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Demonstrating High-Accuracy Orbital Access Using Open-Source Tools

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Demonstrating High-Accuracy Orbital Access Using Open-Source Tools

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Abstract

Orbit propagation is fundamental to almost every space-based analysis. Currently, many system analysts use commercial software to predict the future positions of orbiting satellites. This is one of many capabilities that can be replicated, with great accuracy, without using expensive, proprietary software. NASA's SCA_N (Space Communication and Navigation) Center for Engineering, Networks, Integration, and Communications (SCENIC) project plans to provide its analysis capabilities using a combination of internal and open-source software, allowing for a much greater measure of customization and flexibility, while reducing recurring software license costs. MATLAB[®] (The MathWorks, Inc.) and the open-source Orbit Determination Toolbox created by the NASA Goddard Space Flight Center (GSFC) were utilized to develop tools with the capability to propagate orbits, perform line-of-sight (LOS) availability analyses, and visualize the results. The developed programs are modular and can be applied for mission planning and viability analysis in a variety of Solar System applications. The tools can perform 2 and N-body orbit propagation, find inter-satellite and satellite to ground station LOS access (accounting for intermediate oblate spheroid body blocking, geometric restrictions of the antenna field-of-view (FOV), and relativistic corrections), and create animations of planetary movement, satellite orbits, and LOS accesses. The code is the basis for SCENIC's broad analysis capabilities including dynamic link analysis, dilution-of-precision navigation analysis, and orbital availability calculations.

1.0 Introduction

The desire to predict orbital motion has existed since ancient times. The prediction of orbits is an integral part of any space mission. Satellite orbit determination (OD) is the method for determining the position and velocity (i.e., the state) of an orbiting object at any time based on position and velocity measurements made on the object (Ref. 1). Unfortunately, only the simplest orbit determination problems have exact analytic solutions, so most problems of practical interest require a combination of analytical and numerical solution methods (Ref. 2). The systems of differential equations, which characterize the future motion of a space object (dynamical equations of motion), are found by using theoretically backed force models. The equations of motion depend on the spatial and physical characteristics of the space object. Near-Earth satellites are typically assumed to be influenced by a variety of external forces; including gravity, atmospheric drag, solar radiation pressure, third-body perturbations, Earth tidal effects, and general relativity (Ref. 1). The combination of these forces create a highly complicated set of nonlinear differential equations.

To deal with this complex problem, most space systems analysts use commercial software to predict spacecraft orbits reliably. Unfortunately, there are several downsides to using commercial tools. They can

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be expensive and hard to work with and are sometimes inflexible to the rapidly changing needs of the user. They may also do more than one needs them to, which would mean one is paying for features they do not want. To bypass these problems, SCENIC has plans to utilize a combination of MATLAB® and NASA-developed open-source toolboxes to replicate the analysis capabilities it needs, instead of using commercial, proprietary software. This will enable the project to lower costs, increase the flexibility of possible analyses, and enable high customization of future analysis capabilities.

Besides, orbit propagation, SCENIC is also interested in the ability of satellites to communicate with each other and with ground stations. It is important to be able to know when satellites can see each other and ground stations for both sending commands, and offloading data. Several constraints affect whether or not satellites can communicate. The quality of a received signal can be affected by multiple losses; including path loss, atmospheric attenuation, and other sources of implementation loss. Moreover, a satellite link can become impossible if another space object physically blocks the signal, or the satellite antenna is pointing in the wrong direction. In addition to orbit propagation, these strong, geometric constraints were the primary focus of the intern project. The long-term goal for this work is to integrate the developed capabilities into a web-friendly user interface. In Section 2.0, the methodology and program logic is described. In Section 3.0, the program outputs and analysis results are demonstrated. Section 4.0 outlines the conclusions and possible future work.

2.0 Methodology

The process for creating these finalized capabilities was influenced heavily by the desire to create code that can be easily used. While the actual output of the code was important, having tools whose process could be easily understood was an equally high priority, as the basic orbit propagation and LOS functionalities feed into many future capabilities. The process of enabling a functionality began with writing the code that actually computes the desired quantities. This was done using a combination of MATLAB® and NASA developed open-source toolboxes. These included GSFC's Orbit Determination Toolbox (Ref. 3) and Jet Propulsion Laboratory's SPICE Toolkit (Ref. 4). These toolboxes have the capability to aid in the propagation of orbits as well as transferring between the many relevant reference frames.

Once a basic function had been developed, the logic flow of the program needed to be documented, in detail, and its performance needed to be characterized. Workflows and known test cases were necessary to fulfill these requirements and were created for the main capabilities functions. Additionally, high in-code readability was maintained with extensive headers and commenting. Each function's header contains standard data including a brief and detailed function description, input specifications and examples, output specifications and examples, additional function dependencies, relevant example files, author and revision information, along with any function specific information (e.g., where one might find the most up to date required data files or notes on the orientations of satellite reference frames). Comprehensive example files have been created for the main orbit propagation and LOS access functions to illustrate their functionality and produce a variety of data products, which intuitively display the program outputs. The programs were also designed to use a consistent logical flow and units for calculations. In this way, the programs were designed to reduce possible sources of confusion for users wanting to investigate the functionality of the programs.

With these documented programs in place, the next pursued step was validating the outputs of the programs by comparing the results to those of trusted simulations and measurements of real-world satellites. After these steps were taken, the final step was to provide these verified and validated

capabilities to the SCENIC project for future integration into the web user-interface (UI). In this way, the developed capabilities will eventually be used directly by customers desiring mission analysis products.

3.0 Results

Several capabilities were built up to support the orbit propagation and LOS analysis. For example, methods to transfer between reference frames were created. These included the functions to find the basis vectors for a space craft based on its position and velocity, for a ground station based on its latitude and longitude, and a general function which rotates a given basis based on three Euler angles. Functions were also created to transfer between the different planetary coordinate systems using SPICE (Ref. 4). The main celestial coordinate frames that were used included the J2000/International Celestial Reference Frame (ICRF), body-centered inertial frames, and body-fixed frames. The ICRF referenced in this paper is centered on the Solar System barycenter and has a reference plane is aligned with the Earth's mean equator at the J2000 epoch (noon on January 1, 2000). The ICRF is useful for interplanetary applications. The body-centered inertial frames (inertial frames from here on) are centered on each major body's center of mass, but do *not* rotate with their associated body. They have reference planes parallel to that of the ICRF. The inertial frames are useful for satellite to satellite analysis and orbit propagation. The body-fixed frames (fixed frames from here on) are also centered on each major body's center of mass, but *do* rotate with their associated body. They have reference planes perpendicular to the rotation axis of their bodies. The fixed frames are useful for satellite to ground station analysis and finding motion relative to a terrestrial observer. The ephemeris and orientation data for each major body reference frames is available from the NAIF website and are easily updatable by replacing the desired kernel files and adding their names to a metakernel text file.

3.1 Orbit Propagation

Several methods of orbit propagation were implemented. All the methods that were used assume that the space object whose orbit is being propagated is of insufficient mass to alter the paths of the major Solar System bodies. For this paper, the major Solar System bodies include the Sun, planets, and Pluto. The propagation inputs include the start and end dates, the desired time step to find the orbit positions at, the Keplerian orbit parameters of the space object, the epoch where the parameters are found, and the central body that the orbital parameters are relating to (can be any major body). The orbit parameters include the semi-major axis (*SMA*), inclination (*i*), eccentricity (*e*), right ascension of the ascending node (*RAAN*), argument of periapsis (*AP*), and mean anomaly (*MA*). The generated tools can perform 2 and N-body orbit propagation and, for the Earth, can account for nonspherical gravity, solar pressure, and atmospheric drag. All propagators return the modified Julian dates and the space object states in the inertial frame at each requested time.

The first implemented method was a simple Keplerian 2-body orbit propagation. This type of orbit propagation is one of the few ways that utilize an exact solution to a simplified orbit problem. This type of propagation assumes that the only force acting on the spacecraft is that of a large point mass (or uniform density sphere). This type of orbit propagation is very stable and creates orbits that are slices of conic sections. This type of propagation is shown in Figure 1. The top two figures show the position and velocity of a geostationary Earth orbit (GEO) satellite in Earth's inertial frame propagated one day from the J2000 epoch. The bottom two figures show the position and velocity of the same GEO satellite in the body-fixed frame.

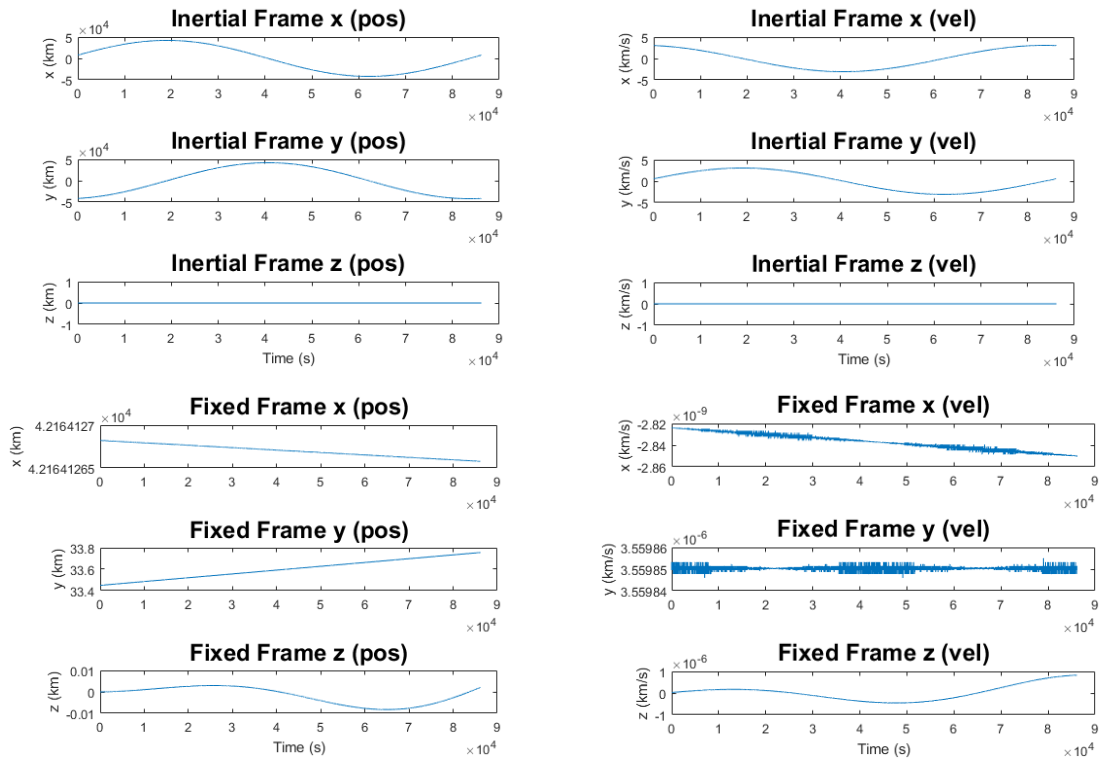


Figure 1.—Keplerian 2-Body Orbit Propagation for a GEO Satellite

As expected, the inertial frame plots show a single, full period of motion, with no motion in the z -axis. The fixed frame coordinates show the coordinates in the x - and y -axes staying essentially constant, with small deviations due to imprecision in the initial conditions. The velocities in the planar axes remain near 0 with small fluctuations due to numerical precision. The motion in the z -axis comes from Earth's polar motion.

The second implemented method was numerical 2-body orbit propagation. This type of orbit propagation was implemented as a first step towards n -body propagation and was used to find differences that exist between the Keplerian method and a numerical method applied to a problem with the same assumptions and constraints. The numerical methods worked by using MATLAB's built in numerical integration methods (defaulting to a variable-order method meant for nonstiff equations) to integrate force models once to get velocity, and twice to get position. The numerical 2-body propagation is shown in Figure 2. The figure is given in the same format, and for the same orbit as Figure 1.

The difference between the inertial frame plots of Figure 1 and Figure 2 is imperceptible, but in the fixed frame there are oscillatory features in the velocities that did not exist before. The motion in the fixed frame z -axis remains unchanged.

The third implemented method was numerical N -body orbit propagation. This type of orbit propagation is meant to allow for the prediction of interplanetary orbits. Each planet is treated as a point mass whose gravity originates from its instantaneous position at each time of propagation. The bodies past the Earth have many moons and are treated as single objects at each planetary barycenter unless otherwise specified. This type of propagation is shown in Figure 3. The figure is given in the same format, and for the same orbit as Figure 1, but now includes gravitational forces from the Moon and Sun.

Numerical 2-Body

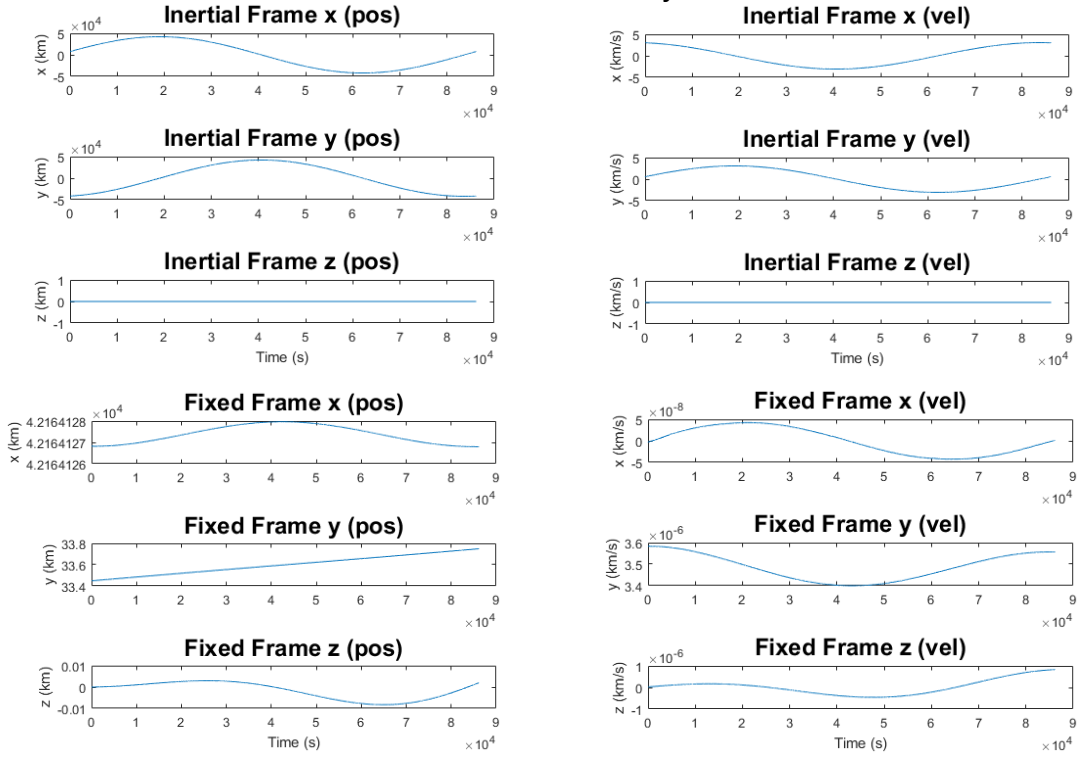


Figure 2.—Numerical 2-Body Orbit Propagation for a GEO Satellite

Numerical N-Body

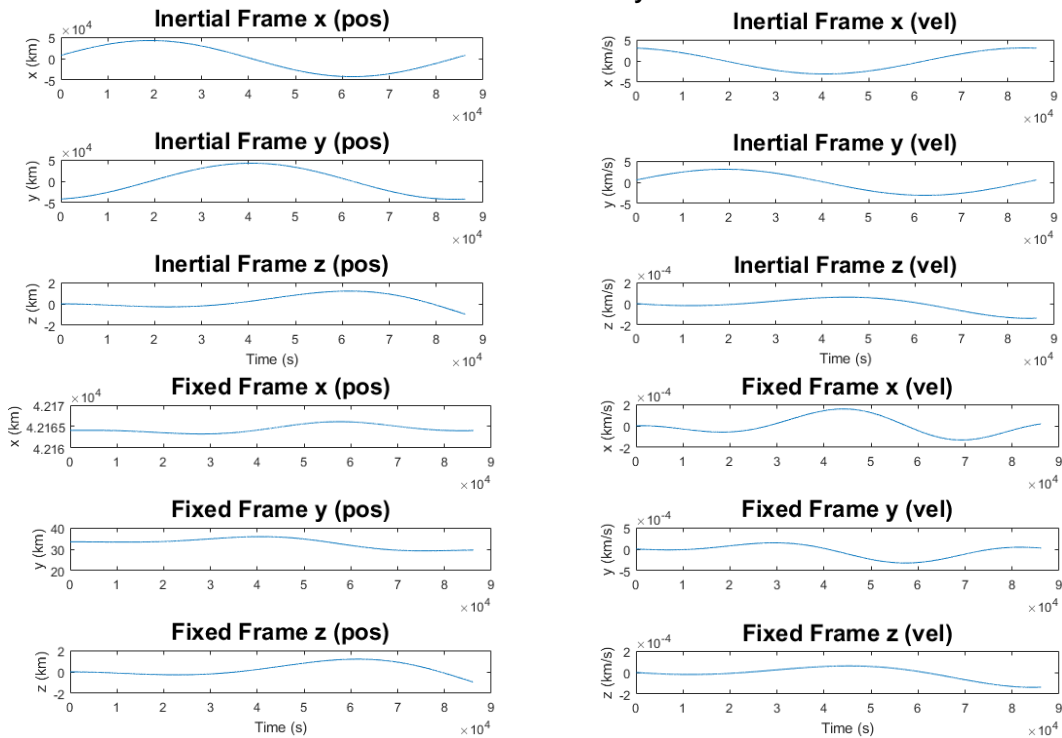


Figure 3.—Numerical N-Body Orbit Propagation for a GEO Satellite

There is now clear motion in the z-axis in the inertial frame as the forces of the Sun and Moon pull the space object out of its original orbit. There are also perturbations on the space object's fixed frame velocities on the order of 0.1 m/s for all axes which cause significant changes to the position profiles.

The final implemented method was a detailed Earth-centric numerical orbit propagation. This type of orbit propagation is meant to allow for the detailed prediction of near-Earth orbits. This model can consider nongravitational parameters (i.e., atmospheric drag and solar pressure). It can also account for solar and lunar gravity, as well as the gravity field of the nonspherical Earth.

Preliminary validation work was performed in order to test the validity of the orbit propagation code. GPS azimuth and elevation data was taken from a GPS antenna at the NASA Glenn Research Center (GRC) on the afternoon of July 19, 2017. In order to compare the results, effective azimuth and elevation angles were calculated from the ground station position and the propagated satellite coordinates in Earth's fixed frame. The differences found between the calculated and measured values are shown in Figure 4.

The figure on the left shows the difference between the azimuth (degrees away from north) values and the figure on the right shows the difference between the measured elevation (degrees from the horizon) values. The numbers in the legend refer to GPS satellites' uniquely identifying pseudorandom noise number (PRN number). The satellites are only shown if they were in view during the observation time. All azimuth and elevation differences were within $\pm 0.4^\circ$ with the exception of one satellite (PRN 11).

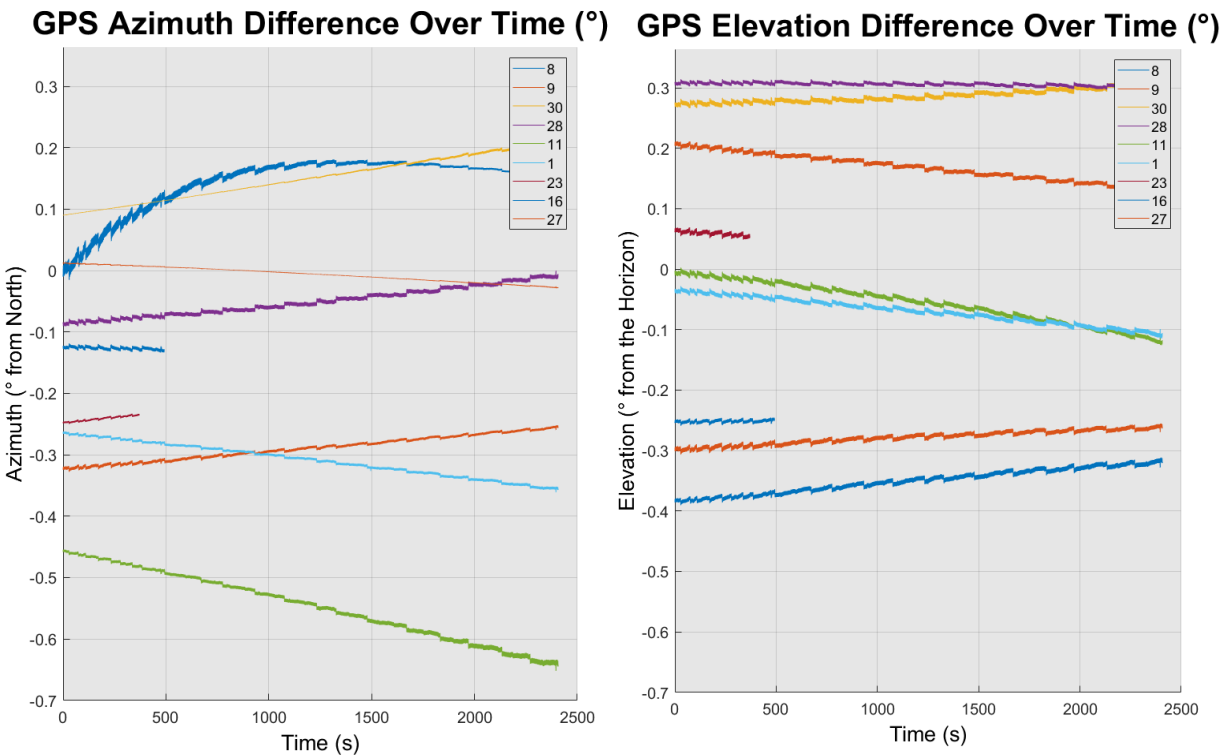


Figure 4.—Orbit Propagation GPS Validation

3.2 Line-of-Sight Analysis

Two modes of satellite LOS analysis were implemented. The LOS access calculation inputs include the inertial frame position, the modified Julian dates associated with each position, the basis type (velocity-in-track-cross-track or radial-normal-binormal for satellites and North-East-down for ground stations), antenna orientation angles, FOV profile and size, and the central body that the positions are relating to (can be any major body). The antenna FOV profile can either be conical (the antenna can only receive signals until some angle from boresight is reached) or rectangular, which is meant to represent a satellite whose antenna can move on a gimbal, creating a swath of possible visibility. In addition to these parameters, ground stations can input minimum elevation constraints in the form of a constant elevation angle, or azimuth-dependent elevation data that the program interpolates between. The generated tools can find inter-satellite and satellite to ground station LOS access accounting for intermediate oblate spheroid body-blocking, antenna FOV geometric restrictions, and relativistic corrections. All LOS access code returns a list (or array for multiple LOSs) of Boolean values expressing whether a LOS can be established. It should be noted that the LOS code works for any major Solar System central body.

The first implemented method was shared-body satellite to ground station LOS access. This code works by finding if the vector connecting the satellite and ground station are above the minimum elevation angle, and then checking to see if the vector is within the FOV sections of both the ground station and the satellite antennas. Multiple ground stations can be passed and code is provided to generate quasi-equidistant surface points so that they user can characterize the planetary coverage for a specific satellite. The code also outputs the average visibility (availability) for each ground station. This type of analysis is shown in Figure 5.

Figure 5 shows the LOS availability between 300 ground stations looking directly up and a nadir-facing Molniya orbit Earth satellite over a single day (two Molniya orbits). A Molniya orbit is a highly eccentric orbit designed to minimize the rate of change of perigee and maintain high levels of visibility with the Northern Hemisphere (~11 hr/orbit) (Ref. 2). The ground stations are differing shades of green

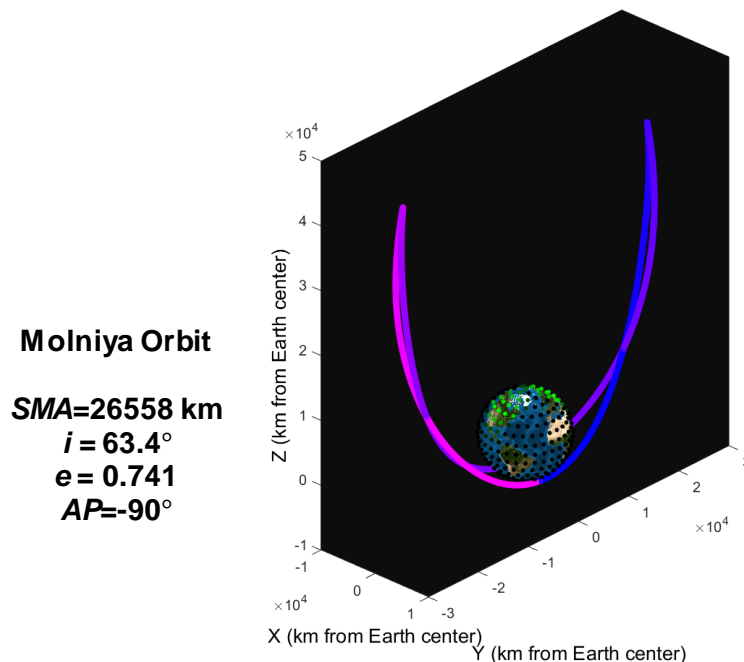


Figure 5.—LOS Availability between Equidistributed Ground Stations and a Molniya Orbit Satellite

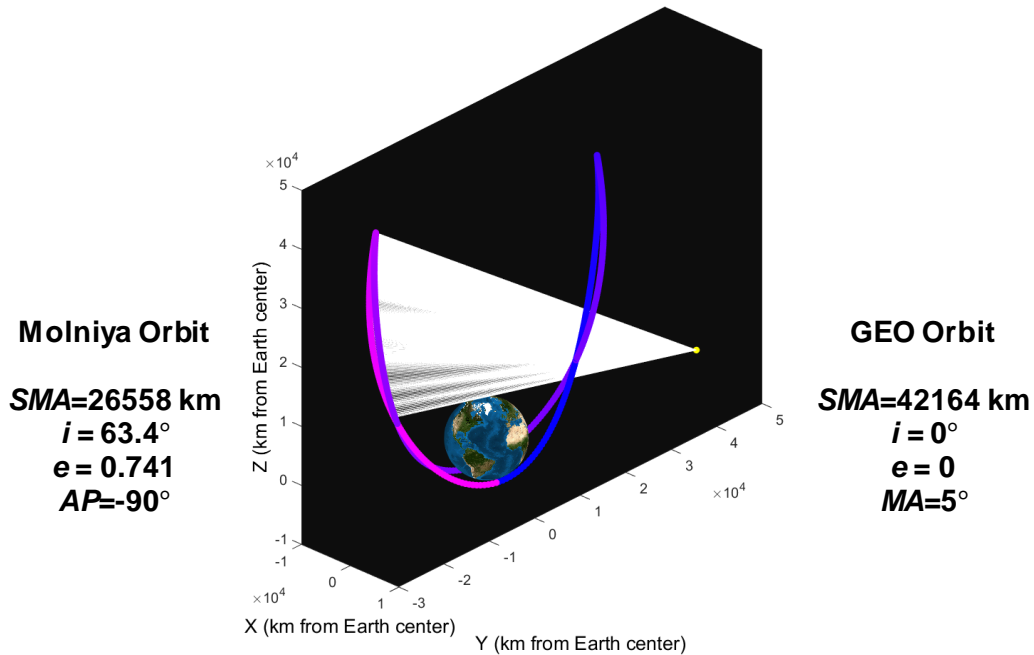


Figure 6.—LOS Availability between GEO and Molniya Orbit Satellites

based on the how often they can communicate with the satellite (from black at 0 percent availability to fully green at 42 percent availability). In this figure, the green is focused on both North America and Russia, as expected for a Molniya orbit.

The second implemented method was satellite to satellite LOS access. This code works by finding if the vector connecting the satellites (with light-time corrections if desired) is blocked by any (properly rotated) intermediate oblate spheroid, and then checking to see if the vector is within the FOV sections of both satellite antennas. This type of analysis is shown in Figure 6.

Figure 6 shows the LOS availability between a nadir-facing GEO satellite and a nadir-facing Molniya orbit satellite over a single day (two Molniya orbits). The white lines connect the satellites at the times when they can see each other. This happens when these satellites are facing each other and the Earth is not between them.

3.3 Visualization

Some visualization capabilities were developed in order to aid in debugging the orbit propagation and LOS calculations. These visualizations can alternatively be useful for mission planning purposes. Three-dimensional plots and animations of planetary movement, satellite orbits, and LOS accesses can be created. Several example scripts which demonstrate the use of the analysis capabilities also have these visualization components that allow the user to display intuitive animations to understand their results. The planetary bodies and satellites are shown in their proper positions and orientations as the animations step through each time point. Satellites have cyan sections showing the geometric antenna FOV constraints to the user so that they can see where the satellite antenna is pointing. This helps the user figure out why the LOS is available or unavailable. These animations can be exported as movies. A series of screen captures of these visualizations are shown in this section.

Figure 7 shows screen captures from the 3D animation of the LOS availability analysis shown in Figure 5. In this case, the body is held fixed and fixed frame positions of the satellite are used. The ground stations are green when the satellite and ground station can see each other, and black when they cannot.

Figure 8 shows screen captures from the 3D animation of the LOS availability analysis shown in Figure 6. In this case, the body is again held fixed and fixed frame positions of the satellites are used. The two satellites are connected by a white LOS line. The white line is solid when the satellites can see each other, and is dashed when they cannot. This animation clearly shows the user when the LOS breaks due to central body blocking or the LOS line exiting the antenna FOV.

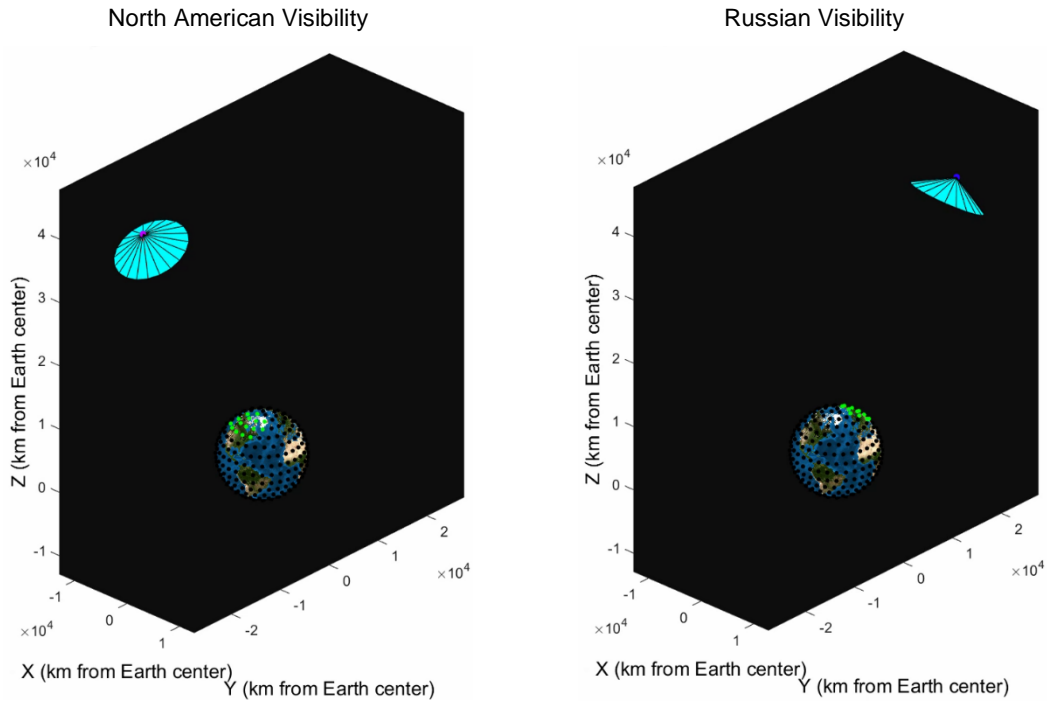


Figure 7.—Animation Stills for LOS between Equidistributed Ground Stations and a Molniya Orbit Satellite

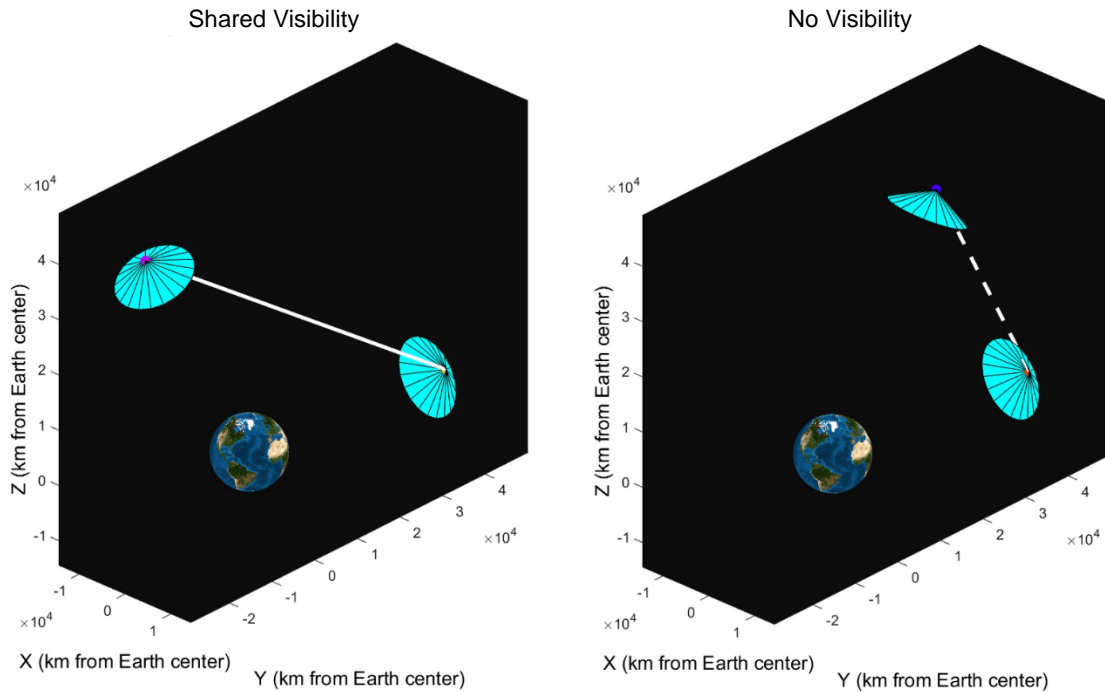


Figure 8.—Animation Stills for LOS between GEO and Molniya Orbit Satellites

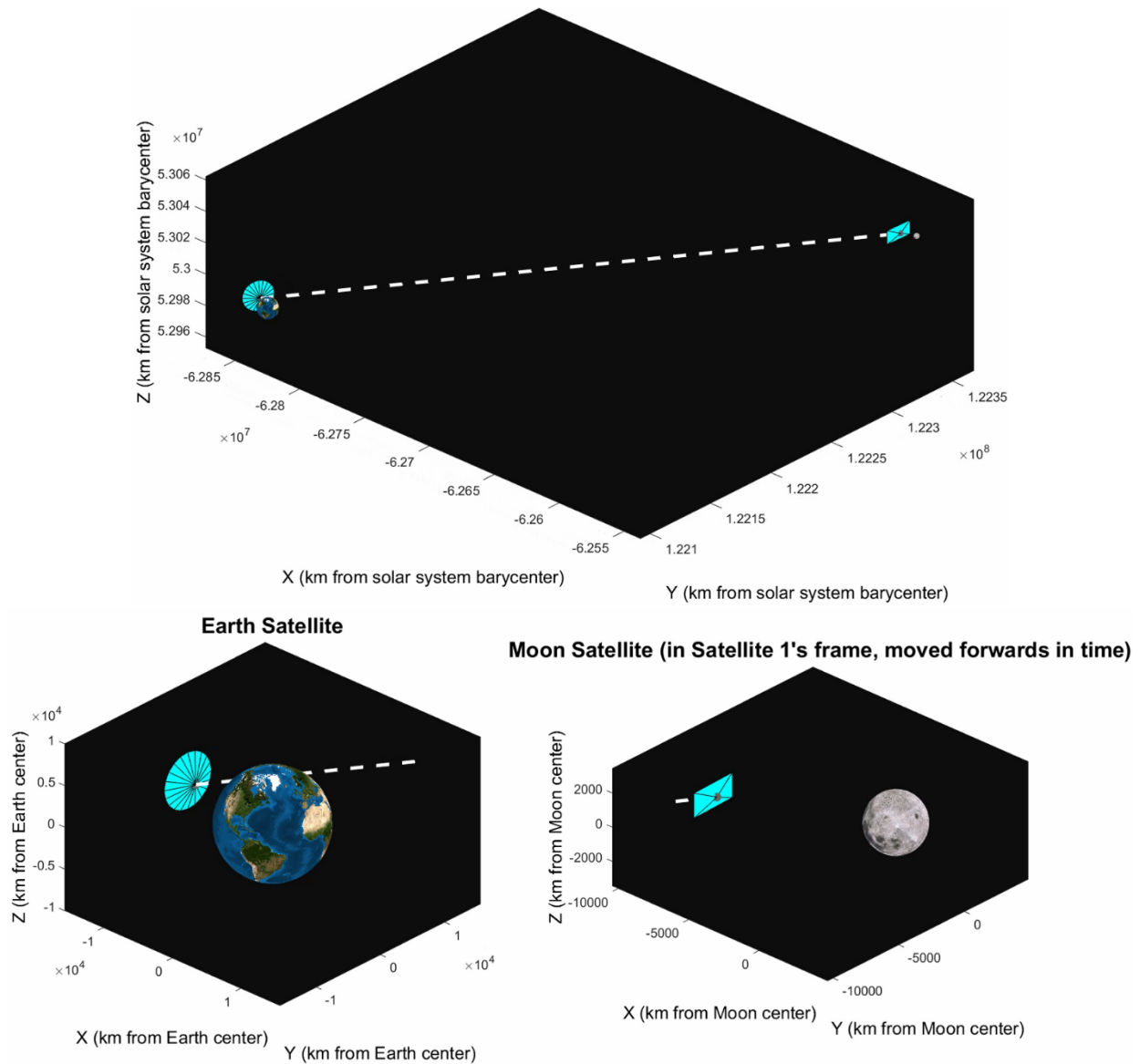


Figure 9.—Animation Stills for LOS between Satellites Orbiting the Earth and Moon

Figure 9 shows screen captures from the 3D animation of the LOS availability analysis shown in Figure 6. In this case, the body is again held fixed and fixed frame positions of the satellites are used. The two satellites are connected by a white LOS line. The white line is solid when the satellites can see each other, and is dashed when they cannot. This animation clearly shows the user when the LOS breaks due to central body blocking or the LOS line exiting the antenna FOVs.

Figure 10 shows screen captures from a 3D animation of the LOS availability between zenith-facing satellites orbiting the Earth and Moon. In this case, the bodies and the satellites are moving. The top plot shows their updating positions in the ICRF. The bottom plot shows their updating positions in central body inertial frames (bottom plots). This figure shows a time when the Moon satellite would not receive a signal sent from the Earth Satellite. Figure 10 shows the same situation but at a later time when the Moon satellite would receive a signal from the Earth satellite.

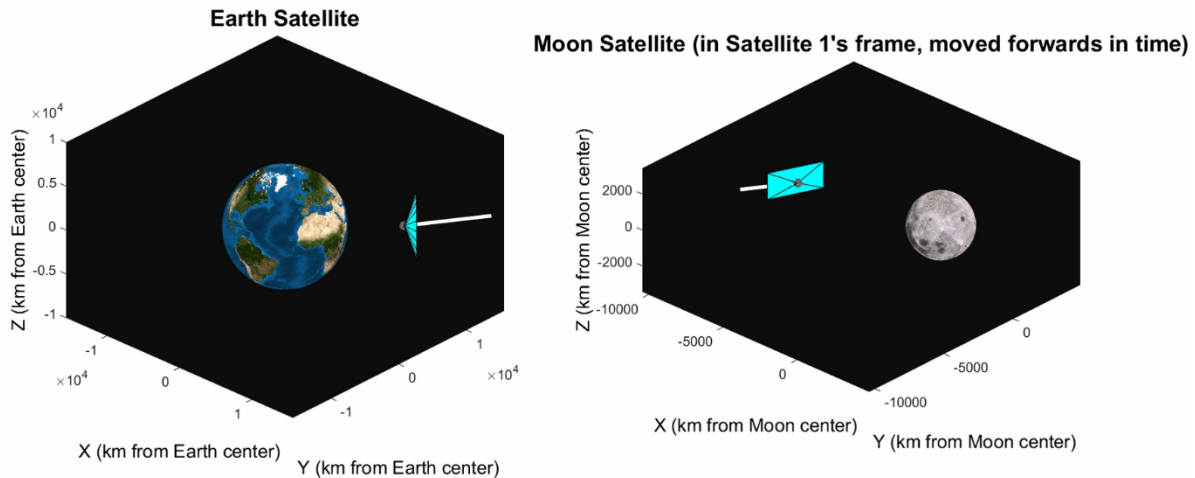


Figure 10.—Animation Stills for LOS between Satellites Orbiting the Earth and Moon

4.0 Conclusion

The orbit propagation and LOS analysis code developed is going to be the basis for SCENIC’s broad analysis capabilities, and, in fact, it is already being used by members of the SCENIC program to perform dilution of precision computations and calculate atmospheric attenuation values for planned NASA missions. Modified versions of this code has also been used to create visualizations for those types of analysis. The code quickly provides useful basic analysis results and is easy to operate by non-authors. The planetary data utilized by the programs is regularly updated and publicly available, ensuring the code maintains relevance over time. The modularity built into the code allows for it to be used as its own toolbox and will make it simple to add on additional capabilities.

Future work can be done in several areas. The orbit propagation code assumes that the space object doesn’t create any propulsive forces. In this vein, trajectory propagation (for deep-space, launch, and suborbital trajectories) could be a next step for this project. Currently, the code uses general fixed-frame coordinate transformations provided by the basic SPICE kernels. More detailed versions of these fixed-frame coordinate transformations exist, and these could be implemented. For the LOS access code, stellar aberration corrections could be added, and it could be condensed and generalized. The LOS functions are split into shared body and separated body versions, but this could be simplified into a single script for both the satellite to ground station and inter-satellite case. There is also only explicit functionality for satellite to ground station connections for a shared body. A separated body case could be added. Additional orbit propagation and LOS validation testing would also be valuable for characterizing the real world performance of the tools. Finally, the developed capabilities need to be continually provided to the SCENIC program for integration into the SCENIC UI.

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