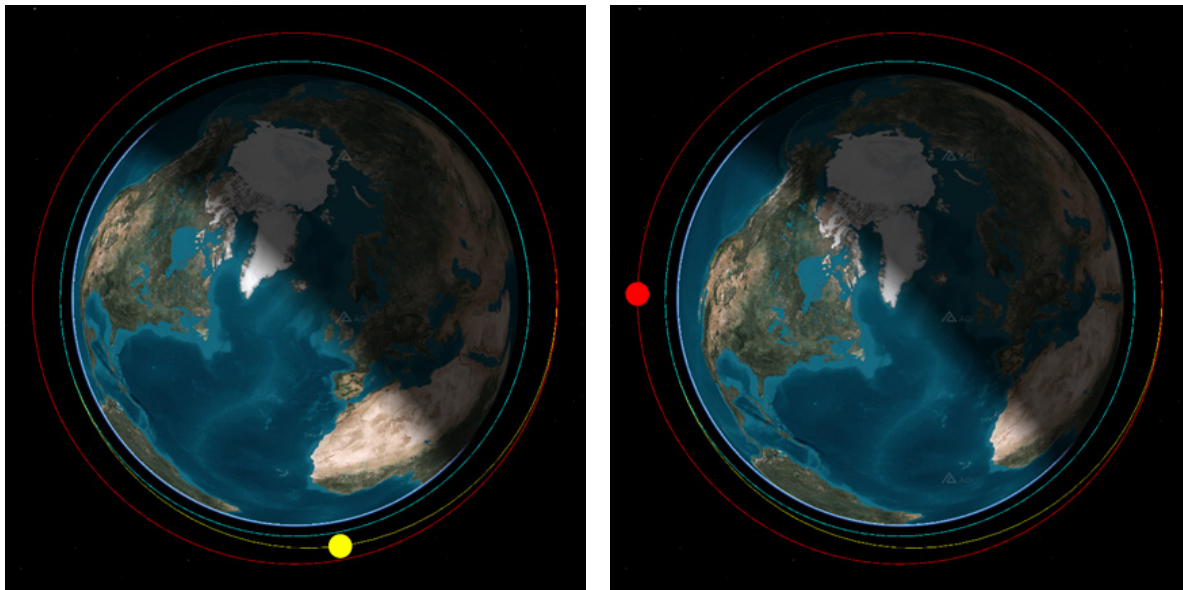


Figure 8. Transfer and final orbit

As mentioned before, we can generate reports for every segment of the MCS. These are very informative and can be used in the verification and validation stages of a mission design.

For example, we can use the report for the first ΔV maneuver to cross-check our fuel calculations, but also to confirm that the maneuver has achieved the desired outcome.

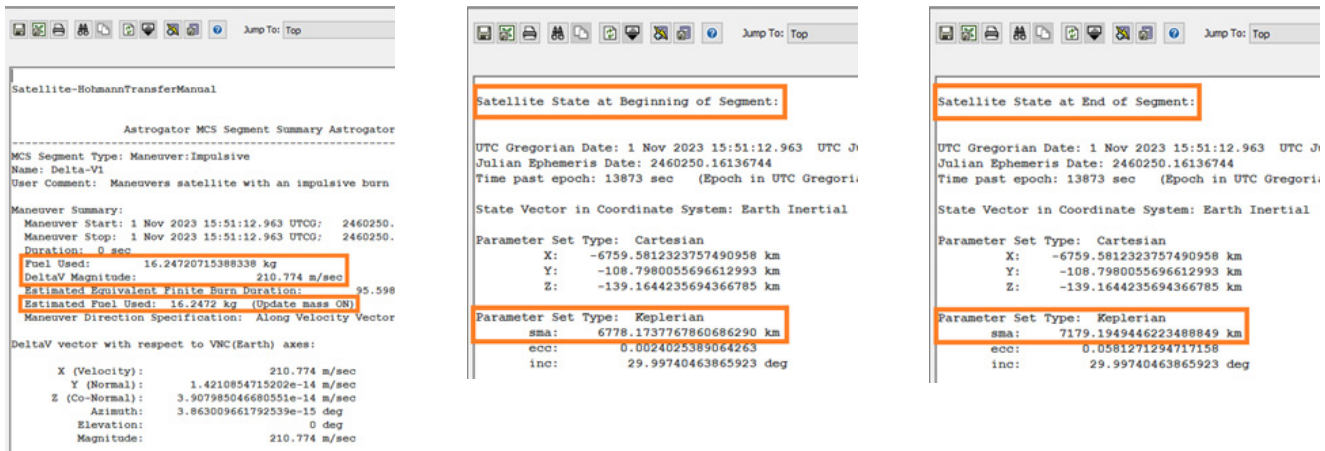


Figure 9. Astrogator segment reports

Extracts of these reports are presented in Figure 9. From these, we can see that Astrogator has calculated we would need approximately 16.25 kg of fuel, which is quite close to our manually calculated value of 15.94 kg. We can also see that the semi-major axis of the satellite's orbit has changed from 6778 km to 7179 km, signifying that it has been successfully placed in the transfer orbit.

Next, we will demonstrate how to use Astrogator's capabilities to calculate the amount of ΔV necessary for a particular maneuver.

4.2 Hohmann transfer – optimized using Targeter

Some satellite maneuvers can be more complex than a simple Hohmann transfer. Even in the case of Hohmann transfers, the formulas used so far do not consider the Earth's oblateness or third-body gravitational influences.

We can achieve more realistic simulation results, and hence plan more accurately, by using the software's numerical optimization capabilities. In the case of the Ansys STK Astrogator tool, these are provided by the Targeter tool, and through the "Target Sequence" segment. An illustration of the initial steps in setting up a Target Sequence is shown in Figure 10.

Same as with the previous models, we use the "Earth J2" orbital propagator, mainly to ensure quick calculations. We also nest the first thruster burn ("Delta-V1"), as well as the orbital propagation segment corresponding to the transfer orbit ("ToApoapsis"). Finally, we place a checkmark next to the input box for "Delta-V Magnitude", as this is our independent variable, or control parameter, the one we want to use Targeter to find out.

We could have left the latter outside of the "Target Sequence" segment since we are only interested in the "Delta-V Magnitude". However, that would have led to inaccurate results.

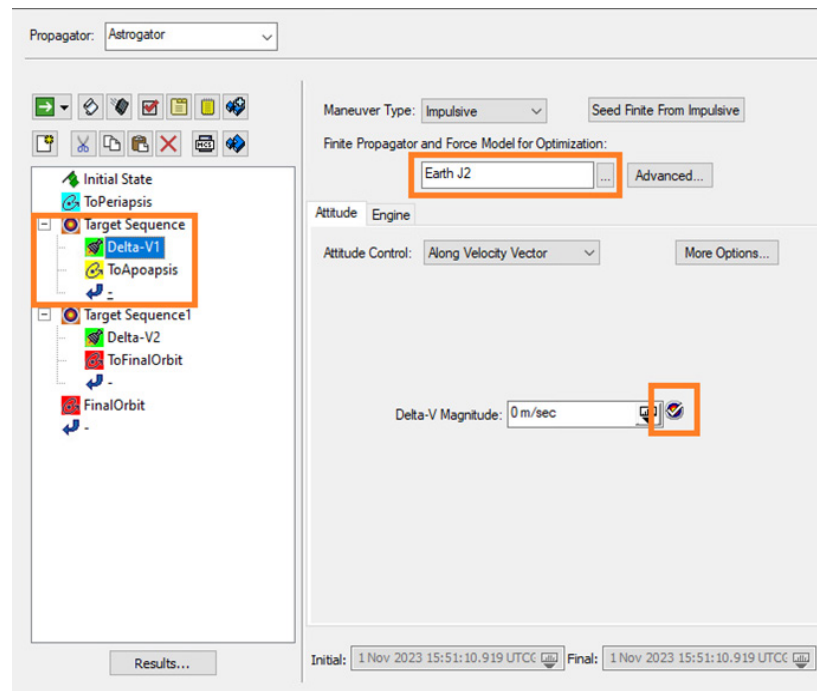


Figure 10. Astrogator Targeter Setup

After all, we want Targeter to find a value for the first ΔV burn such that the satellite will be in the correct transfer orbit *by the time it finishes the “ToApoapsis” segment*. By placing both segments within the same “Target Sequence”, we make sure that any perturbations experienced by the satellite during its flight time on the “ToApoapsis” segment are taken into account when optimizing the “Delta-V1” segment.

The next step is to define our constraint, which in our case will be the apoapsis altitude for the transfer orbit. Figure 11 shows the steps that we need to take to do this – selecting the appropriate segment (“ToApoapsis”), clicking on “Results”, and then finding the “Altitude of Apoapsis” value.

As you can see, there are numerous different parameters which can be used as constraints. This is quite useful and helpful when designing complex trajectories, working with constellations of satellites, or performing trade-off studies.

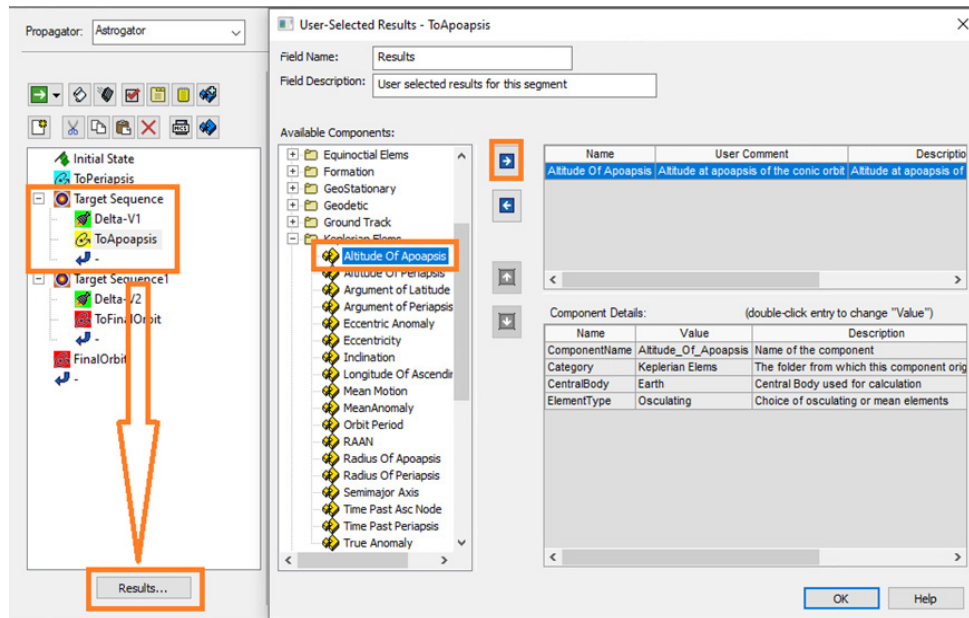


Figure 11. Selecting which parameters to be used by Targeter

Finally, we need to tell Targeter which independent variables and which constraints to use, and what our desired values are for the latter. In this case, we only have one of each, however in more complex situations there can be more. The steps are illustrated in Figure 12. The way they are set up is similar to setting up any optimization routine, including specifying the tolerance for the constraint.

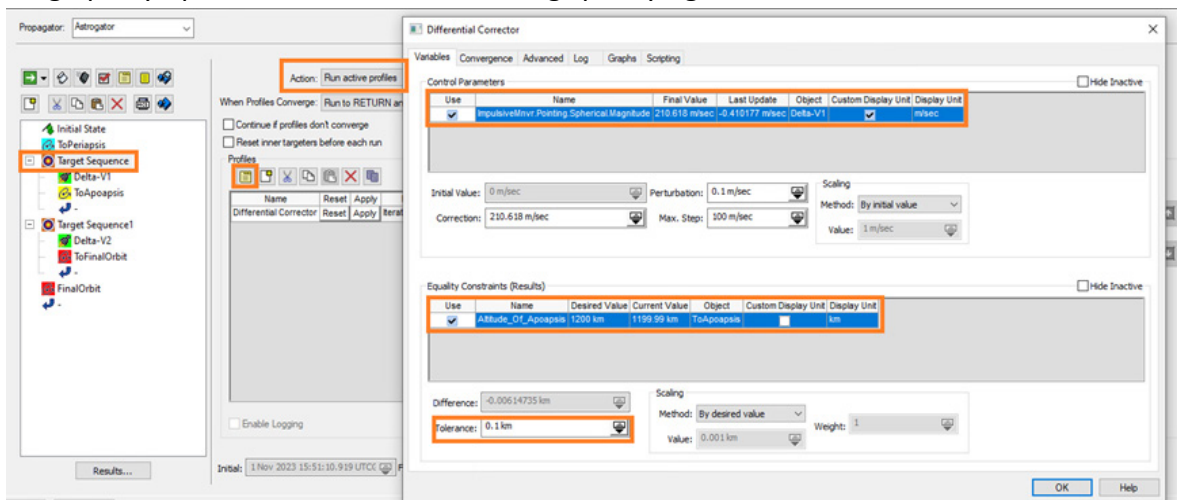


Figure 12. Setting up Targeter's Differential Corrector

We need to remember to also change the "Action" field to "Run active profiles" as this is what enables Targeter. Otherwise, the MCS will run with the default values provided by each segment.

Here the description of the second Target Sequence (named "Target Sequence 1") is missing. Up to this point, the reader only had instructions to build the first TS. The second TS needs to correct for eccentricity = 0; this is a Result which should be applied to the Delta-V2 segment.

Once we have configured both target sequences, we can run the MCS as we would normally do. Provided we have selected "Run active profiles" on all of them, we would get a new window per Target Sequence open with information similar to the ones in Figure 13.



Control	New Value	Last Update	Constraint	Desired	Achieved	Difference	Tolerance
Delta-V1 : ImpulsiveMnvr.Pointing.Spherical.Magnitude	210.618 m/sec	-0.41018 m/sec	ToApoapsis : Altitude Of Apoapsis	1200 km	1199.99 km	-0.0061474 km	0.1 km
Control	New Value	Last Update	Constraint	Desired	Achieved	Difference	Tolerance
Delta-V2 : ImpulsiveMnvr.Pointing.Spherical.Magnitude	209.728 m/sec	0 m/sec	ToFinalOrbit : Altitude Of Periapsis	1200 km	1199.96 km	-0.03596 km	0.1 km

Figure 13. Targeter results for the two Target Sequences

These results refer to the second ΔV burn, the one used to raise the periapsis and circularize the final orbit. Comparing the result obtained by numeric optimization to the one we calculated by hand in Table 1 we can see that the two are quite close. The desired value of 1200 km has also been achieved within the specified tolerance of 0.1 km.

We can also double-check the summary reports for the two Targeter segments to confirm that the Keplerian orbital elements are as expected. These are illustrated in Figure 14 and show that the initial and transfer semi-major axes have virtually identical values to those from Figure 9. We can also confirm that the eccentricity of the final orbit is low (0.0011), meaning that the second ΔV burn has succeeded in circularizing the final orbit.

<p>Satellite State at Beginning of Segment:</p> <p>UTC Gregorian Date: 1 Nov 2023 15:51:10.919 UTC Ju Julian Ephemeris Date: 2460250.16134378 Time past epoch: 13870.9 sec (Epoch in UTC Gregor)</p> <p>State Vector in Coordinate System: Earth Inertial</p> <p>Parameter Set Type: Cartesian</p> <p>X: 95.7069530260984891 km Y: -6759.9082322849453703 km Z: -131.4158386126338485 km</p> <p>Parameter Set Type: Keplerian</p> <p>sma: 6778.1766101034581880 km ecc: 0.0024068060380361 inc: 29.9999387878801 deg</p>	<p>Satellite State at End of Segment:</p> <p>UTC Gregorian Date: 1 Nov 2023 15:51:10.919 UTC Ju Julian Ephemeris Date: 2460250.16134378 Time past epoch: 13870.9 sec (Epoch in UTC Gregor)</p> <p>State Vector in Coordinate System: Earth Inertial</p> <p>Parameter Set Type: Cartesian</p> <p>X: 95.7069530260984891 km Y: -6759.9082322849453703 km Z: -131.4158386126338485 km</p> <p>Parameter Set Type: Keplerian</p> <p>sma: 7178.8817962774846819 km ecc: 0.0580896794793019 inc: 29.99993878788009 deg</p>	<p>Satellite State at End of Segment:</p> <p>UTC Gregorian Date: 1 Nov 2023 16:41:32.666 UTC Ju Julian Ephemeris Date: 2460250.19631771 Time past epoch: 16892.7 sec (Epoch in UTC Gregor)</p> <p>State Vector in Coordinate System: Earth Inertial</p> <p>Parameter Set Type: Cartesian</p> <p>X: -115.3515836248910063 km Y: 7575.4063065198924960 km Z: 167.2737969280883306 km</p> <p>Parameter Set Type: Keplerian</p> <p>sma: 7586.5128659941747173 km ecc: 0.0011048571993140 inc: 29.99749734183736 deg</p>
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Figure 14. Astrogator segment reports - Targeter

Checking the 3D Graphics Window we can see a visualization similar to that of Figure 15, which shows the intermediate orbits that Targeter had tried before finding the optimal ones.

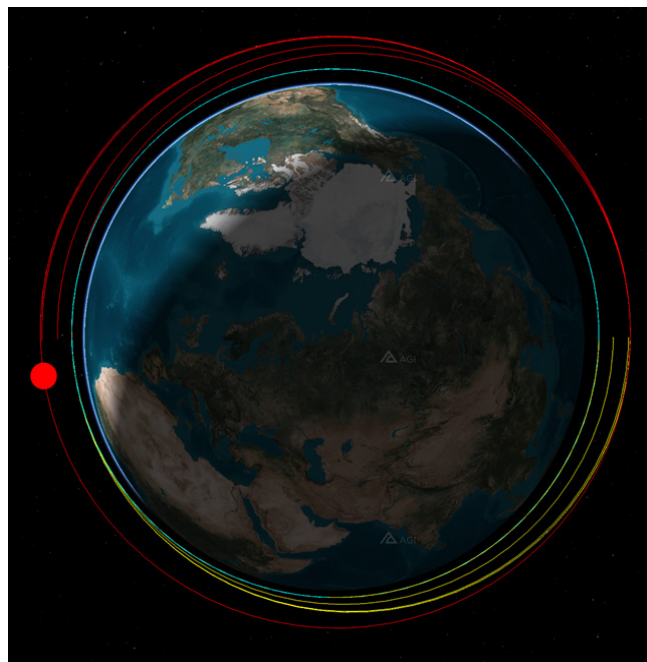


Figure 15. Additional Targeter results



4.3 Plane change maneuver

As previously mentioned, in contrast to a Hohmann Transfer, a plane change maneuver requires thruster burns in a more general direction rather than along the instantaneous velocity vector.

We can model such maneuvers in Targeter by changing some of the parameters of our “Maneuver” segments. We begin by inserting a new satellite or starting a new scenario. We place the satellite in a geosynchronous initial orbit with an inclination of 30 degrees, with the goal of maneuvering it to a geostationary orbit.

We then configure the Astrogator MCS and the “Target Sequence” segment as per Figure 16 and Figure 17.

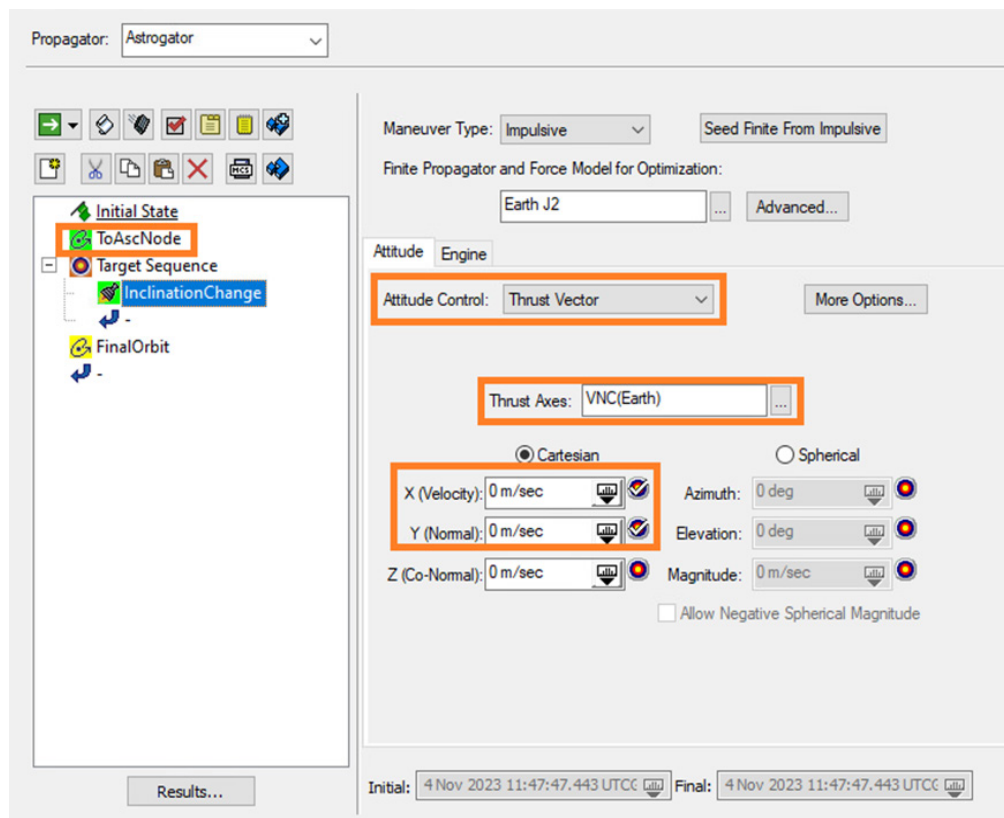


Figure 16. Plane change maneuver MCS setup

The things to note are that we propagate our satellite to its Ascending Node, rather than to apoapsis or periapsis. The reason is that we need to “rotate” the satellite’s orbit around the line of nodes in order to achieve an orbit with zero inclination.

We have also changed the “Attitude Control” field to “Thrust Vector” which gives us a more general control of the direction of the ΔV burn. In this case, we have selected parameters corresponding to two directions as independent values – X, along the instantaneous velocity vector, and Y, normal to the orbital plane.

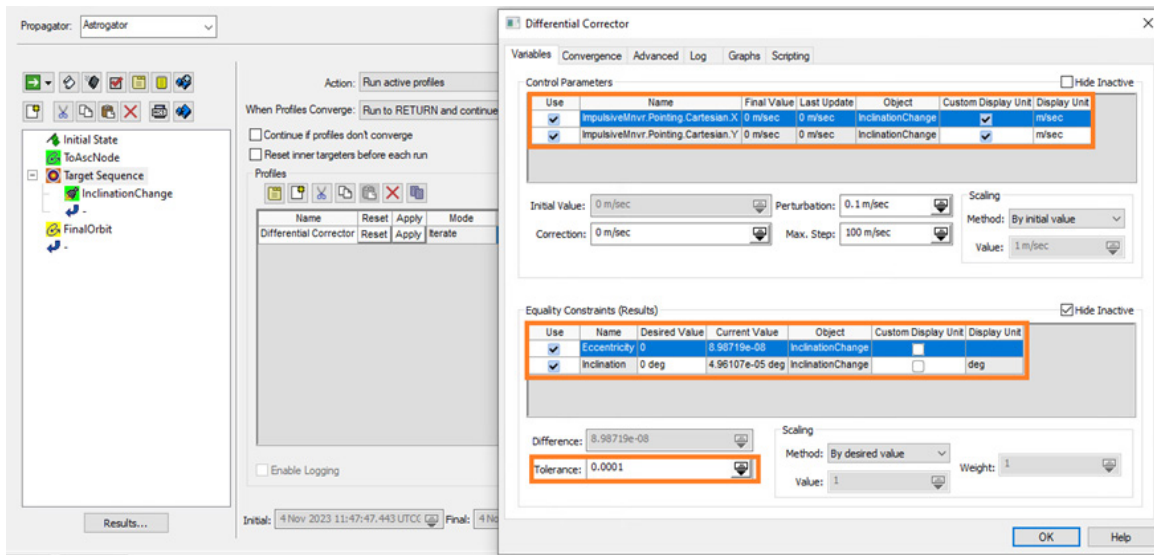


Figure 17. Plane change Targeter setup

When configuring Targeter, we enable these two parameters, and we also enable two constraints – inclination and eccentricity. We include eccentricity as well to make sure the shape of the orbit is preserved after the ΔV burn.

The results from running this MCS are shown in Figure 18, demonstrating that our desired values for eccentricity and inclination have been achieved.

Control	New Value	Last Update	Constraint	Desired	Achieved	Difference	Tolerance
InclinationChange : ImpulsiveMnvr.Pointing.Cartesian.X	-288.071 m/sec	-0.017319 m/sec	InclinationChange : Eccentricity	0	3.57729e-08	3.5773e-08	0.0001
InclinationChange : ImpulsiveMnvr.Pointing.Cartesian.Y	-1299.41 m/sec	-0.03719 m/sec	InclinationChange : Inclination	0 deg	2.93758e-05 deg	2.9376e-05 deg	0.0001 deg

Figure 18. Targeter results

Figure 19 shows an excerpt from the Astrogator report file for this satellite. You can clearly see how much more ΔV is required to perform an inclination change maneuver, compared to the ΔV required for the Hohmann Transfer.

```

Astrogator MCS Segment Summary Astrogator MCS InclinationChange Summary
-----
MCS Segment Type: Maneuver:Impulsive
Name: Target Sequence.InclinationChange
User Comment: Maneuvers satellite with an impulsive burn or finite burn

Maneuver Summary:
Maneuver Start: 4 Nov 2023 11:47:47.443 UTCG; 2460252.99152134 UTC Julian Date
Maneuver Stop: 4 Nov 2023 11:47:47.443 UTCG; 2460252.99152134 UTC Julian Date
Duration: 0 sec
Fuel Used: 0 kg
DeltaV Magnitude: 1330.954164785772 m/sec
Estimated Equivalent Finite Burn Duration: 535.2950715997686 sec
Estimated Fuel Used: 90.9748 kg (Update mass OFF)
*** Estimated fuel usage higher than initial fuel mass ***
Maneuver Direction Specification: Thrust Vector

```

Figure 19. Targeter Segment Report

Finally, Figure 20 shows the satellite's orbit before and after the inclination change maneuver.

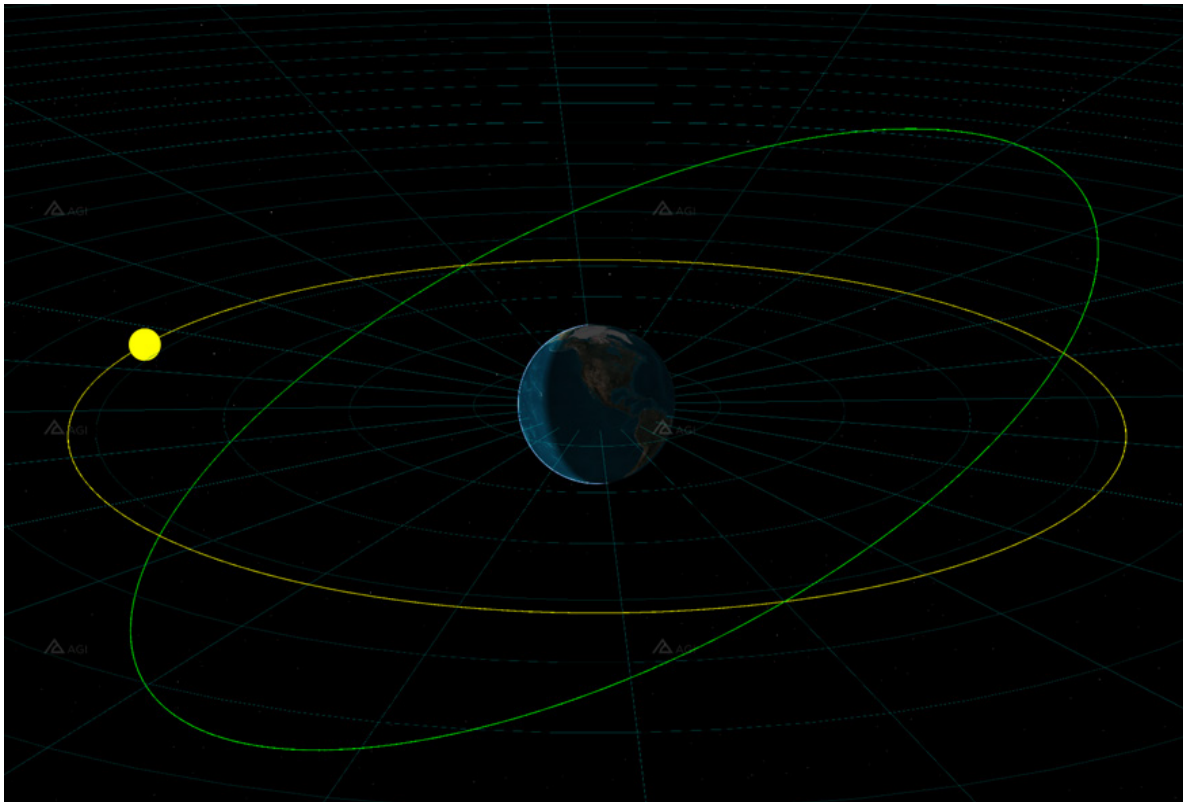


Figure 20. Before and after an inclination change maneuver

5. How does Ansys STK software improve understanding?

The first thing the software helps with is visualizing the orbits of satellites before, during, and after any maneuvers. The ability to observe and inspect 3D visualizations of what are inherently 3D problems can be incredibly helpful with abstract concepts. Using Astrogator also helps put theoretical studies of astrodynamics and calculating ΔV for different scenarios into practice. Another aspect is increased fidelity of calculations when using Targeter, and links to other concepts such as third-body gravitational influences. Finally, incorporating the vast number of reports that Ansys STK software can generate, such as Sun angles, access times and durations, attitude and orientation information, and others, with the design of orbital maneuvers can help significantly in systems engineering studies and overall mission design and development.

6. References

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