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Asteroid Redirect Mission (ARM) using Solar Electric Propulsion (SEP) for Research, Mining, and Exploration Endeavors of Near-Earth Objects (NEOs)

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By

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Asteroid Redirect Mission (ARM) using Solar Electric Propulsion (SEP) for research, mining, and exploration endeavors of Near-Earth Objects (NEOs)

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The feasibility of relocating a small (~500,000 kg) Near-Earth Asteroid (NEA) to High Earth Orbit via Solar Electric Propulsion (SEP) is evaluated with the orbital simulation software General Mission Analysis Tool (GMAT). Using prior research as a basis for the mission parameters, a retrieval mission to NEA 2008 HU4 is simulated in two parts: approach from Earth and return of the Asteroid Redirect Vehicle (ARV) with the asteroid in tow. Success of such a mission would pave the way for future missions to larger NEAs and other deep space endeavors. It is shown that for a hypothetical launch time of 24 May 2016, the ARV could arrive within 25 km of 2008 HU4 on 28 Jun 2017 with a Delta V of 0.406 km/s, begin return maneuver on 08 Dec 2017 and reach Earth altitude of 450,000 km by 23 Apr 2026 with a Delta V of 44.639 m/s.

DEDICATION

I offer my sincerest appreciations to all of my family, my friends, and my teachers and professors throughout my life who believed in me and helped me get this far in achieving my dreams. I would not have been able to accomplish this feat without these people in my life.

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NOMENCLATURE

ARM – Asteroid Redirect Mission

ARV – Asteroid Redirect Vehicle

AU – Astronomical Unit (1 AU = 149,597,870.700 km)

ΔV – Delta V

ECI – Earth-Centered Inertial coordinate frame

Ec – Ecliptic plane

Eq – Equatorial plane

GMAT – General Mission Analysis Tool

HEO – High Earth Orbit

HET – Hall Effect Thruster

HLO – High Lunar Orbit

I_{sp} – Specific Impulse

J2000 – 12h (midday) on 1 Jan 2000 as reference frame

LEO – Low Earth Orbit

NEA – Near-Earth Asteroid

NEO – Near-Earth Object

SEP – Solar Electric Propulsion

VNB – Velocity-Normal-Binormal non-inertial coordinate system

CHAPTER I

INTRODUCTION

In recent years, interest in acquiring Near-Earth Asteroids (NEAs) for the purpose of research, mining, and expanding upon deep-space interplanetary missions has grown substantially, along with the technologies available to accomplish such feats. While controversial, there are numerous advantages and benefits, such as allowing for synergy with near-term human exploration, bringing hundreds of tons of material within usable range of Earth, which would be exorbitantly expensive to launch into orbit from Earth, test-proven methods for planetary defense against larger, more dangerous asteroids, and the return of valuable resources which can greatly reduce the amount of materials needed to be launched into orbit for subsequent missions. [1] This study aims to examine one particular proposed Asteroid Redirect Mission (ARM) to a small, ~8-meter-diameter, ~500-metric-ton NEA as a proof-of-concept for future endeavors.

1.1 Previous Research

In order to properly grasp the magnitude of an Asteroid Redirect Mission, prior research is needed to form a basis for the study. The primary material that this study will be referencing is the work by Sam Wagner and Bong Wie, which also investigates the feasibility of remotely capturing an NEA and returning it within High Lunar Orbit for ease of access for future missions. Asteroid candidates are selected based on their distance to Earth (in this case, less than 0.2 AU) as well as their relative velocity to Earth (a maximum allowable velocity of 3.0 km/s.) [2]

Two types of missions are possible for the asteroid candidates that meet the specified requirements above. The first type involves capturing a small, 7-8-meter-diameter asteroid with a mass of roughly 500 metric tons, or 500,000 kg in its entirety and towing it back to Earth. The second mission type involves locating a much larger asteroid and retrieving a small piece of it, such as a boulder on the surface, of approximately the same size as given in the first mission type. This study will focus on the first mission type.

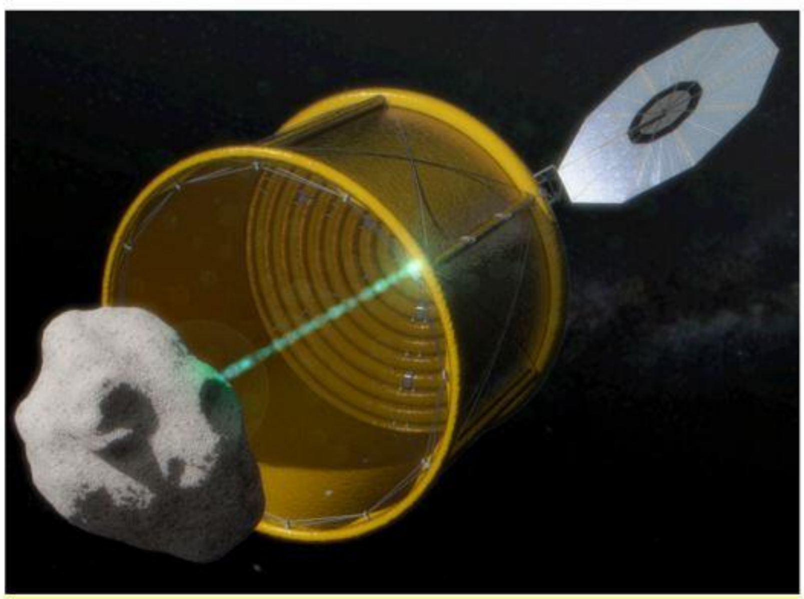


Figure 1.1 NEA Whole-Capture Concept [3]

A secondary reference being used to formulate this study is the work by Brophy and Friedman, which provides a more detailed analysis of other aspects of the Asteroid Redirect Mission. Most notably, Brophy and Friedman address the matter of costs involved in such a mission, estimated by the NASA GRC COMPASS team to have a full-cycle cost (capture and return) of approximately $2.6 billion. [1] Their research also covers the launch stage of the mission as well as the lunar gravity assist stages, which this study omits due to time constraints, which will be discussed in the concluding chapters.

Both works by Wagner and Wie and Brophy and Friedman use very similar spacecraft specifications and launch windows, making them ideal for comparison with the results of this study. These comparisons will be analyzed in depth in the following chapters.

1.2 Research Objectives

The main goal of this study is to simulate the capture of a Near-Earth Asteroid based on the scenarios given in Wagner and Wie's research as well as the research of Brophy and Friedman to test both the feasibility of an Asteroid Redirect Mission as well as the versatility of the orbital simulation program being used for this study, which will be discussed in detail. The simulation will ideally provide similar timeframes and ΔV requirements as found by the previous research.

The orbital simulation will be conducted in two sets of code: the script responsible for the approach for rendezvous with the asteroid, and the script responsible for the subsequent return of the spacecraft with the asteroid in tow. The data and orbital telemetry from the first script will be directly inputted to the second script, making the asteroid's return directly dependent on the results of the approach. Once the final results have been acquired, the overall the orbital simulation program used will be evaluated in terms of accuracy and effectiveness.

CHAPTER II

ORBITAL MECHANICS

2.1 Orbital Elements

A few key orbital terms will be discussed in order to better understand the process and dynamics of the orbital simulation being tested in the study. The following section will provide definitions and explanations as needed.

2.1.1 Periapsis, Apoapsis

In *The Fundamentals of Astrodynamics and Applications*, a periapsis and an apoapsis are defined as the most extreme points in an elliptical orbit, with the periapsis being the closest point to the center of attraction and the apoapsis being the farthest. [4] These orbital points will be targeted in order to reach the asteroid and return it to Earth.

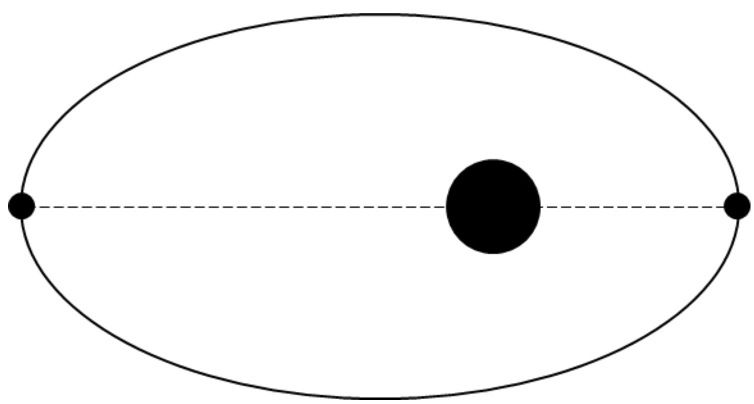


Figure 2.1 Illustration of Apoapsis and Periapsis

2.1.2 Elliptic Orbital Elements

There are six classical orbital elements that will be used to define initial parameters of the spacecraft in this study. They are defined as follows: [4]

- Semi-Major Axis (a) – half the distance across the orbit
- Eccentricity (e) – defines the roundness or flatness of an orbit
- Inclination (i) – the tilt of the orbital plane
- Right Ascension of the Ascending Node (RAAN, Ω) – angle in the equatorial plane measured positive and eastward from the $\hat{I}$ unit vector to the ascending node.
- Argument of Periapsis (ω) – measured from ascending node in direction of orbiting motion, locating the periapsis
- True Anomaly (ν) – angular displacement measured from periapsis to position vector through the direction of motion

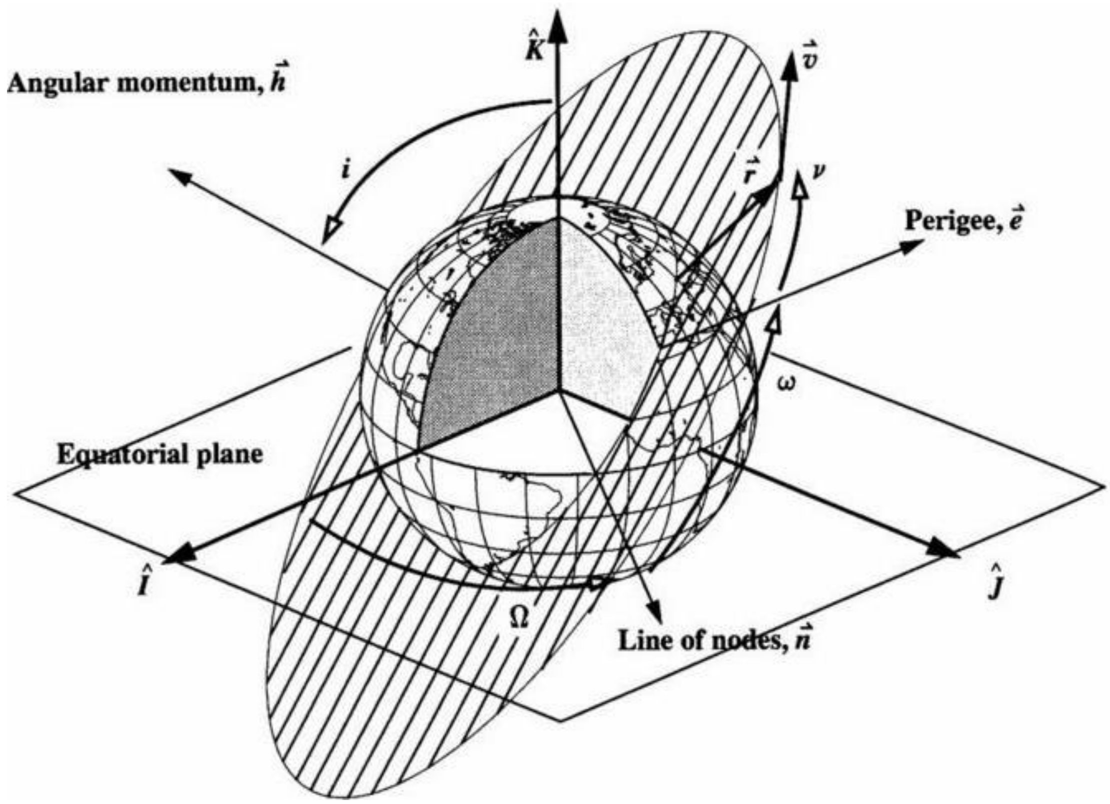


Figure 2.2 Graphical Representation of the Classical Orbital Elements. [4]

2.2 Orbital Maneuvers

The following orbital maneuvers are regularly referenced in the orbital simulation and should be understood on definition.

2.2.1 Impulsive Burn, Finite Burn

The official reference guide for the orbital simulation program being used in this study defines an Impulsive Burn as a maneuver that initiates an instantaneous ΔV. It works on a three-variable solution: the three directions in a coordinate system determining the vector of the burn. A finite burn, as implied by its name, burns for a finite amount of time to achieve a ΔV gradually. This is a more realistic albeit more complex as it introduces a fourth variable of time along with the three directional variables. [5]

2.2.2 Delta-V (ΔV), Specific Impulse

ΔV, or delta-V, as defined by Vallado, is the “ideal rocket equation” [4], or

$$\Delta V = g \cdot I_{sp} \cdot \ln\left(\frac{m_{init}}{m_{init} - m_{prop}}\right) \tag{2.1}$$

where

g = gravity

ln = the natural logarithmic function

m_{init} = the vehicle’s initial mass

m_{prop} = the propellant mass

I_{sp} = specific impulse

This equation represents the maximum change in velocity of the vehicle.

I_{sp}, the specific impulse, is defined to be the total impulse per unit weight of propellant,

$$I_{sp} = \frac{I_{tot}}{m_{prop} \cdot g} \tag{2.2}$$

wherein the total impulse I_t is the thrust force F integrated over burning time t.

$$I_{tot} = \int_0^t F\, dt \tag{2.3}$$

When reduced to constant thrust and miniscule start and stop times, impulse is proportional to the total energy released by all of the propellant in a propulsion system. [6]

$$I_{tot} = Ft \tag{2.4}$$

2.2.3 B-Plane Trajectory

One particular aspect of the orbital simulation involves utilizing a particular planar coordinate system called the B-Plane in order to achieve targeting during a gravity assist. In the simplest sense, it can be considered a target attached or connected to the assisting body. It must be perpendicular to the incoming asymptote of the approach hyperbola. The following figures will provide a better understanding of the geometry of this coordinate system. [5]

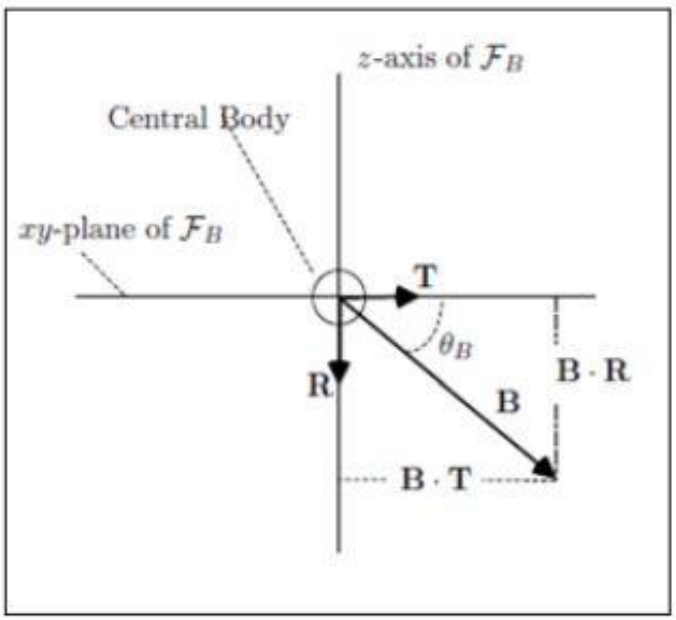


Figure 2.3 B-Plane Geometry

B-Plane geometry as viewed perpendicularly to the B-Plane. B·T (BdotT) and B·R (BdotR) values can be specified in the orbital simulation.

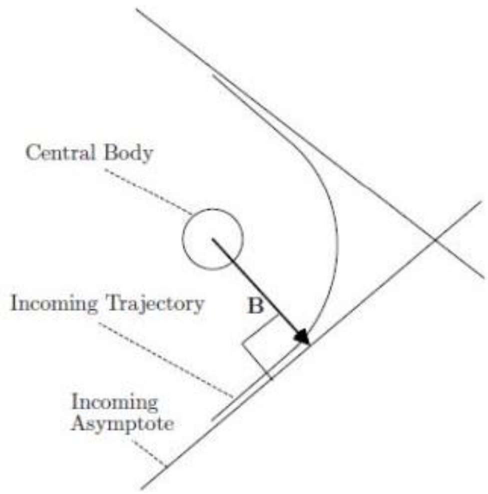


Figure 2.4 B-Vector Perpendicular View

Another view displaying the B-vector as seen from a perpendicular viewpoint in relation to the orbit plane.

While the candidate asteroid used in the study is much too small to have a gravitational pull great enough to make proper use of a gravity assist maneuver, it is still useful in setting the spacecraft up for an optimal pre-capture approach distance, which will be further discussed in a later chapter.

CHAPTER III

MISSION OBJECTIVES

3.1 Goals and Specifications

The primary objective of this study is to simulate an Asteroid Redirect Mission using orbital simulation tool by recreating a given scenario as provided by prior research and compare the results. In order to accomplish this, there are numerous specifications as set forth by the prior research that must be met.

3.1.1 Acquire Orbital Parameters for Asteroid Rendezvous

The first goal of the study is to acquire orbital parameters and telemetry for a given candidate asteroid. This data, containing calculated positions of the specified celestial object over regularly spaced intervals of time, is known as an ephemeris. The orbital simulation program contains numerous ephemeris sets for common celestial bodies such as the planets of the Solar System, but the candidate asteroid will require its own ephemeris to be imported into the program. An ideal location to acquire such data is the Solar System Dynamics website hosted by NASA's Jet Propulsion Laboratory (JPL) at ssd.jpl.nasa.gov.

This process involves using the JPL HORIZONS online Solar System data and ephemeris computation service. The service is accessed through a telnet interface and provides highly accurate ephemerides of celestial objects, with 711,958 asteroids on file. [7] Once connected through horizons.jpl.nasa.gov telnet port 6775, a terminal session is created in which the user is taken through a series of prompts to generate the appropriate data. Once generated, the ephemeris is saved in a batch-style input file which can then be retrieved via FTP access for a limited amount of time.

3.1.2 Reach Optimal Range of Asteroid

According to the research by Brophy and Friedman, a small NEA of approximately 7-m has less than 10^{-6} m/s^2 of gravitational force. Therefore, achieving a stable orbit around the NEA would be significantly difficult and would instead require an incremental approach. An NEA of 7-m in diameter would be visible from a distance of 100,000 km to 200,000 km with a navigation camera onboard the Asteroid Redirect Vehicle similar to the framing camera used on the NASA's Dawn spacecraft. The optimal rendezvous point for the ARV would be approximately 20-30 km away from the asteroid. [1] This is the ideal range for pre-capture maneuvers. As such, the goal for the orbital simulation will be to get the ARV within 25 km of the candidate asteroid. Achieving that amount of accuracy will require the use of B-Plane trajectory targeting.

3.1.3 Acquire Orbital Parameters for Asteroid Return

Once the spacecraft has successfully captured the asteroid, it will need to be towed back to a High Earth Orbit which can then be transitioned to a High Lunar Orbit. With the added mass of the asteroid, this process will take much longer and will require careful trajectory planning. Once the spacecraft with the asteroid in tow is within 380,000 km to 450,000 km of Earth, data will be collected and the results will be compared with the prior research.

3.2 Requirements for Mission Success

The spacecraft's cruise to the asteroid and the subsequent return of the spacecraft with the added mass of the asteroid to Earth are the two main aspects of the Asteroid Redirect Mission being simulated. The initial launch, separation and deployment of the spacecraft, initial lunar gravity assist, and return lunar gravity assist to lunar high orbit with the asteroid are omitted for the sake of succinctness in this particular study. In order to consider this study a successful endeavor, the orbital simulation must recreate to a reasonable degree of accuracy an ARM scenario given in the prior research, namely that of Wagner and Wie. The ΔV values and timeframes should be very similar in the simulation, and the results of the simulation itself should be easily repeatable.

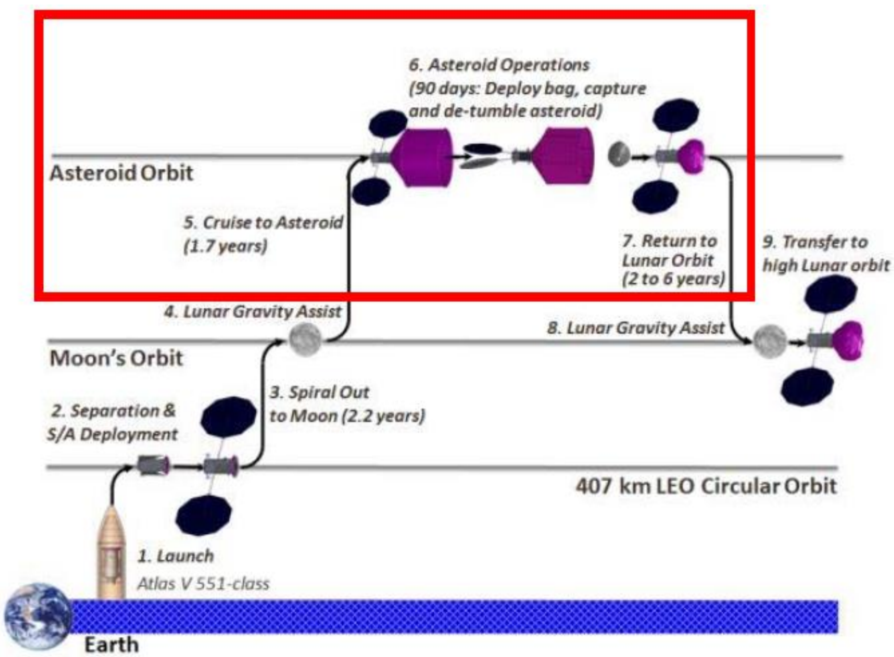


Figure 3.1 Asteroid Redirect Mission Cycle Diagram [1]

A visual representation of the entire mission cycle of an Asteroid Redirect Mission as given by the research of Brophy and Friedman. The outlined portion represents the main focus of this study

3.2.1 Cost and Timeframe

Based on NASA's NLS-II agreement, the cost of delivering a single kilogram of mass to High Lunar Orbit (HLO) would be roughly $100,000; delivery of 500 metric tons of material to HLO from the surface of Earth would be exorbitant, totaling around $50 billion. According to estimates by the NASA GRC COMPASS team, a full mission life-cycle cost of an Asteroid Redirect Mission delivering 500 metric tons of material in the form of the asteroid would be around $2.6 billion by comparison. [1] This takes into consideration the cost of the spacecraft, the launch to low-Earth orbit (LEO) on an Atlas V-class rocket, and the cost of propellant required to power the spacecraft to the asteroid and return with the asteroid in tow. With all costs considered, it would be nearly 20 times cheaper to acquire the material from a Near-Earth Asteroid rather than launching it from Earth.

The timeframe for such a mission would include a total flight time of approximately 6 to 8 years, depending on the type of propulsion and available propellant onboard the spacecraft, as well as the size of the asteroid being captured. This does not include the time required to position the spacecraft from LEO to HLO for the lunar gravity assist, which would take an additional 2.2 years. [1]

3.3 Potential Benefits

Undertaking a mission to relocate an asteroid to a relatively accessible orbit around Earth has several key benefits that should be considered. In addition to being far more cost-effective in putting material and resources into HLO, the Asteroid Redirect Mission would allow for human NEA exploration endeavors much sooner, providing valuable knowledge and experience for future missions to larger asteroids much farther away. More time would also be allotted to such expeditions due to the close proximity to Earth, allowing for more thorough research. Additionally, having hundreds of tons of material at such readily accessible distances would allow for mining and extraction of valuable resources such as propellants, radiation shielding materials, and other valuable materials including life support fluids depending on the composition of the captured asteroid. Accomplishing a complete asteroid relocation would be invaluable to the ever-expanding framework of planetary defense as well, as it would serve as a proof-of-concept for handling larger, potentially more dangerous NEAs. This ambitious mission can also serve as an inspiration for younger generations to take interest in space programs and continue building upon the foundation set forth for pushing humanity toward even bolder endeavors in space exploration. [1]

CHAPTER IV

ASTEROID SELECTION

4.1 Type, Composition, Size

The work by Mazanek et al. provides a useful table for various types of asteroids, potential resources contained within, and various methods to extract those resources.

Type of Asteroid	Possible Available Resources	Methods to Extract Resources
C-Type (Carbonaceous)	Volatiles (simple compounds of hydrogen, oxygen, carbon, sulfur and nitrogen) Rarely nitrogen, halogens and noble gases.	Heating: Microwave or Solar Thermal processes
	Water – primarily as hydrated minerals	A. Heating: Microwave or Solar Thermal processes B. Ionic Liquid Acid dissolution, C. H_2SO4 or HF dissolution
	Metal Oxides (for oxygen)	A. Ionic Liquid Acid dissolution B. H_2SO4 or HF dissolution C. Hydrogen Reduction D. Carbothermal Reduction E. Molten Oxide Electrolysis
	Metal Oxides (for metal)	A. Molten Oxide Electrolysis B. Ionic Liquid dissolution followed by electrolysis
	Elemental Metals	A. Heating: Microwave or Solar Thermal processes B. Molten Oxide Electrolysis C. Ionic Liquid dissolution followed by electrolysis
S-Type (Silicaceous)	Platinum Group Metals	A. Heating: Microwave or Solar Thermal processes B. Molten Oxide Electrolysis C. Ionic Liquid dissolution followed by electrolysis
	Metal Oxides (oxygen and metals)	A. Molten Oxide Electrolysis B. Ionic Liquid dissolution followed by electrolysis
M-Type (Metals)	Elemental Metals (Primarily Iron and Nickel)	A. Heating: Microwave or Solar Thermal processes B. Molten Oxide Electrolysis C. Ionic Liquid dissolution followed by electrolysis
	Platinum Group Metals	A. Heating: Microwave or Solar Thermal processes B. Molten Oxide Electrolysis C. Ionic Liquid dissolution followed by electrolysis

Figure 4.1 Asteroid Types and Resource Extraction Methods [8]

By and large, the most desirable kind of asteroid to target at present is the carbonaceous C-type asteroid. These C-type asteroids are the most diverse in composition and can contain a varied mixture of metals, complex organic molecules, volatiles which can be used as fuel, water, and dry rock. Due to their low mechanical strength, C-type asteroids can be cut or crushed with relative ease, making them ideal candidates for an Asteroid Redirect Mission. The preferred size of the candidate C-type asteroid is approximately 7 meters in diameter, which can be returned to Earth in the 2020s. [1]
It should be noted that an asteroid of this size poses no risk to Earth as it is much too small to cause any damage should the asteroid return encounter any problems. Even if the asteroid survived atmospheric entry, it would disintegrate before reaching the surface. [9]

4.2 Suitable Candidate

The biggest challenge in locating a suitable candidate asteroid meeting the type and size requirements lies in its orbital parameters. An asteroid with orbital parameters in close vicinity to Earth with a synodic period, or the time taken to make a full revolution and return to the same relative location, of roughly one decade. [1] One such asteroid has already been identified and will be used as the primary asteroid candidate of this study.

4.2.1 Asteroid 2008 HU4

The research by Brophy and Friedman and that by Wagner and Wie both respectively choose the asteroid designated 2008 HU4 as a suitable candidate for a proof-of-concept Asteroid Redirect Mission. 2008 HU4 is approximately 8 meters in diameter and estimated to be roughly 500-1000 metric tons. It also has a ten-year synodic period, with a close approach to Earth in 2016 followed by its next close approach in 2026. [1] In the Wagner study, it is assumed that 2008 HU4 is 500 metric tons, which will be reflected in this study as well. [2]

4.3 2008 HU4 Orbital Parameters

The orbital parameters for 2008 HU4 is generated by NASA JPL's HORIZONS system and shown in the figure below.

```
JPL/HORIZONS                      (2008 HU4)                 2016-Apr-11 13:06:08
Rec #:562998 (+COV)   Soln.date: 2014-Jun-13_09:21:26         # obs: 71 (41 days)

FK5/J2000.0 helio. ecliptic osc. elements (au, days, deg., period=Julian yrs):

  EPOCH=  2454601.5 ! 2008-May-15.00 (TDB)         Residual RMS= .5377
   EC= .07784132155374582  QR= 1.011081468627788   TP= 2454569.2058606502
   OM= 222.9637313337805   W=  339.6849407571114   IN= 1.31810322401867
   A= 1.096428946839561    MA= 27.72407401273707   ADIST= 1.181776425051334
   PER= 1.1481             N= .858486208           ANGMOM= .01795775
   DAN= 1.01564            DDN= 1.1756             L= 202.6536185
   B= -.4575862            MOID= .00704657         TP= 2008-Apr-12.7058606502

Asteroid physical parameters (km, seconds, rotational period in hours):
   GM= n.a.                RAD= n.a.               ROTPER= n.a.
   H= 28.2                 G= .150                 B-V= n.a.
                           ALBEDO= n.a.            STYP= n.a.

ASTEROID comments:
1: soln ref.= JPL#21, OCC=4
2: source=ORB
```

Figure 4.2 Orbital Parameters Generated by NASA JPL HORIZONS [7]

The ephemeris data for 2008 HU4 is generated from 1950 to 2050 to ensure that accurate orbital data is available for an entire century, more than enough data points to model the asteroid's orbit during the mission timeframe. This data is exported from HORIZONS in a separate SPK (short for Spacecraft and Planet Kernel) file and imported into the orbital simulation program.

4.4 Launch Window

The optimal launch window to transfer the spacecraft to 2008 HU4 with the lowest ΔV requirement is 24 May 2016. This date can be adjusted to a later time with minimal impact to the amount of mass returned to Earth, however. [2] For the sake of consistency with the research results of Wagner and Wie, this study will be using 24 May 2016 as the departure date of the spacecraft from LEO. The main goal of this study is to test the feasibility and accuracy of the orbital simulation program, so it will need to follow the mission timeframe provided in the previous research.

CHAPTER V

SPACECRAFT SPECIFICATIONS

5.1 Propulsion, Power, and Propellant Type

The Asteroid Redirect Mission relies on Solar Electric Propulsion (SEP) for its success. Out of all of the previous research, Wagner and Wie provide the simplest design specifications for the spacecraft to be used, so that will be the basis of the design in this study, combined with elements from other studies. The Asteroid Redirect Vehicle in the orbital simulation will utilize two Busek BHT-20K Hall effect thrusters, each with 20 kW maximum power allocation and 1.08 N maximum thrust per thruster. Thruster efficiency is rated at 70% with a specific impulse or I_{sp} of 2,750 s. [2] The SEP system thus requires a 40-kW power supply at minimum for its Solar Electric Propulsion Module, or SEPM. The SEPM will utilize two 50 kW solar arrays, feeding 40 kW into the SEP system. [3]

Xenon is the choice of propellant for this SEP system. The ARV has a fuel capacity of up to 10 metric tons or 10,000 kg of xenon, which would require five 2-ton tanks held within the SEPM core structure. This capacity is scalable up to eight tanks and 16 metric tons of propellant for future deep space missions. [3] For the sake of simplicity, the five fuel tanks will be consolidated into one effective “tank module” in the orbital simulation program.

5.2 Physical Parameters

Based on Wagner and Wie's specifications, the dry mass of the ARV using just two active Busek BHT-20K Hall effect thrusters will have a dry mass of 5,500 kg, so that will be the assumed mass in the orbital simulation program. The overall structure of the spacecraft will match the configuration shown below by Gates et al.

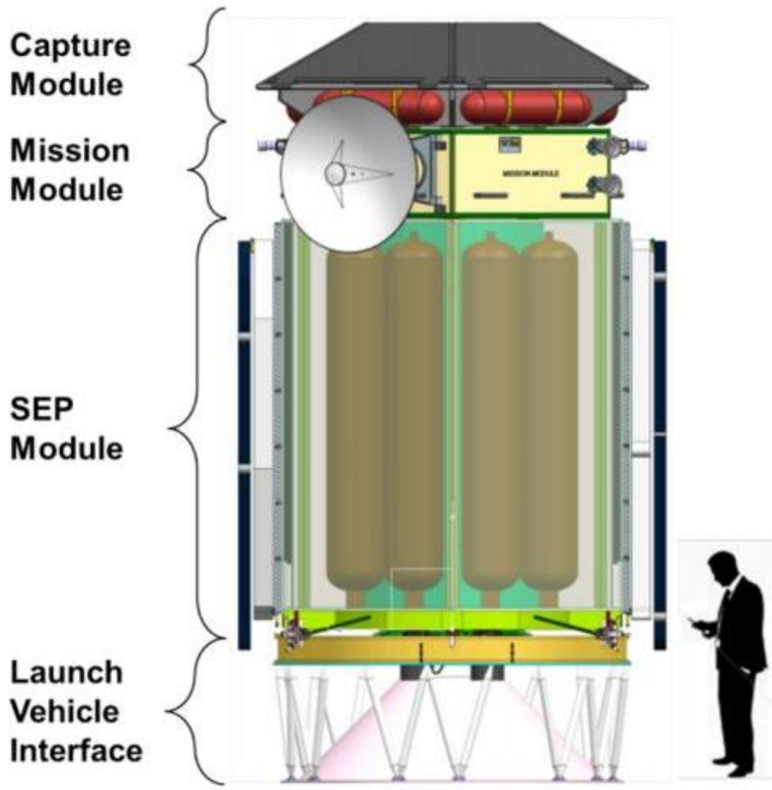


Figure 5.1 Asteroid Redirect Vehicle Configuration, courtesy of Gates et al. [3]

The dimensions and Capture Module configurations of the Asteroid Redirect Vehicle are illustrated in the study conducted by Brophy and Friedman, as presented below. While these aspects are not directly implemented in the orbital simulation program beyond the power, fuel tank, and number of solar panels, they are still important to understand in order to grasp how the actual asteroid capture procedure will be conducted.

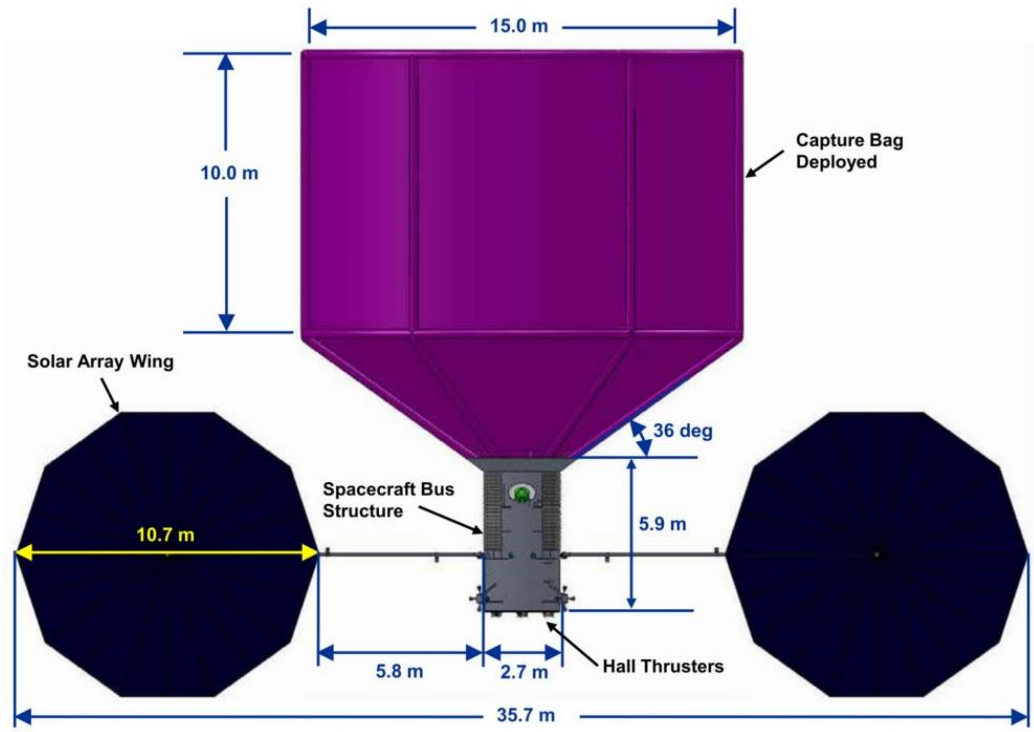


Figure 5.2 Asteroid Redirect Vehicle Dimensions [1]

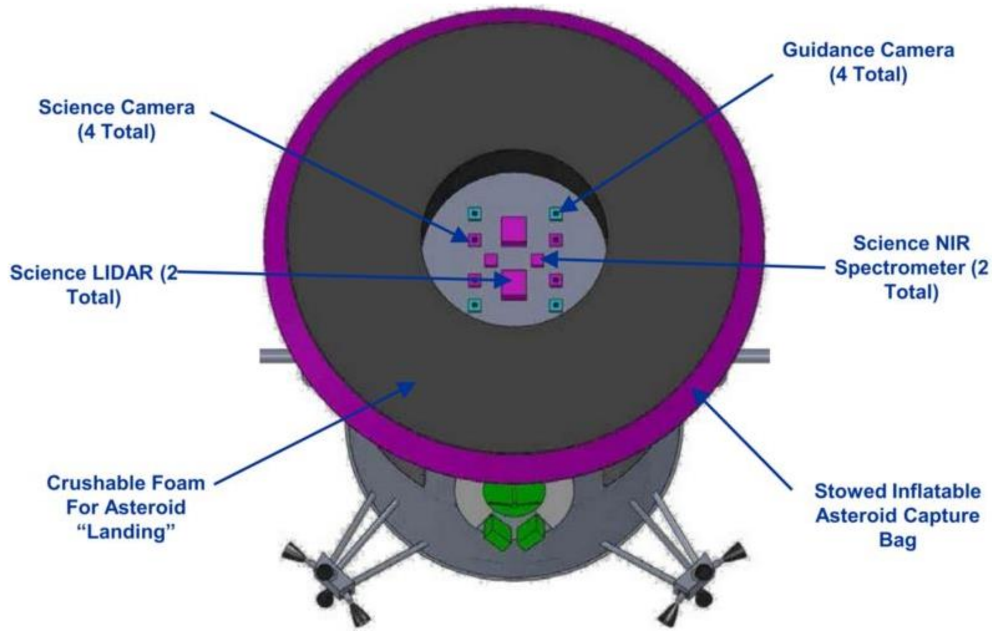


Figure 5.3 ARV Top View of Capture Module [1]

As can be seen in Figure 5.3, the ARV is equipped with a number of science and spectrometry devices for analyzing the asteroid on initial approach and during pre-capture maneuvers. The crushable foam for the asteroid "landing" referenced in the figure is when the spacecraft makes contact with the asteroid as the capture bag is positioned over it, protecting the rest of the spacecraft from a potentially rough collision.

5.2.1 Busek BHT-20K Hall Effect Thrusters

Hall Effect Thrusters (HETs) are chosen for this mission for their high power performance and versatility. HETs function via crossed electric and magnetic fields that produce and accelerate ions, resulting in thrust and acceleration. The particular high power thruster system being used for the ARV is the Busek BHT-20K HET, which can be propelled by either of the noble gases xenon (Xe) or krypton (Kr). [10] The BHT-20K model has an input power range from 5.0 kW to 20.0 kW and approximately 1 N of thrust with a specific impulse of up to 3,000 s. [11]

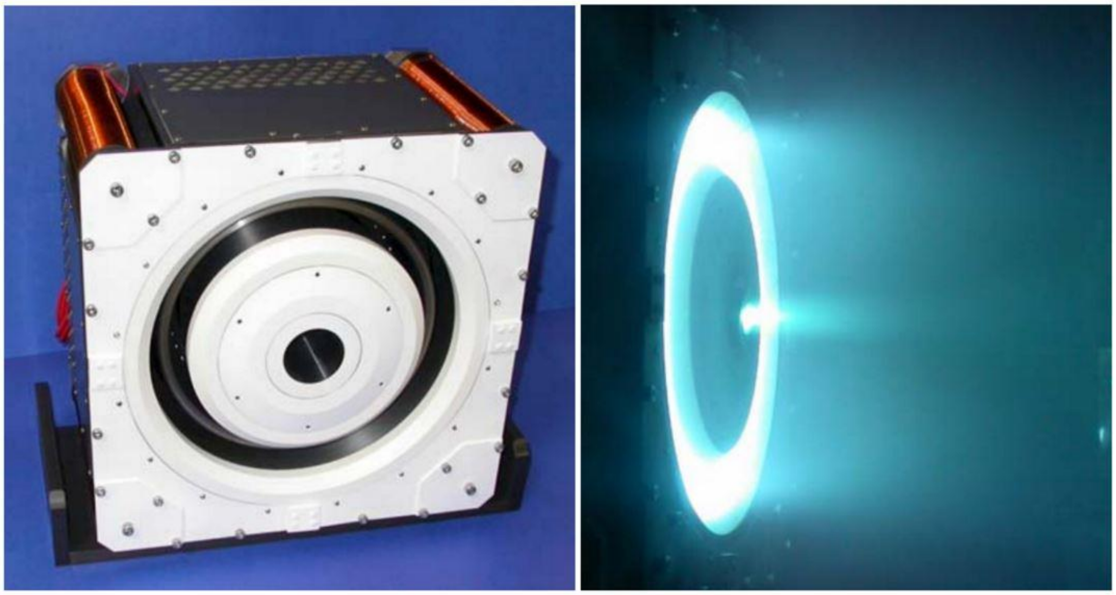


Figure 5.4 Busek Hall Thruster BHT-20K [10]

CHAPTER VI

GENERAL MISSION ANALYSIS TOOL (GMAT)

The orbital simulation program known as the General Mission Analysis Tool will be used for the recreation and evaluation of the Asteroid Redirect Mission as specified by Wagner and Wie. It is highly versatile and free-to-use software with several built-in tutorials, a detailed user's guide, and an extensive amount of online material to aid with its learning curve.

6.1 Development

The General Mission Analysis Tool, or GMAT for short, is a comprehensive, open-source orbital trajectory simulation, analysis, and optimization program developed by a team composed of NASA Goddard Space Flight Center engineers and private industry partners. It is the world's only open-source enterprise, multi-mission orbital simulation software for space mission design, optimization, and navigation. GMAT has undergone a number of iterations, the latest of which is version R2015a, as of November 2nd, 2015. This latest version also introduces solar electric propulsion as a supported propulsion model in the program, making it ideal for this study. [5]

6.2 Uses

GMAT provides a graphical user interface and programming environment ideal for trajectory design and computation incorporating gravitational fields at varying amounts of detail and fidelity as needed by the type of mission. [12] GMAT has found extensive usage in various NASA projects over the years. The first notable usage was for optimization in LCROSS during 2009. The LRO project also benefited from GMAT's use in 2010, along with ARTEMIS (from 2007 to the present), MMS in 2012, ACE from 2009 to 2010, MAVEN in 2013, OSIRIS in 2013, and recently TESS in 2014. In total, GMAT has supported 8 NASA missions to date as well as 8 NASA proposal efforts, and is currently in use by over 30 organizations, with 15 universities, and 12 commercial firms publishing open-literature results. [13]

6.3 Functionality

GMAT has numerous key features that are advantageous for simulating the Asteroid Redirect Mission. These include but are not limited to:

- High fidelity dynamics models for gravity and relativistic corrections
- High fidelity spacecraft modelling
- Impulsive and finite maneuver modeling and optimization
- Propulsion system modeling, in particular solar electric propulsion
- High fidelity Solar System modelling with high fidelity ephemerides, custom celestial bodies, libration points, and barycenters
- Rich set of coordinate systems including J2000, ICRF, fixed, rotating, topocentric, and others
- SPICE kernel propagation
- Interactive 3D graphics and fully featured graphical user interface
- Post computation animation

- Custom targeter controls and constraints
- User defined variables, arrays, and strings
- MATLAB syntax
- Interfaces for MATLAB and Python
- Custom scripting language for complex, custom analysis

CHAPTER VII

APPROACH TRAJECTORY DESIGN

7.1 Resources

In order to build the mission to be simulated in GMAT, the first step is to import the Asteroid 2008 HU4 ephemeris data. This is an SPK kernel file compiled via NASA JPL's HORIZONS online database system, retrieved via telnet prompts and FTP server access. The NASA NAIF (short for Navigation and Ancillary Information Facility) ID for 2008 HU4 is provided as NAIF ID 3409707. [7] Both the SPK file and the NAIF ID (also known as an SPK ID) are inputted into GMAT as a custom celestial body.

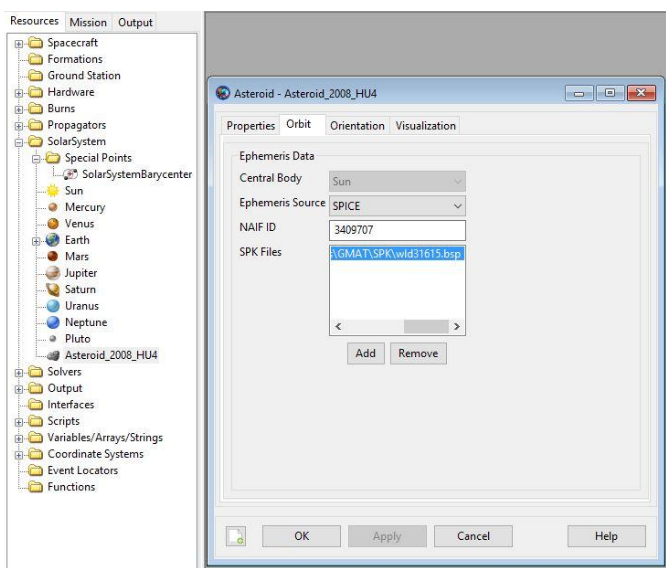


Figure 7.1 GMAT Custom Celestial Body Interface

An asteroid can be added to GMAT by browsing through SolarSystem, right-clicking on Sun, hovering over Add, then selecting Asteroid.

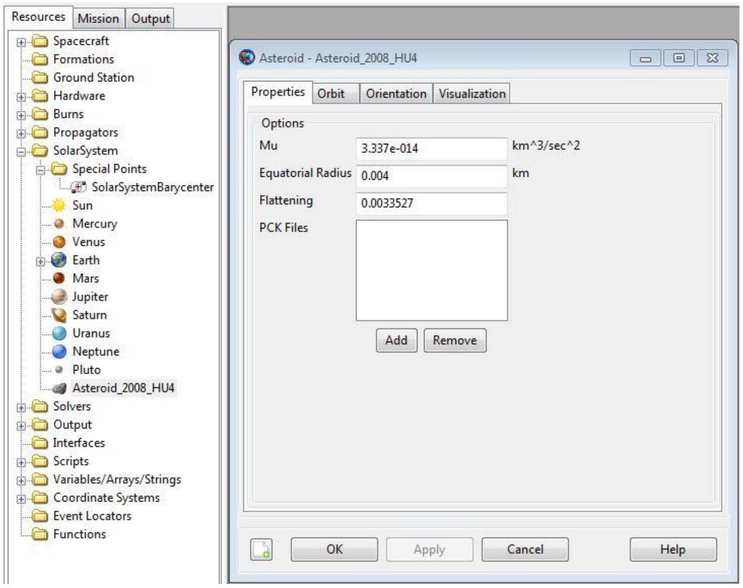


Figure 7.2 Asteroid Properties

For 2008 HU4's properties, its approximate mass of 500,000 kg means it has a gravitational parameter $\mu = 3.337\text{e-}014$ km^3/s^2 (almost negligible.) Its equatorial radius is estimated to be 4 meters, or 0.004 km, as its diameter is estimated to be 8 meters.

The next step is to build the Asteroid Redirect Vehicle through the Spacecraft and Hardware design menus.

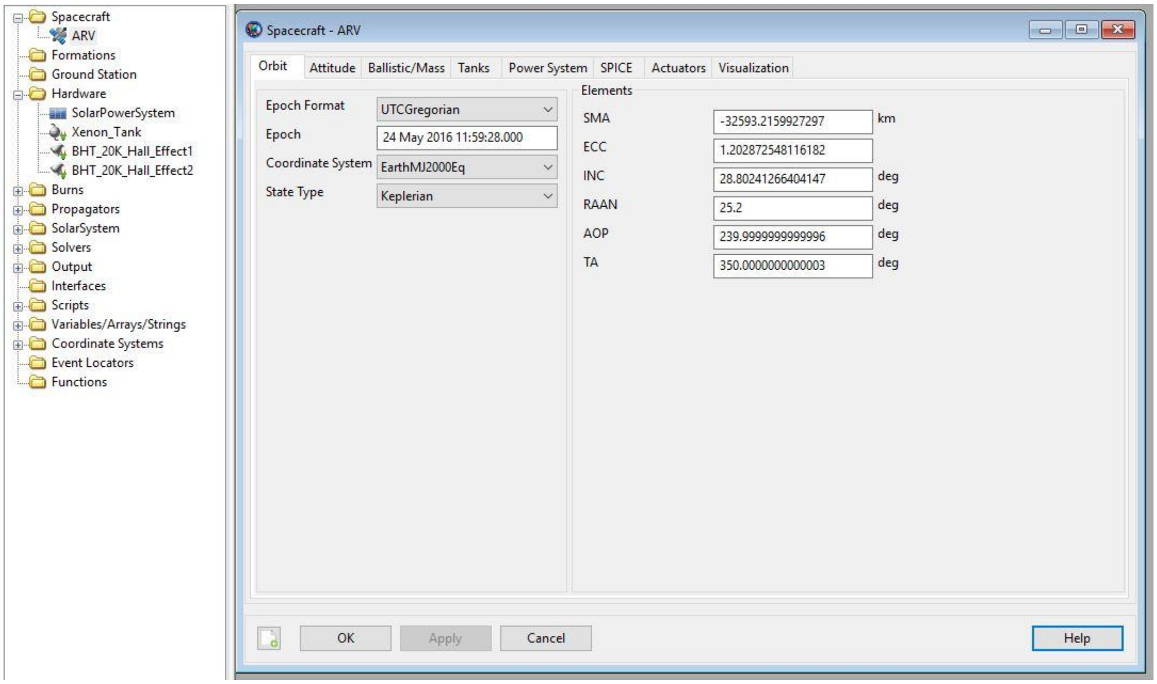


Figure 7.3 GMAT Spacecraft Customization Interface

The ARV is created through the Spacecraft and Hardware menus as shown with the configurations above. It is set to use the default EarthMJ2000Eq coordinate system as it starts off near Earth.

The Epoch value is the “starting” time for the spacecraft in the simulation. In this case, it is represented by the beginning of the mission once the spacecraft has reached LEO—24 May 2016. The orbital elements are set as follows (rounded to three decimal places):

- Semi-Major Axis: -32593.216 km
- Eccentricity: 1.203
- Inclination: 28.802°
- RAAN: 25.2°

- Argument of Periapsis: 240.0°
- True Anomaly: 350.0°

Next the ARV's dry mass is configured.

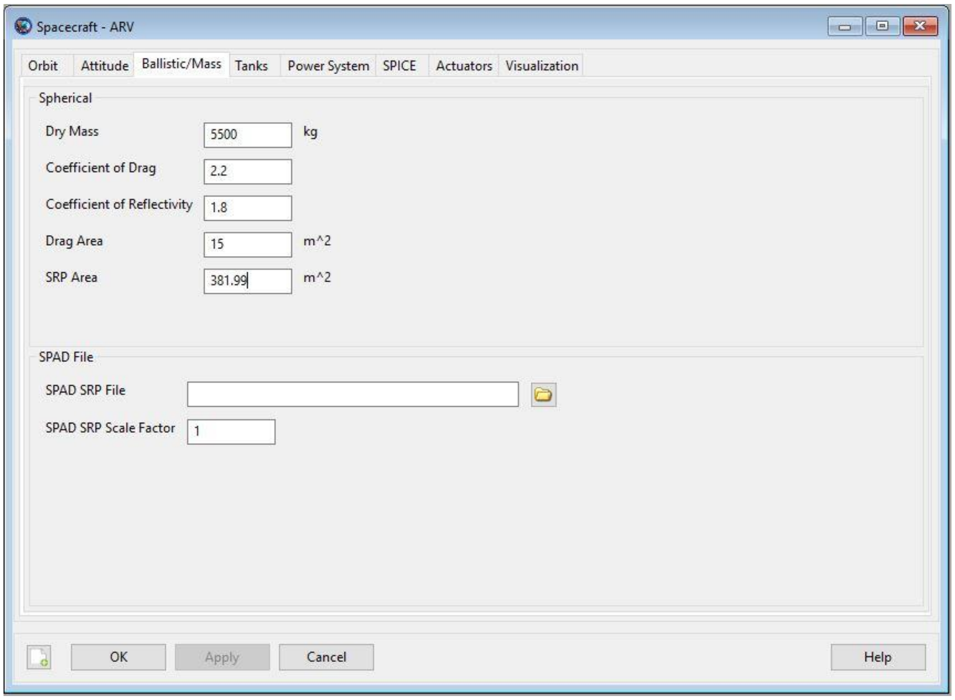


Figure 7.4 Spacecraft Ballistic/Mass Configuration

The ARV's dry mass is set as 5,500 kg. The solar radiation pressure (SRP) area is considered to be the total width of the spacecraft with the solar panels fully extended (35.7 meters) by the height of the spacecraft with the capture bag stowed (10.7 meters, the diameter of the solar panels when they're oriented perpendicularly to the body of the spacecraft) for a total of 389.99 square meters that can be exposed to SRP at most. The rest of these values are left to default settings. The hardware must be created and attributed next, beginning with the fuel tank, assumed to contain 9,500 kg of xenon propellant. [2]

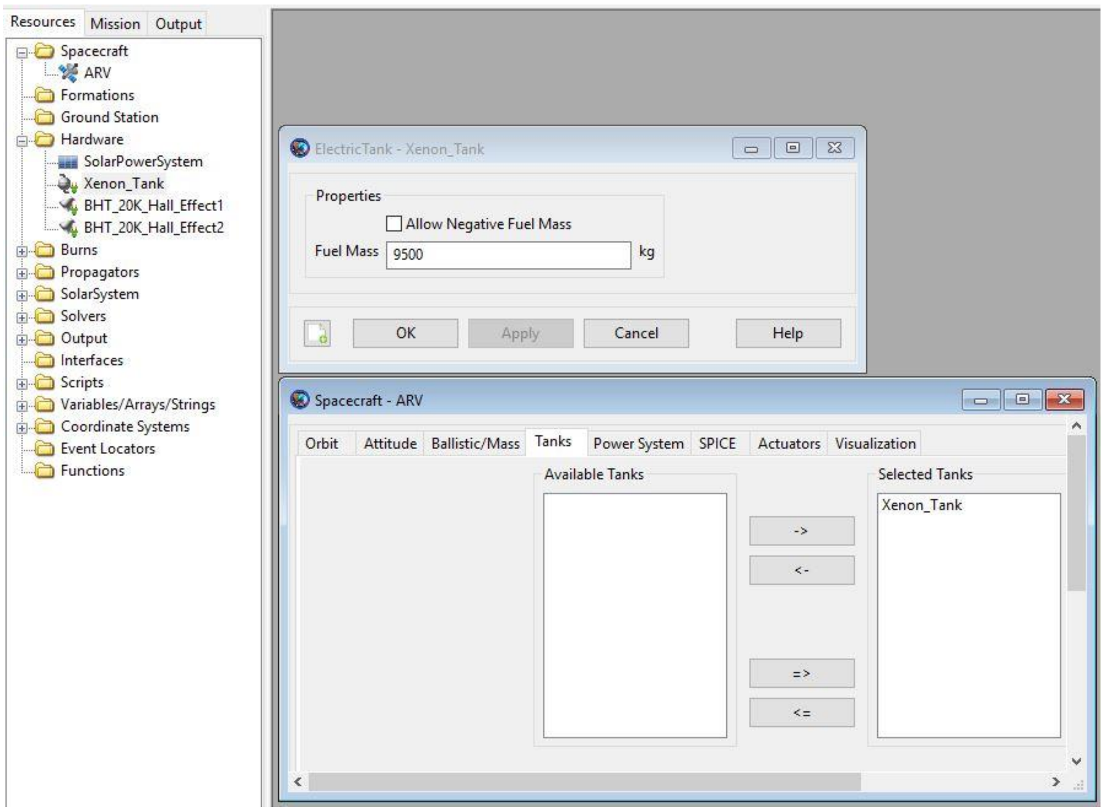


Figure 7.5 Electric Fuel Tank Configuration

The electric fuel tank is added by right-clicking on Hardware, hovering over Add, and selecting Electric Tank. 9,500 kg of propellant is attributed to the tank, assumed to be xenon. The fuel tank then becomes an available option in the Spacecraft configuration interface.

The spacecraft's power source must be configured next. By using the same method and right-clicking on Hardware, the Solar Power System is added.

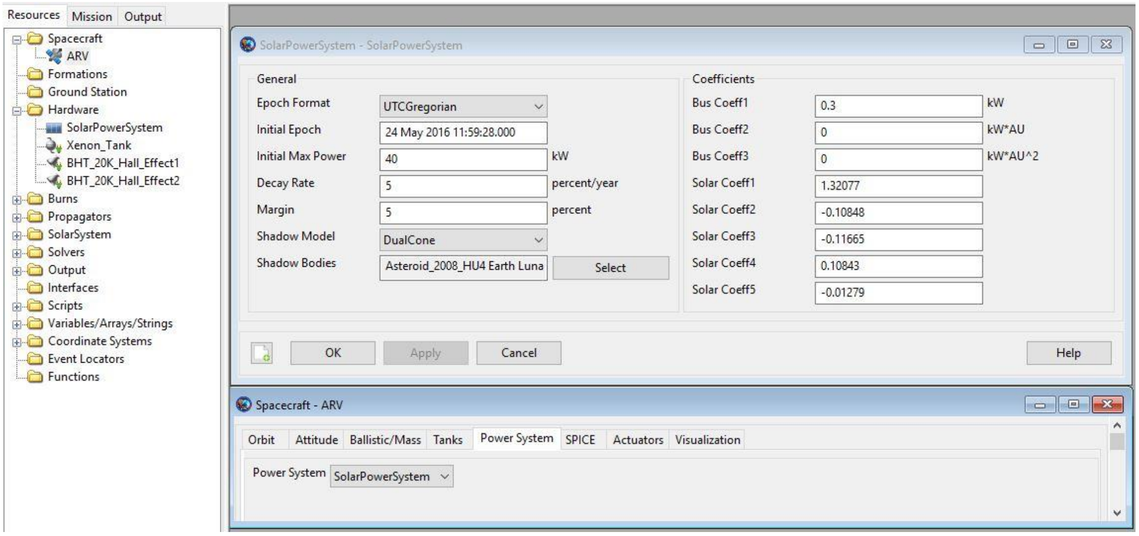


Figure 7.6 ARV Solar Power System Configuration

In this configuration dialog, the Solar Power System is set to have the same Epoch value as the whole spacecraft. Initial Max Power is set to 40 kW, as that is the amount required to power the thrusters. Earth, Luna, and 2008 HU4 are set as shadow bodies, capable of blocking light from the spacecraft. The rest are left at default settings. In the Spacecraft configuration interface, the Solar Power System is then selected as the ARV's power system.

Finally, the Busek Hall Effect Thrusters are added. In the Hardware drop-down menu, these are created by selecting the Electric Thruster option.

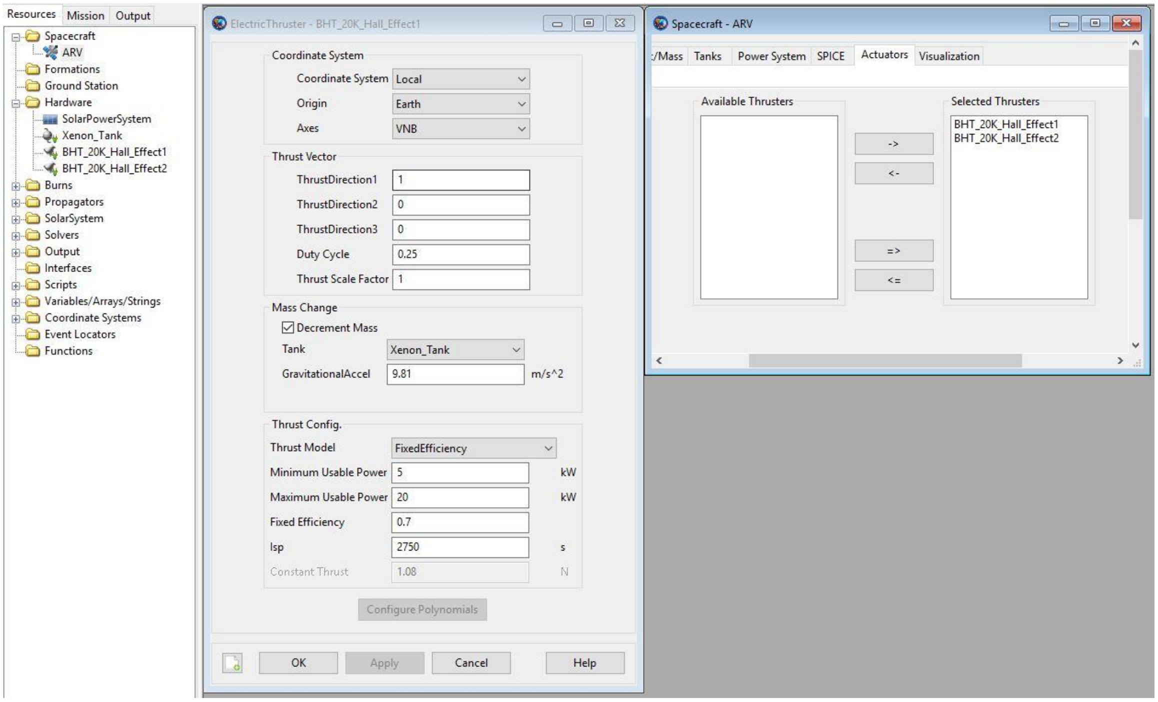


Figure 7.7 ARV Electric Thruster Configuration

Two identical electric thrusters are created to match the specifications of a BHT-20K thruster. Each thruster has 5 kW of minimum usable power, 20 kW of maximum usable power, a fixed efficiency of 70%, and a specific impulse of 2750 s. The axes are left as VNB, or Velocity-Normal-Binormal, a non-inertial coordinate system set on the spacecraft's motion with respect to its origin. [5] Both of these thrusters are added in the Spacecraft configuration interface as Actuators.

Under the Mass Change section, Decrement Mass is enabled for this study. This allows the program to model the amount of propellant being consumed throughout the mission sequence. This will provide a general basis for possible future optimizations to improve the propellant usage.

This particular thruster configuration uses a fixed efficiency model, meaning that the following polynomials are used to compute mass flow rate and thrust: [5]

$$\dot{m} = f_d * \frac{(2\eta P)}{(I_{sp} g_0)^2} \tag{7.1}$$

$$\bar{T} = f_d * f_s * \frac{(2\eta P)}{(I_{sp} g_0)} * \overline{R_{\iota T}} * \hat{T} \tag{7.2}$$

where

f_d = duty cycle

η = fixed efficiency

P = power provided to the thruster

I_{sp} = the specific impulse

g_0 = gravity

f_s = thrust scale factor

$\overline{R_{\iota T}}$ = rotation matrix from thrust coordinate system to inertial system

$\hat{T}$ = thrust unit vector

In this instance, the values plugged into the thruster configuration are:

$f_d = 0.25$

$\eta = 0.70$

$I_{sp} = 2750$ s

$g_0 = 9.81$ m/s^2

$f_s = 1$

The rotation matrix, thrust unit vector, and power are adaptively added as the simulation runs.

Next the ARV needs a way to use its two active thrusters, so a Finite Burn command is created by right-clicking on Burns. The Impulsive Burn option is generally used for preliminary mission designs and is not at all accurate for electric thrusters, so this option will be ignored entirely.

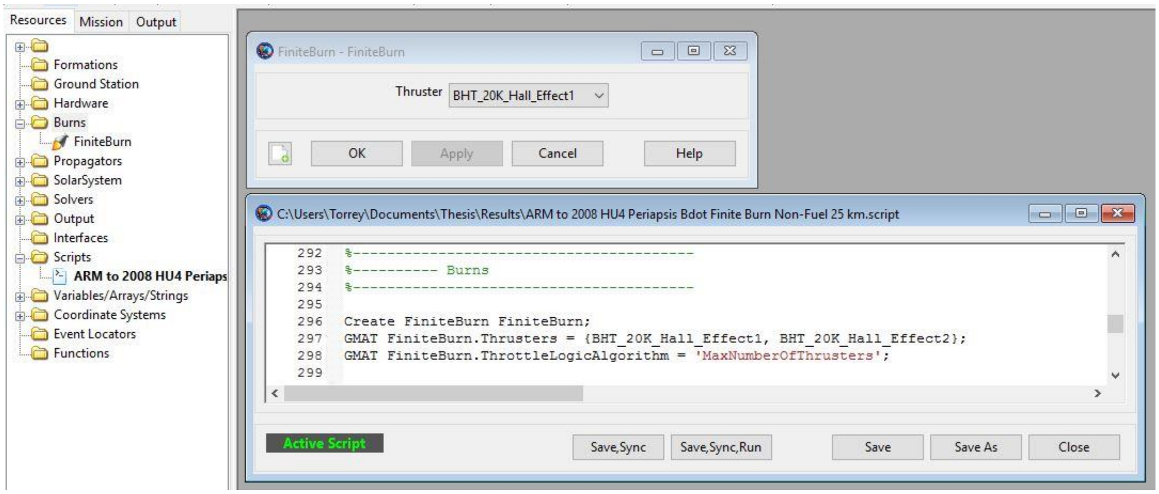


Figure 7.8 Finite Burn Configuration

Unfortunately the user interface only allows for the selection of one thruster at a time, so in order to attribute both thrusters to the same Finite Burn command, it must be done so manually in the mission script, by setting GMAT FiniteBurn.Thrusters = {BHT_20K_Hall_Effect1, BHT_20K_Hall_Effect2};.

Once the ARV has been fully configured with each hardware component attributed to it, the mission propagators are configured next. These propagators control how the orbital simulation is iterated using a variety of mathematical integration methods. The propagators will be configured to use the RungeKutta89 integrator type, which is described as an adaptive step, ninth order integrator with eighth order error control. [5]

This particular part of the orbital simulation will need three propagators: Near-Earth, Deep Space, and Near-Asteroid.

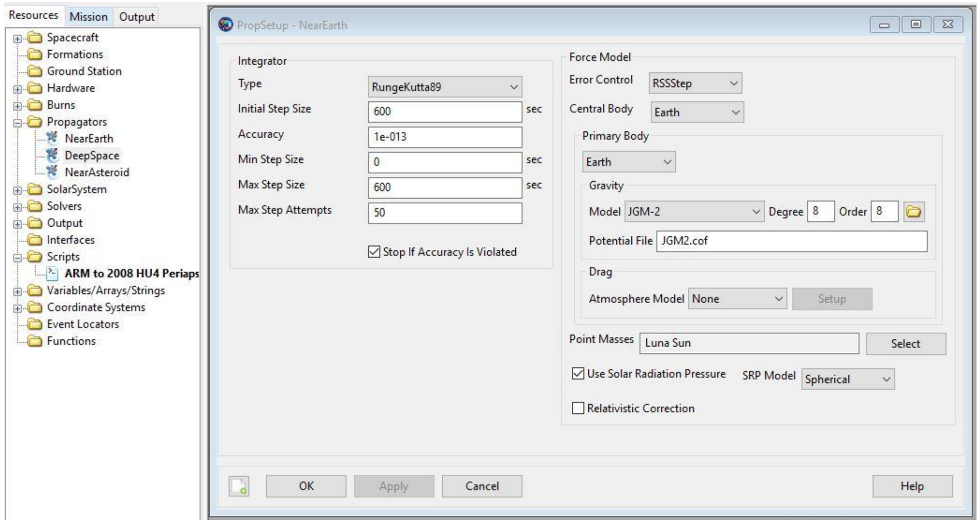


Figure 7.9 Near-Earth Propagator Configuration

The Near-Earth propagator is configured to have an initial and maximum step size of 600, a minimum step size of 0, and an accuracy of 1e-013. The central body is selected to be Earth, with one of the built-in gravity models. Point masses in this propagator include the Sun and Earth's moon Luna. Solar Radiation Pressure is enabled.

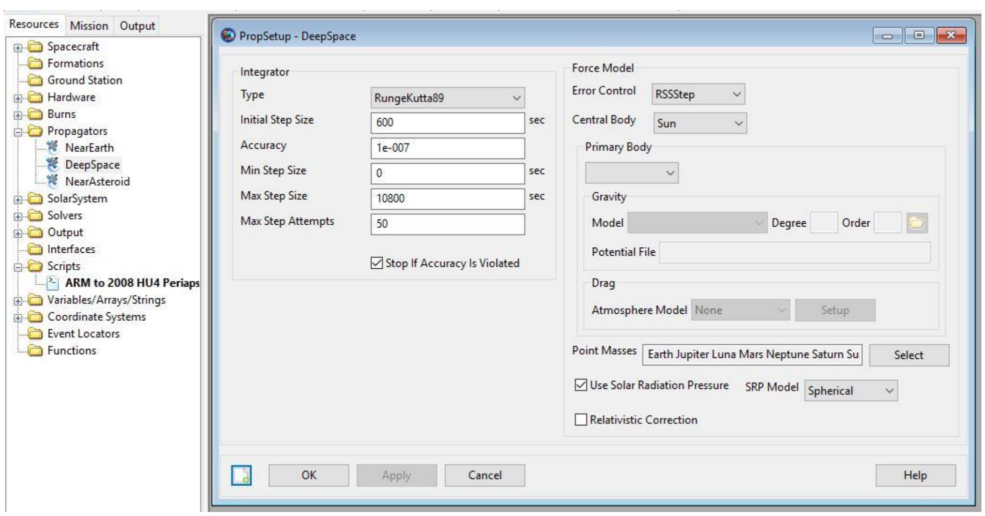


Figure 7.10 Deep Space Propagator Configuration

The Deep Space propagator is configured with an initial step size of 600, a maximum step size of 10800, and a minimum step size of 0. The central body is the sun, setting this as a heliocentric integrator with no primary body. The major celestial bodies chosen for point masses include the Sun, Venus, Earth, Luna, Mars, Jupiter, Saturn, Uranus, and Neptune. Mercury and Pluto are excluded as negligible point masses. As this propagator covers a much larger area of space, the accuracy is reduced to 1e-007 to save on computational time.

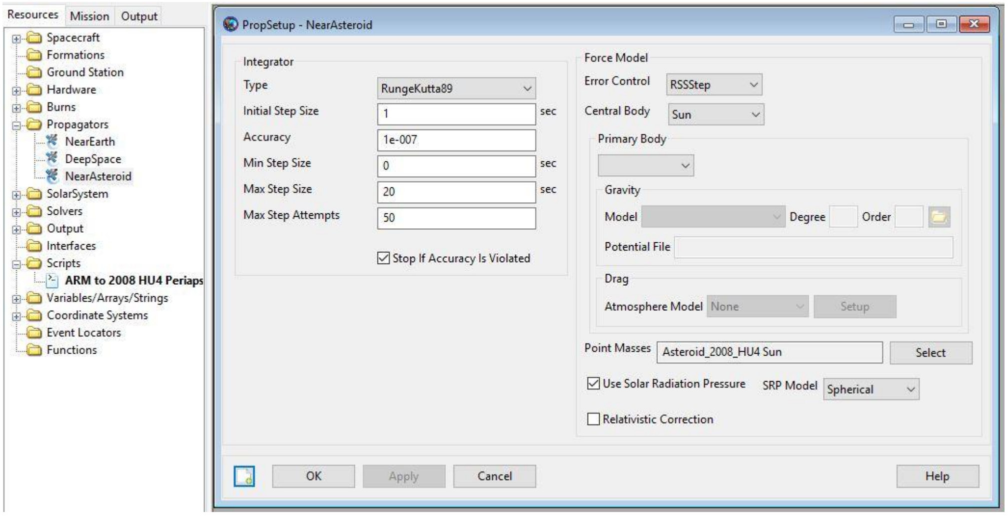


Figure 7.11 Near-Asteroid Propagator Configuration

The Near-Asteroid propagator is very similar to the Deep Space propagator, except in this instance, only the asteroid 2008 HU4 and the sun are considered for point masses, and the maximum step size is set to 20. At this stage in the trajectory, it becomes very easy to overshoot the target, so smaller step sizes are needed.

In addition to the propagators, the orbital simulation will also require a differential corrector to be used as a boundary value solver for the trajectory targeting sequence. The differential corrector is added by expanding Solvers, right-clicking on Boundary Value Solvers, and adding DefaultDC.

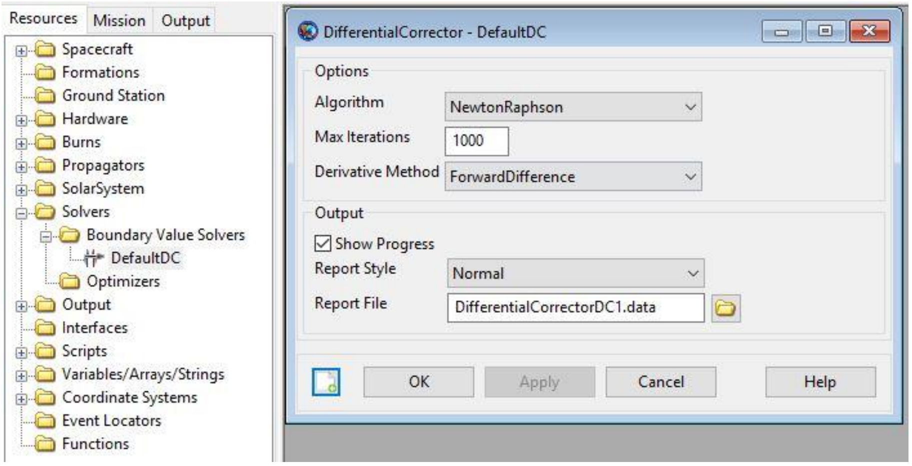


Figure 7.12 Differential Corrector Configuration

All settings are kept default, with the Newton-Raphson algorithm selected with forward difference as the derivative method. The number of maximum iterations is changed to 1000 to allow for longer run times as needed.

In order to visualize the orbital simulation and see how the ARV's trajectory is lining up with the target asteroid, new coordinate systems will be required for the different areas of space to correspond with the different propagators. These are added by right-clicking on Coordinate Systems.

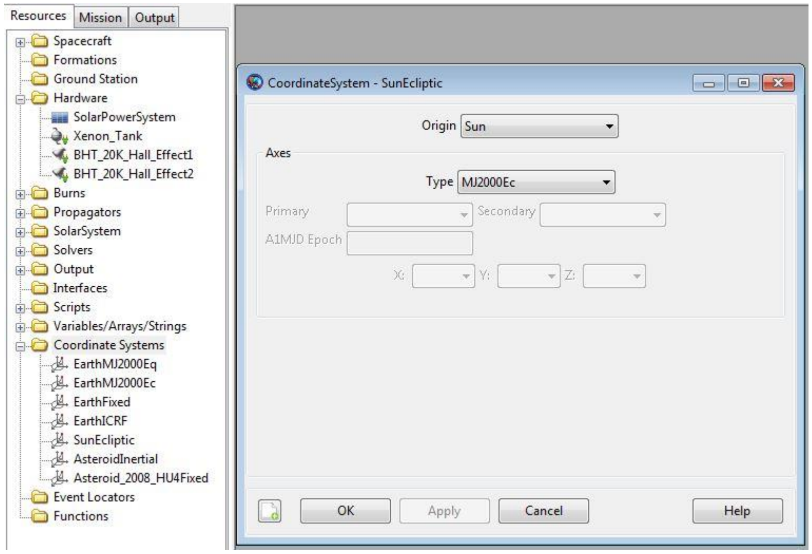


Figure 7.13 Sun Ecliptic Coordinate System

An ecliptic coordinate system centered on the sun (heliocentric) is created by selecting the sun as the origin and MJ2000Ec as the type. MJ2000Ec is an inertial coordinate system with the nominal x-axis pointing along the line formed by the intersection of the celestial body's mean equator and the mean ecliptic plane at the J2000 (January 2000) epoch. [5]

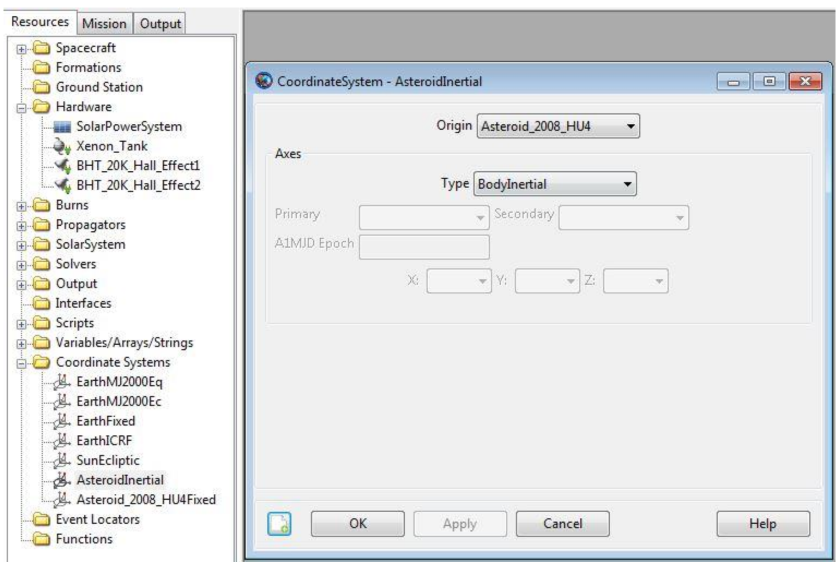


Figure 7.14 Asteroid Inertial Coordinate System

A body inertial coordinate system is created by selecting the asteroid 2008 HU4 as the origin with BodyInertial as the type. The BodyInertial coordinate type has the x-axis pointing along the line formed by the intersection of the body's equator and the mean equator at J2000. The z-axis aims along the body's spin axis direction at the J2000 epoch, and the y-axis completes the right-hand set.

Next the orbit views are configured. These are added via right-clicking Output. For the approach, three different orbit views can be utilized: an Earth view, a heliocentric Solar System view, and a close asteroid view of 2008 HU4.

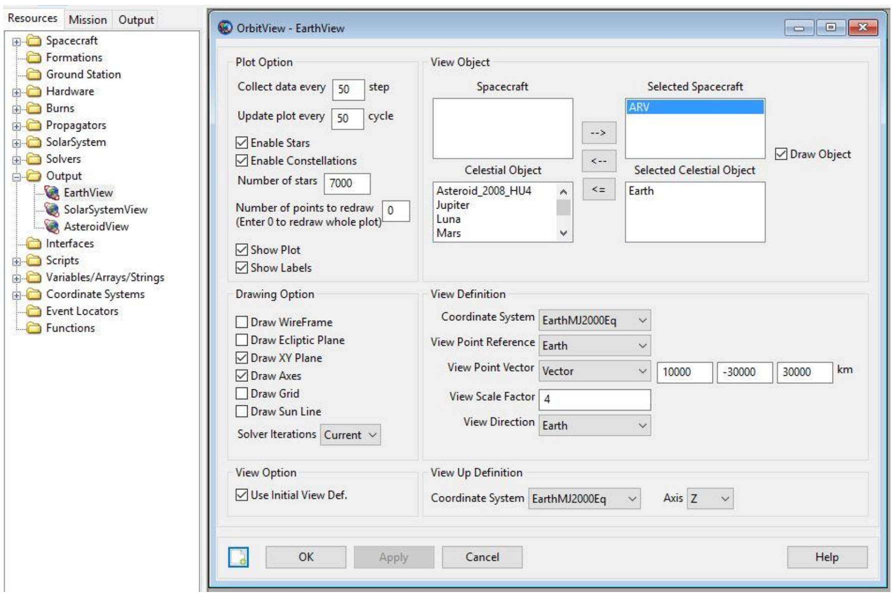


Figure 7.15 Earth Orbit View

For the Earth orbit view, only the ARV and Earth are set to be visible. Because so many steps are taken in the propagation, the number of data points must be limited by having data collected every 50 steps instead of in smaller increments in order to see a full plot of the spacecraft’s trajectory. The default EarthMJ2000Eq coordinate system is used for the view definitions.

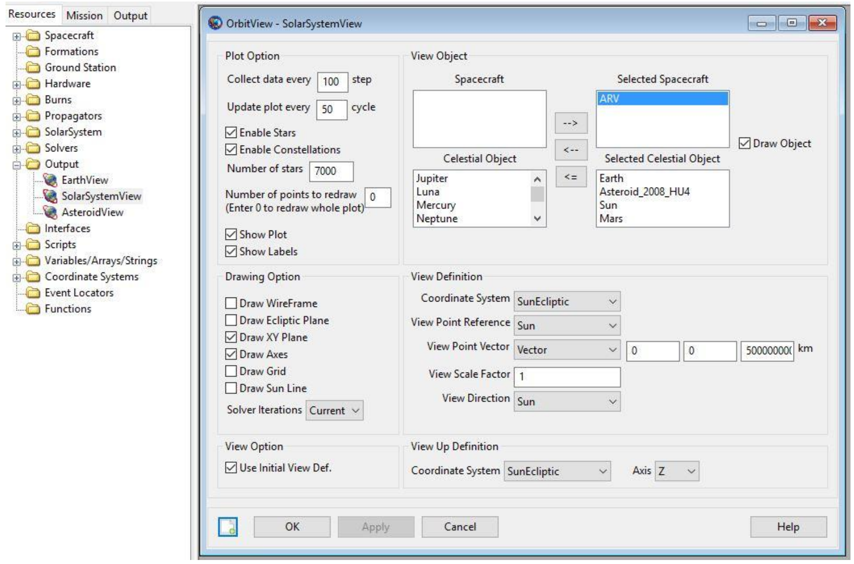


Figure 7.16 Heliocentric Solar System Orbit View

For the heliocentric Solar System orbit view, only the ARV, Earth, 2008 HU4, Mars, and the sun are selected as they will be the only celestial objects visible at the set distance. Data is collected every 100 steps, but because there are larger steps being taken in the Deep Space propagator, the plot will still appear relatively smooth. The SunEcliptic coordinate system established in a previous step is used for the view definitions.

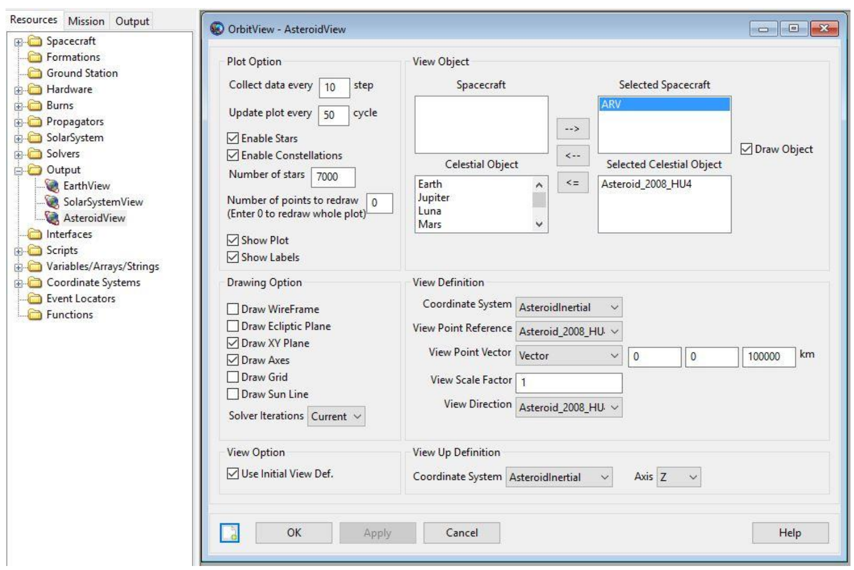


Figure 7.17 Asteroid Orbit View

The asteroid orbit view only needs to show the ARV as it approaches 2008 HU4. The Near-Asteroid propagator is taking much smaller steps than the other two propagators, so its plot will collect data in more frequent increments at every 10 steps in order to produce a smooth plot of the ARV's flight path. The AsteroidInertial coordinate system is used for the view definitions.

In order for the orbital simulation to properly target a trajectory using the finite burn command, a fourth variable will need to be introduced in addition to the three directions of the electric thrusters: a burn time variable.

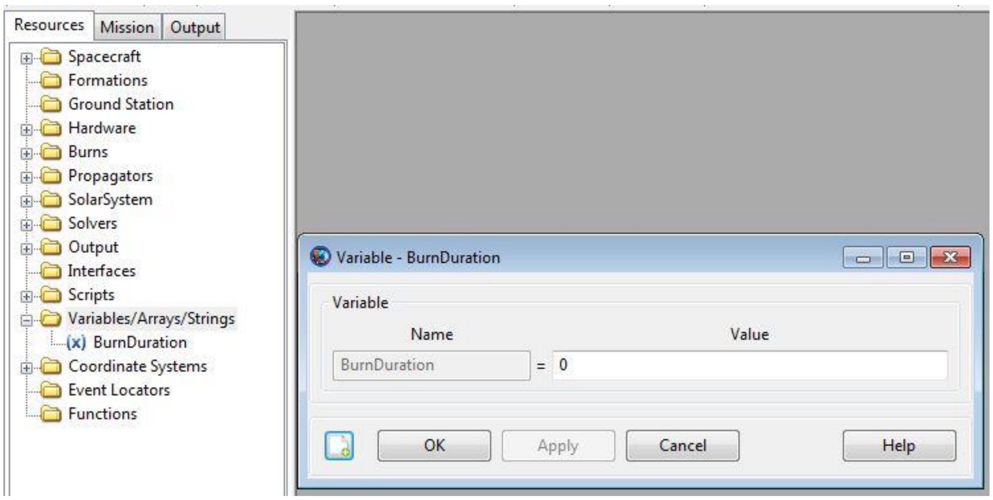


Figure 7.18 Burn Duration Variable

The Burn Duration variable is created by right-clicking Variables/Arrays/Strings and adding a new Variable. Since the variable must be created before it can be called in the mission sequence, the value is arbitrarily set to 0 and will be replaced by an initial guess with a Vary command during the mission sequence.

7.2 Mission Sequence

In GMAT, the entire mission sequence can be configured in one tab. The structure of this particular mission sequence is modeled after the Mars MAVEN tutorial's initial approach maneuver provided in the user guide, except instead of using impulsive burns, use of the more complex finite burn will be employed, and thus some significant changes are required.

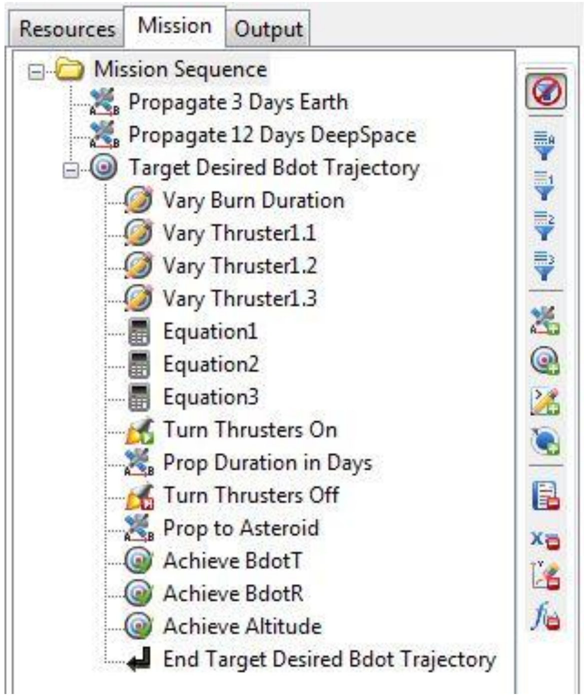


Figure 7.19 GMAT Mission Sequence Structure

This is the full mission sequence for the ARV's initial approach to 2008 HU4. First the spacecraft is set to propagate through near-Earth space, leading into the deep space propagator. The target command is then initiated, with what becomes a four-variable problem. Propagation through deep space is controlled by the burn duration variable as the thrusters are activated. The ARV then propagates toward the asteroid to achieve specific parameters.

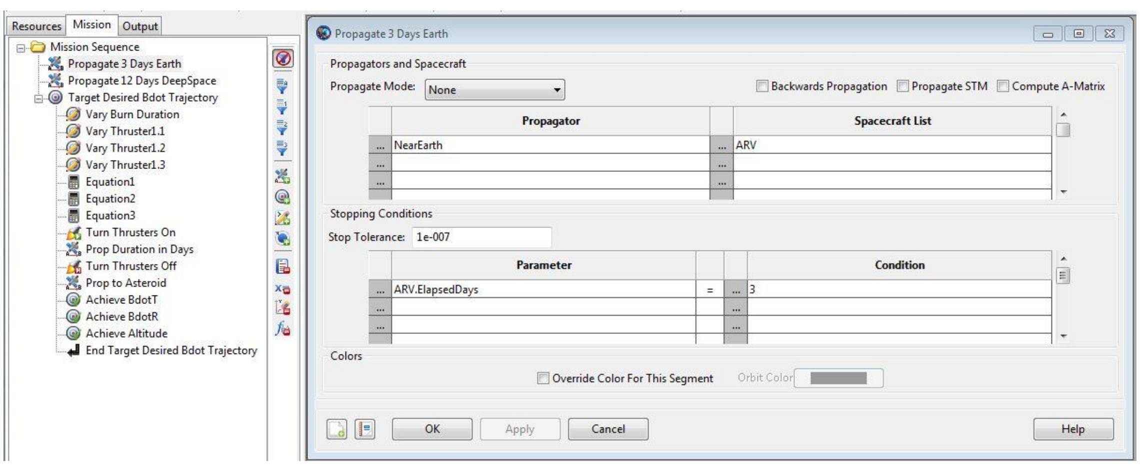


Figure 7.20 Propagate Near Earth for 3 Days

The first stage of this mission sequence is to have the spacecraft propagate near Earth based on its initial orbital parameters. The Propagate command is added by right-clicking on Mission Sequence, hovering over Append, and selecting Propagate. It is then configured to utilize the NearEarth propagator with the ARV spacecraft with a set parameter condition of 3 elapsed days.

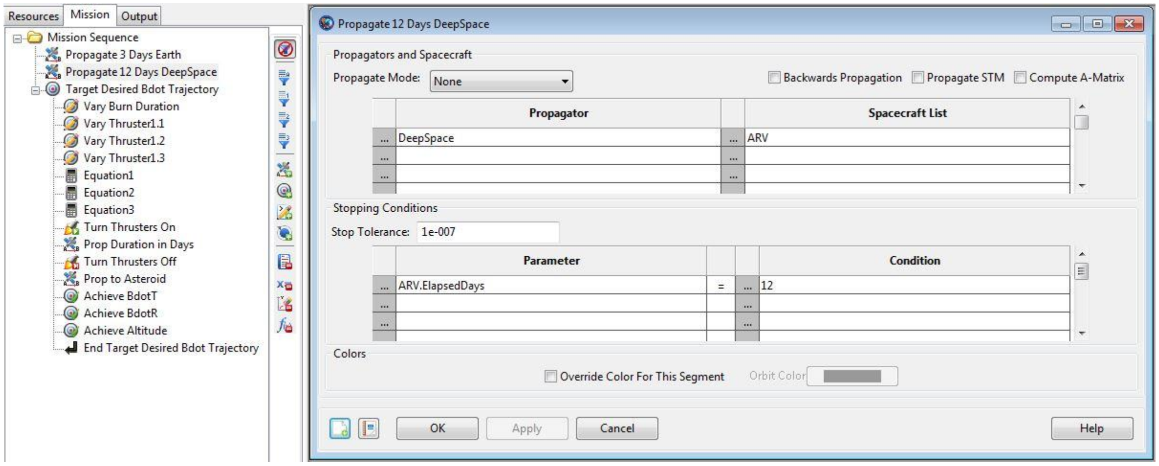


Figure 7.21 Propagate in Deep Space for 12 Days

This propagate command is configured almost exactly like the previous one in Figure 7.20, except in this case the DeepSpace propagator is used, and the parameter condition is set as 12 elapsed days.

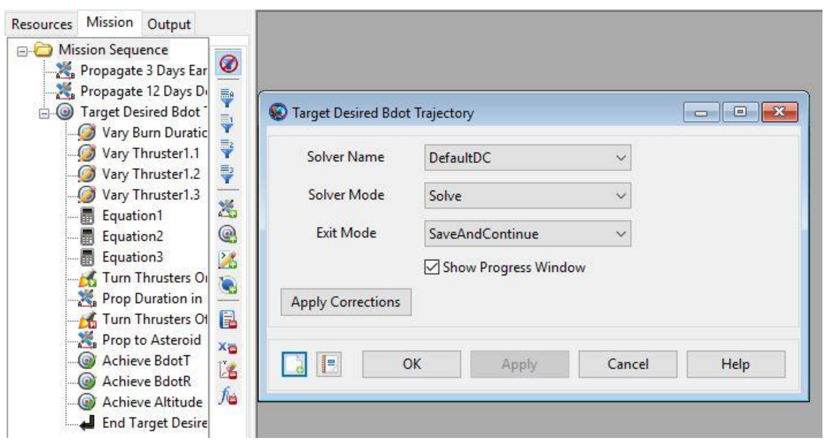


Figure 7.22 Target Trajectory Sequence

The part of the mission sequence needed to compute the trajectory to the asteroid is added by selecting the Target option in the Append menu after right-clicking on Mission Sequence. The default differential corrector will be used, and the Exit Mode is set to SaveAndContinue in order to see the progress of each iteration.

For the next four parameters of the target trajectory sequence, Vary commands are introduced. These have a Variable field that determines what is being varied, an Initial Value field that functions as an initial guess, a Perturbation field that controls the amount of change varied through each iteration, fields for Upper and Lower limits for the range of variation, and a Max Step to set just how much the variable can be changed in one iteration. The Additive Scale Factor and Multiplicative Scale Factor fields will be left at their default values.

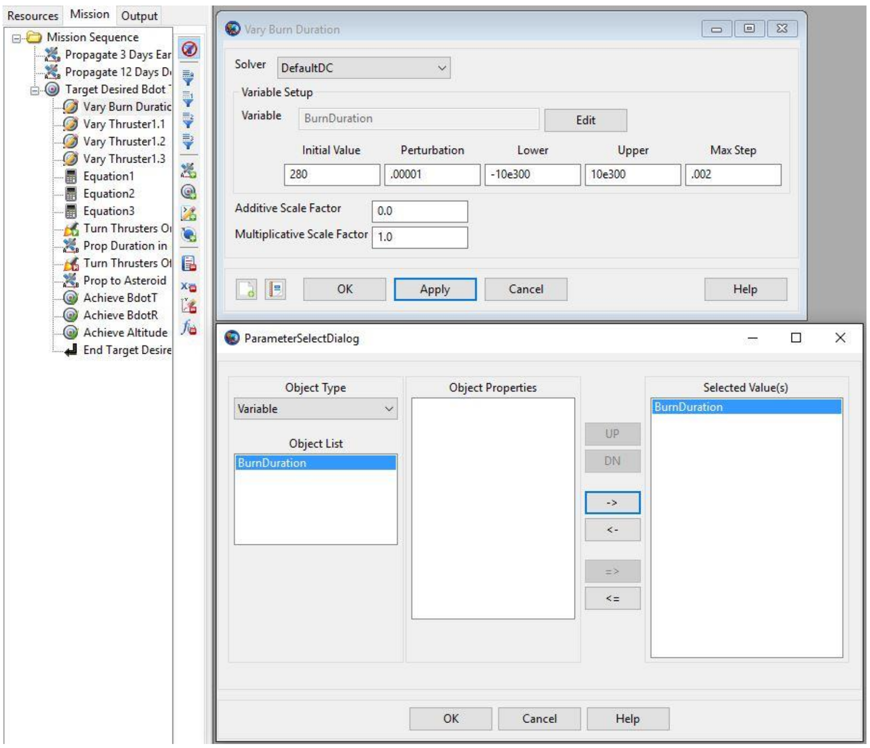


Figure 7.23 Varying Burn Duration

A Vary command is added for the finite burn duration. The BurnDuration variable created earlier is selected. The Initial Value will be set as the number of days for the finite burn. In this case, the initial guess is 280 days. Perturbation is set to be 0.00001, with -10e300 Lower and 10e300 Upper, and a Max Step of 0.002.

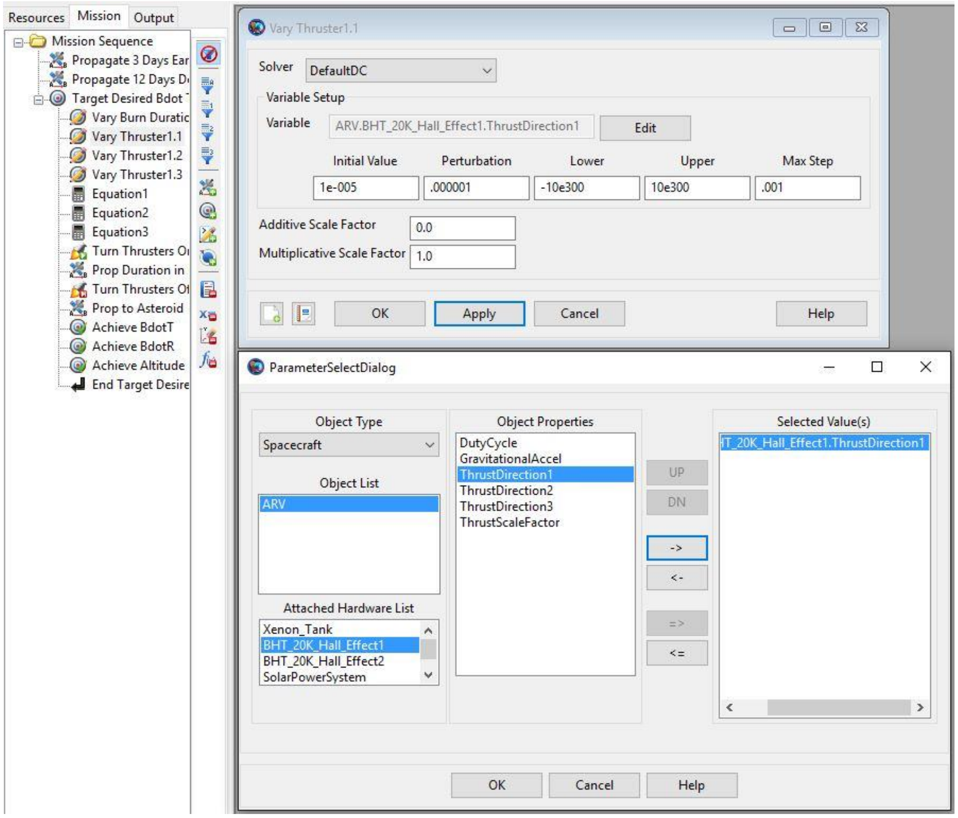


Figure 7.24 Varying the First Thruster Direction

The next three Vary commands will be for thrust directions 1, 2, and 3, which represent the coordinate system chosen for the thrusters on the spacecraft, in this case VNB. Each Vary command is set for BHT_20K_Hall_Effect1, with an Initial Value of 1e-005, Perturbation set at 0.00001, -10e300 for Lower and 10e300 for Upper, and a Max Step of 0.001.

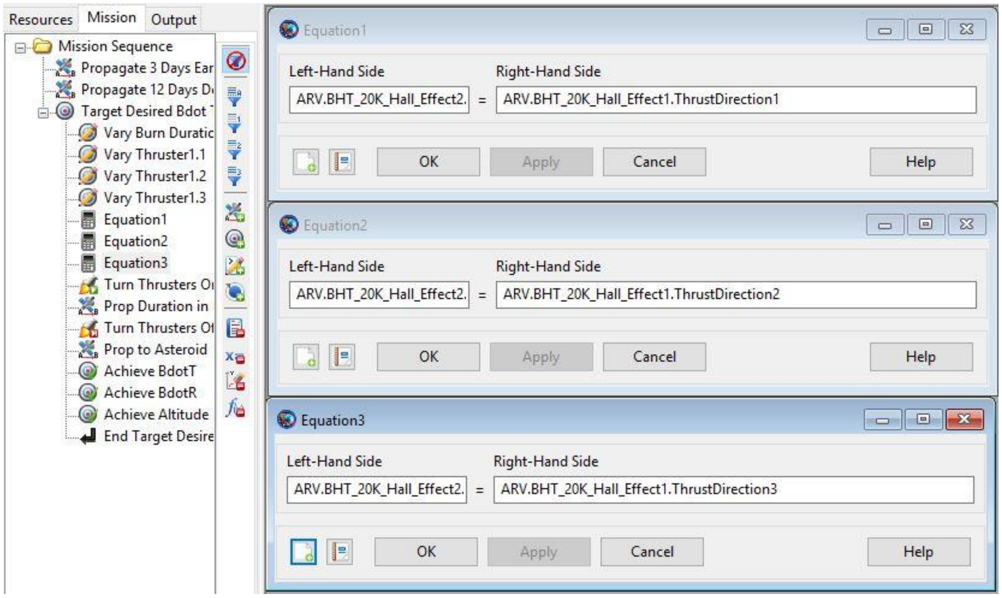


Figure 7.25 Vary the Second Thruster Direction

The second HET is assumed to be oriented in the same position and direction as the first HET, as multiple electric thrusters can be bundled and treated effectively as one unit. [10] This also simplifies the calculation run time by reducing the number of active variables from seven (six separate thrust directions and one burn time) to four (three thrust directions shared between two thrusters and one burn time.) This is implemented by adding an Equation element to the target sequence and setting each thrust direction for the second thruster to equal the corresponding thrust direction of the first thruster. For instance,
ARV.BHT_20K_Hall_Effect2.ThrustDirection1 =
ARV.BHT_20K_Hall_Effect1.ThrustDirection1, and so on.

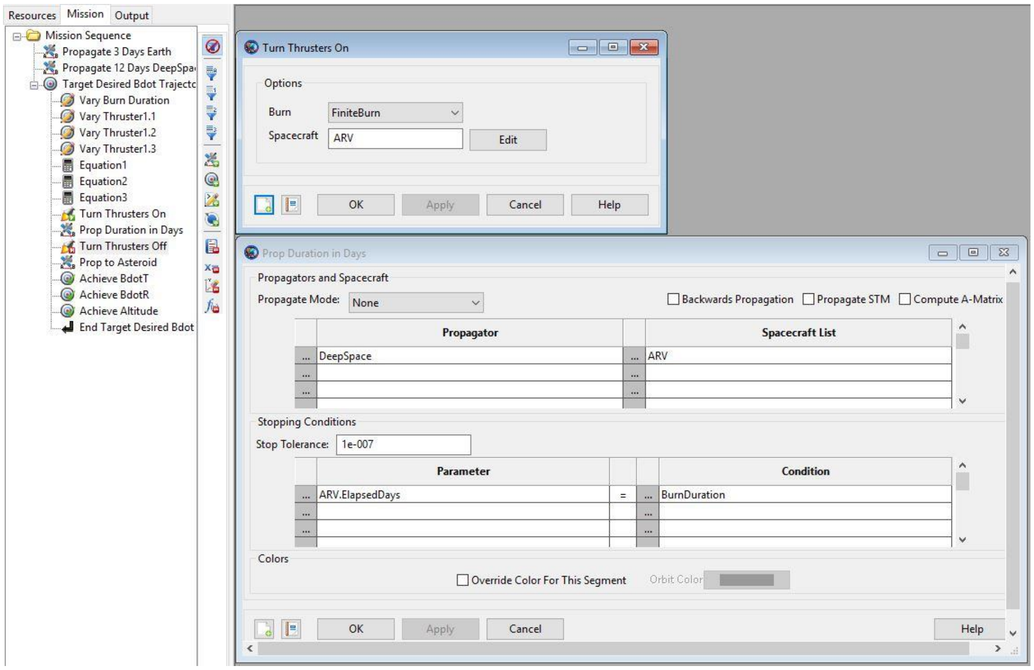


Figure 7.26 Turn Thrusters On

The next step in the mission sequence is to ignite the thrusters for the deep space burn toward Asteroid 2008 HU4. A BeginFiniteBurn command is added, followed by another Propagate command. This time the propagate command, set to use the DeepSpace propagator, will utilize the BurnDuration variable as the parameter condition for elapsed days. Whatever value is contained within the BurnDuration variable will be inserted as the ElapsedDays value.

The spacecraft is assumed to have a steady finite burn toward the asteroid following the trajectory calculated by the Vary commands. The thrusters are adjusted to point the ARV on the correct path to reach 2008 HU4 on the time of the burn value given. Once that burn time has been reached, the thrusters are then deactivated, concluding the finite burn.

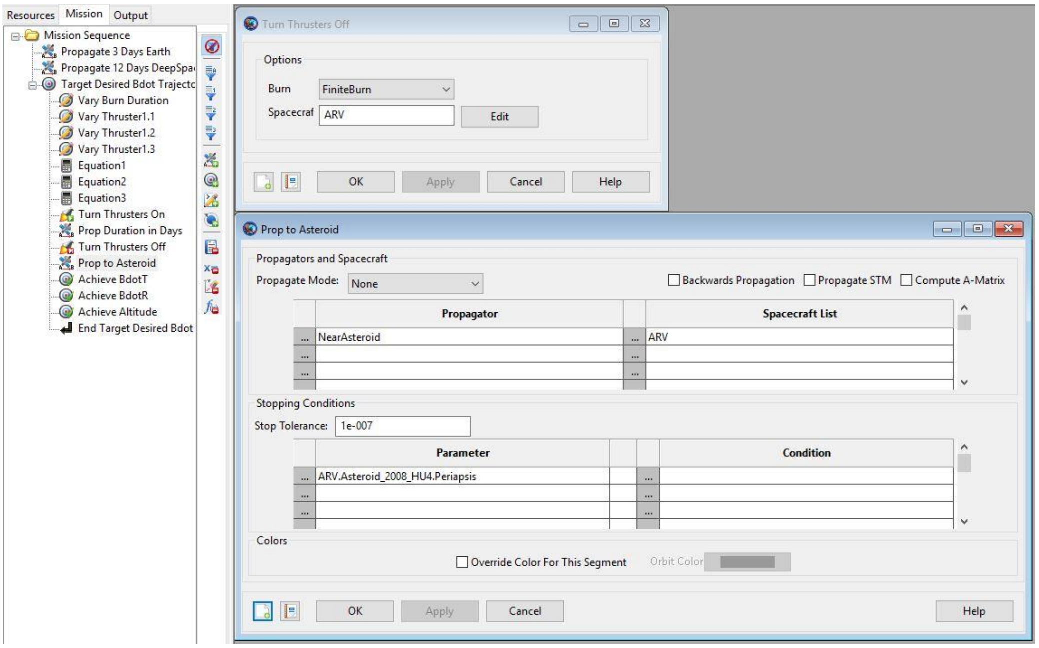


Figure 7.27 Turn Thrusters Off

The EndFiniteBurn command is added to conclude the finite burn through deep space, followed by another Propagate command. This propagate command utilizes the NearAsteroid propagator, and the Parameter is set as the asteroid's periapsis, or closest point of orbit.

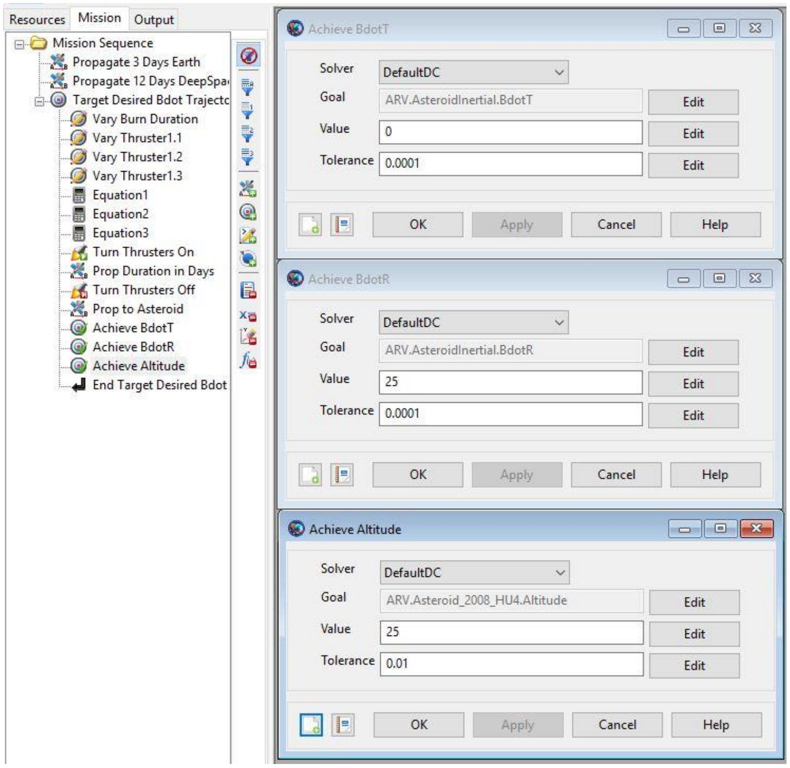


Figure 7.28 Mission Sequence Achieve Parameters

To conclude the structure of the mission sequence, Achieve parameters are required. This is where the B-Plane is targeted as a goal for the Vary commands to reach. As shown in Figure 2.3, there are two elements of the B-Plane: BdotT and BdotR. The goal is to reach a BdotT value of 0 km and a BdotR value of 25 km. This will put the spacecraft in position for a polar orbital approach at an altitude of approximately 25 km.

The final component of the Target command is the End Target command, which is added automatically.

7.3 Running the Mission Sequence

Running the mission sequence will take a considerable amount of time to process. In order to expedite the process, the propagators can be manually adjusted to take larger, less accurate steps, until no further progress is made, then adjusted again to take slightly smaller, more accurate steps, and so on, until convergence is met.

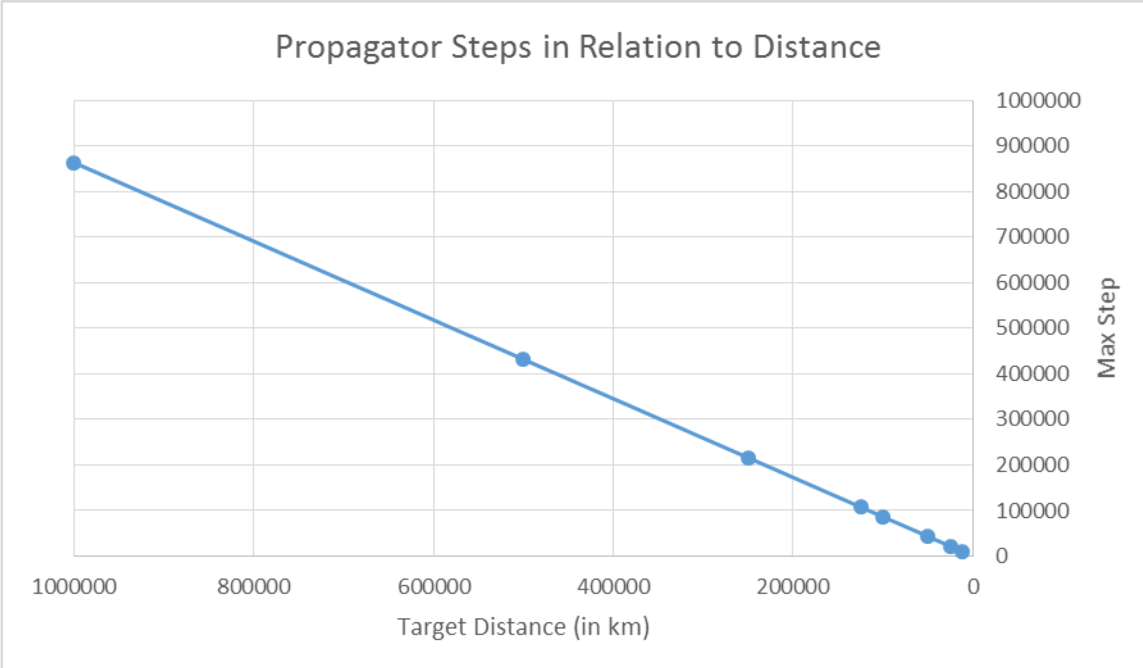


Figure 7.29 Relation of Propagator Max Step Size to Target Distance

There is a nearly linear relation between the chosen Max Step value of the propagator and the distance to convergence on the trajectory target seen in this particular simulation. This is an example based on manually adjusting the DeepSpace propagator.

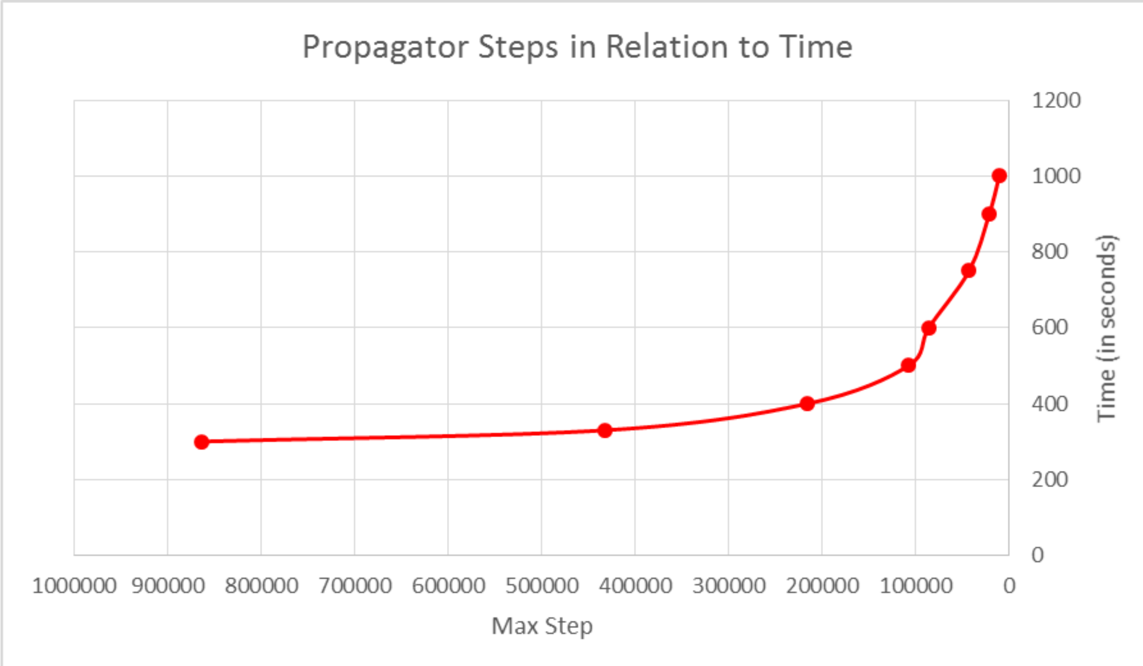


Figure 7.30 Relation of Propagator Max Step Size to Closest Convergence Time

The computational time required to reach the closest point of convergence given the maximum step size experiences a nearly logarithmic increase as the propagator's Max Step value is decreased. The time values are an approximation and will vary depending on the machine running the orbital simulation. This is an example based on the DeepSpace propagator.

The plots in Figures 7.29 and 7.30 are not representative of actual data; they serve as examples of what one particular iteration cycle may present when manually adjusting the propagators. In this case, "closest convergence" is defined to be the point at which no further progress is made in the iterations due to the step size. Once this state is reached, the simulation is halted, corrections are applied to the initial guesses in each Vary Command, and the Max Step Size value is decreased before the simulation is run again. This process is repeated until actual convergence is met.

7.4 ARV Approach Results

The results for the ARV's approach to 2008 HU4 are as follows, rounded to the nearest thousandths decimal place.

Table 7.1 ARV Approach to 2008 HU4

Burn Duration	277.770 Days
Thruster V-Direction	-0.002 km/s
Thruster N-Direction	-0.405 km/s
Thruster B-Direction	0.026 km/s

As a reminder, the VNB (Velocity-Normal-Binormal) coordinate system is set in relation to the spacecraft's initial orientation.

In order to determine the ΔV requirement for this maneuver, each of those thruster direction values must be treated as components of a three-dimensional vector to find its magnitude.

$$|\Delta V| = \sqrt{\Delta V_x^2 + \Delta V_y^2 + \Delta V_z^2} = \sqrt{(-0.002)^2 + (-0.405)^2 + (0.026)^2} \quad (7.3)$$

$$= 0.4056574909945581 \text{ km/s}$$

Therefore ARV approach ΔV = 0.406 km/s, rounded to the nearest thousandths place. With this ΔV, the spacecraft arrives within 25 km of 2008 HU4 on 28 June 2017, having used up 384.867 kg of xenon propellant.

Table 7.2 Initial Approach Results

Time to Reach 2008 HU4	400 Days or 1.10 Years
ΔV Required	0.406 km/s
Propellant Remaining	9115.133 kg

This represents the initial approach to within 25 km/s for the pre-capture evaluation of the asteroid. The remaining propellant is shown in the Target Command Summary. Refer to Appendix A.1 for more information.

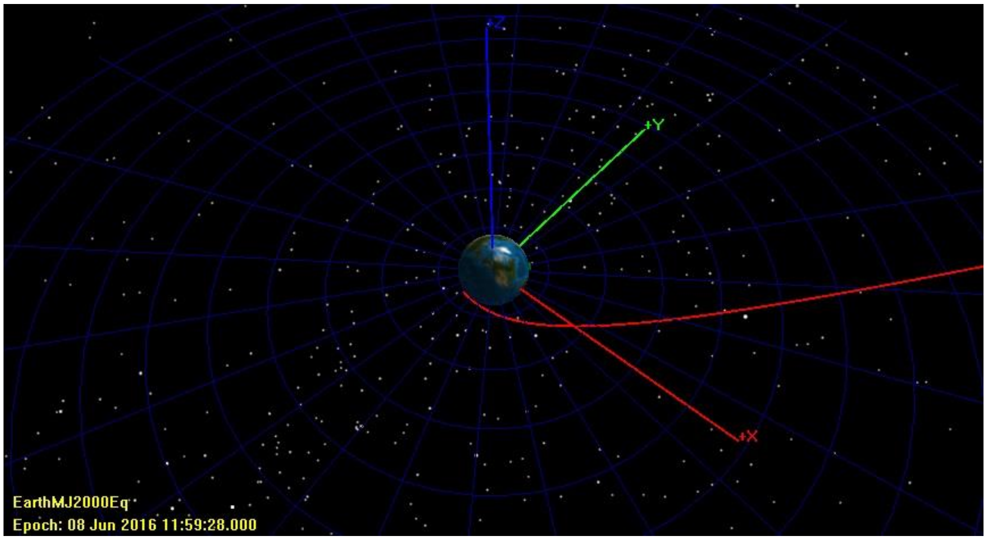


Figure 7.31 ARV Launch from Earth

This is an orbital view of the ARV leaving LEO on approach to the target asteroid 2008 HU4.

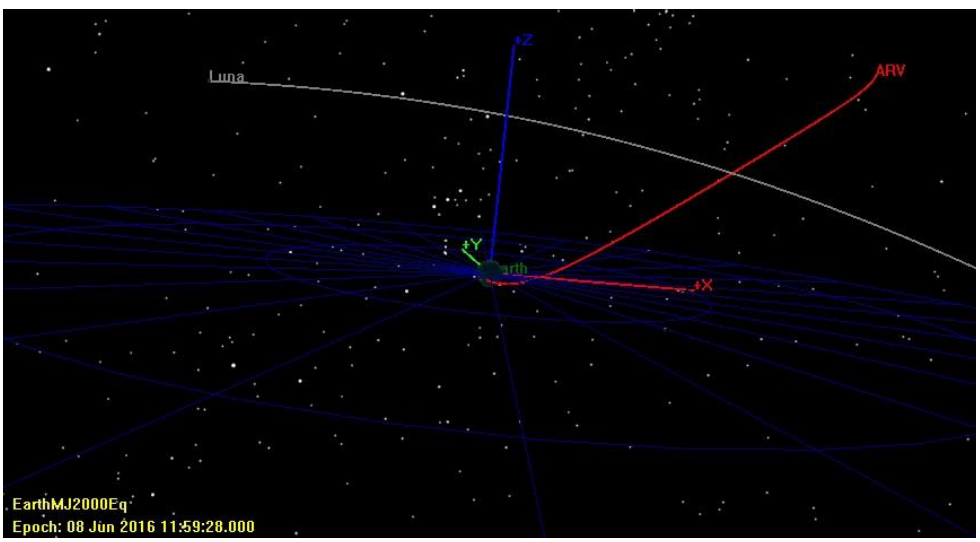


Figure 7.32 ARV passing Luna, Earth’s Moon

An alternate orbital view of the ARV is shown passing the orbit of Earth’s moon Luna.

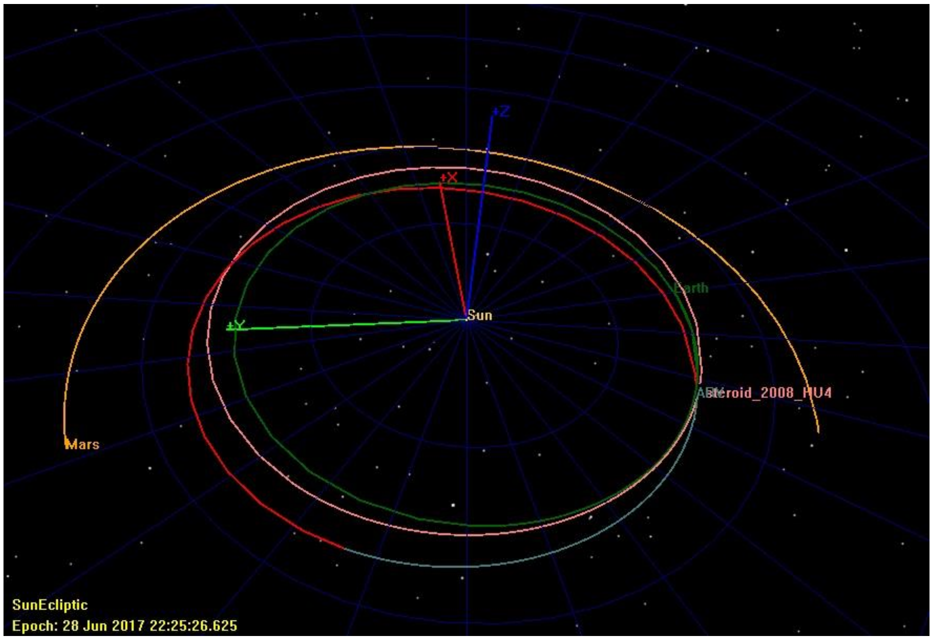


Figure 7.33 Heliocentric View of ARV Arriving at 2008 HU4

The heliocentric Solar System view shows the positions of the sun, Earth, Mars, and the ARV converging on the asteroid 2008 HU4's position on 28 June 2017.

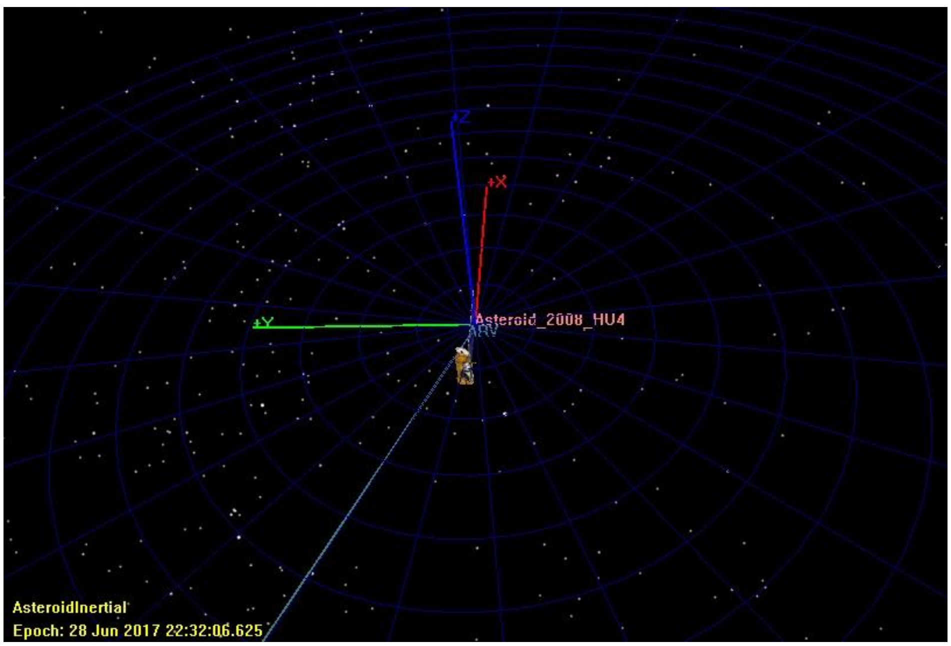
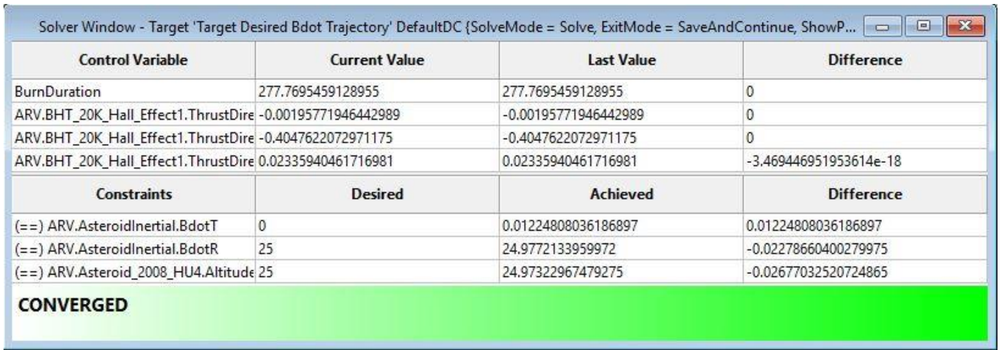


Figure 7.34 Asteroid-Centric View of ARV Arriving

This shows the direct approach of the ARV approaching within 25 km of 2008 HU4. Note that the 3D model used for the spacecraft in this image is not an accurate representation of the ARV, nor is it to scale. Please see Figures 5.1, 5.2, and 5.3 for a more accurate portrayal of the spacecraft.



Solver Window - Target 'Target Desired Bdot Trajectory' DefaultDC {SolveMode = Solve, ExitMode = SaveAndContinue, ShowP...

Control Variable	Current Value	Last Value	Difference
BurnDuration	277.7695459128955	277.7695459128955	0
ARV.BHT_20K_Hall_Effect1.ThrustDire	-0.00195771946442989	-0.00195771946442989	0
ARV.BHT_20K_Hall_Effect1.ThrustDire	-0.4047622072971175	-0.4047622072971175	0
ARV.BHT_20K_Hall_Effect1.ThrustDire	0.02335940461716981	0.02335940461716981	-3.469446951953614e-18

Constraints	Desired	Achieved	Difference
(==) ARV.AsteroidInertial.BdotT	0	0.01224808036186897	0.01224808036186897
(==) ARV.AsteroidInertial.BdotR	25	24.9772133959972	-0.02278660400279975
(==) ARV.Asteroid_2008_HU4.Altitude	25	24.97322967479275	-0.02677032520724865

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Figure 7.35 GMAT Solver Window for ARV Approach

In addition to the orbital views, this solver window is presented as an output of the variables and constraints. As indicated, the spacecraft successfully converges on the correct B-Plane trajectory and altitude from the asteroid.

CHAPTER VIII

RETURN TRAJECTORY DESIGN

8.1 Capture Maneuver and Drift Time

While the capture maneuver itself is not directly simulated in this study, prior research shows that the Asteroid Redirection Mission will require approximately 90 days to characterize the asteroid, rendezvous with it, then de-tumble it. [1] An additional two months of time is allotted for further evaluations of the asteroid and mission preparation time for the return maneuver. This sets the beginning of the ARV's return mission as 08 December 2017.

8.2 Resources

Most of the resources from the ARV approach script will be retained for the return mission.

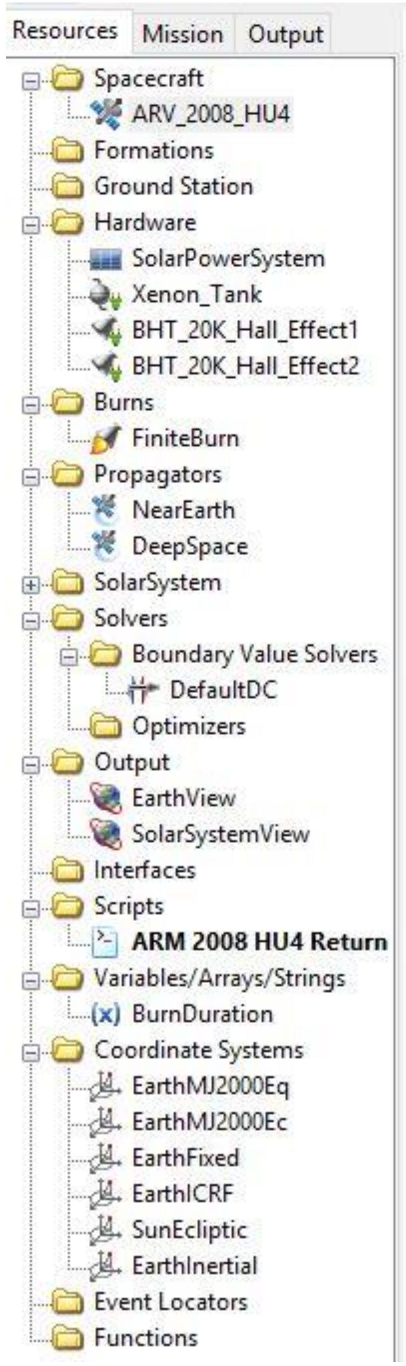

Figure 8.1 ARM Asteroid Return GMAT Resources

As can be seen in Figure 8.1, most of the resources used in the return portion of the Asteroid Redirect Mission remain the same.

One main difference in the resources structure for the return trip is that the elements related directly to the asteroid are removed as they are no longer needed. The NearAsteroid propagator is omitted as well as the AsteroidView output and the AsteroidInertial coordinate system. Asteroid 2008 HU4 is also omitted from the SolarSystem structure. The ARV now accounts for both the spacecraft and the asteroid, thus it has been renamed as “ARV_2008_HU4” to indicate that it includes both.

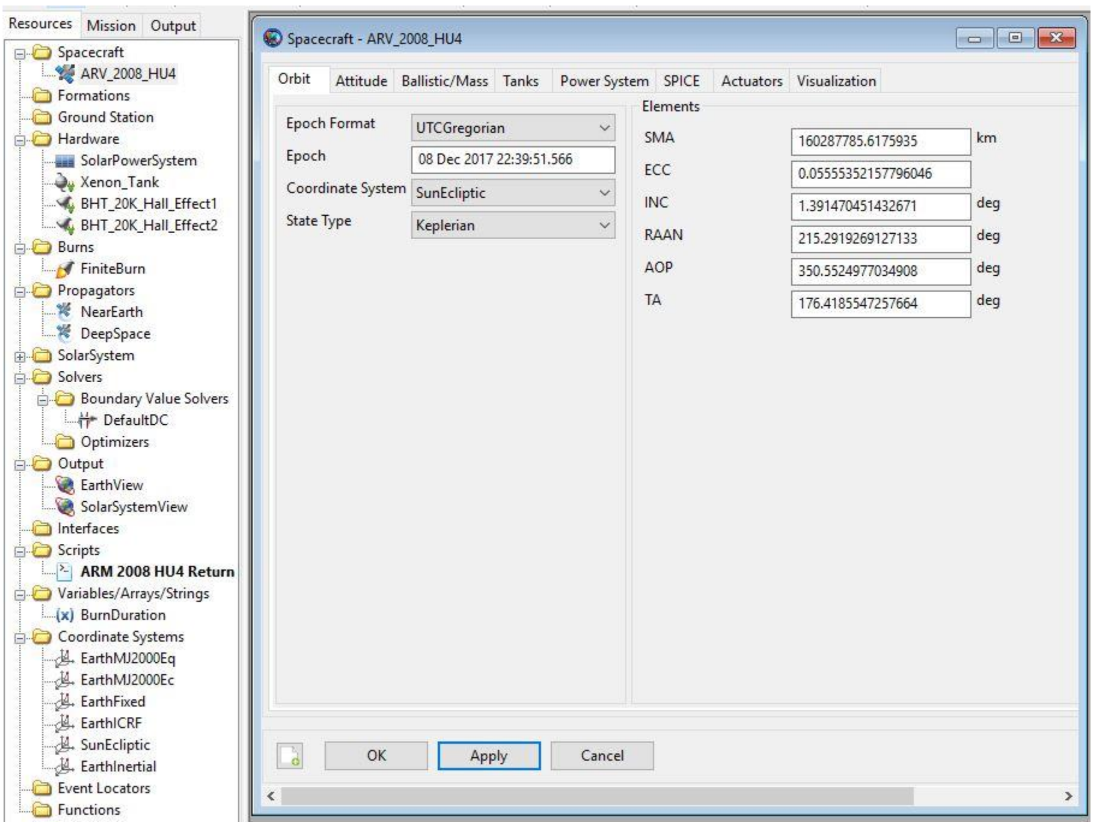

Figure 8.2 ARV Return Orbital Configuration

The ARV now has a starting Epoch of 08 December 2017 to correlate with the amount time spent capturing, de-tumbling, and preparing the asteroid for the return maneuver. Cartesian elements are extracted from the asteroid’s known position at the time of 08 December 2017 and then converted to a Keplerian state and used as the elements for the spacecraft on the assumption that both the spacecraft and the asteroid will be in the same position once 2008 HU4 is captured. These numbers are shown in relation to the SunEcliptic coordinate system.

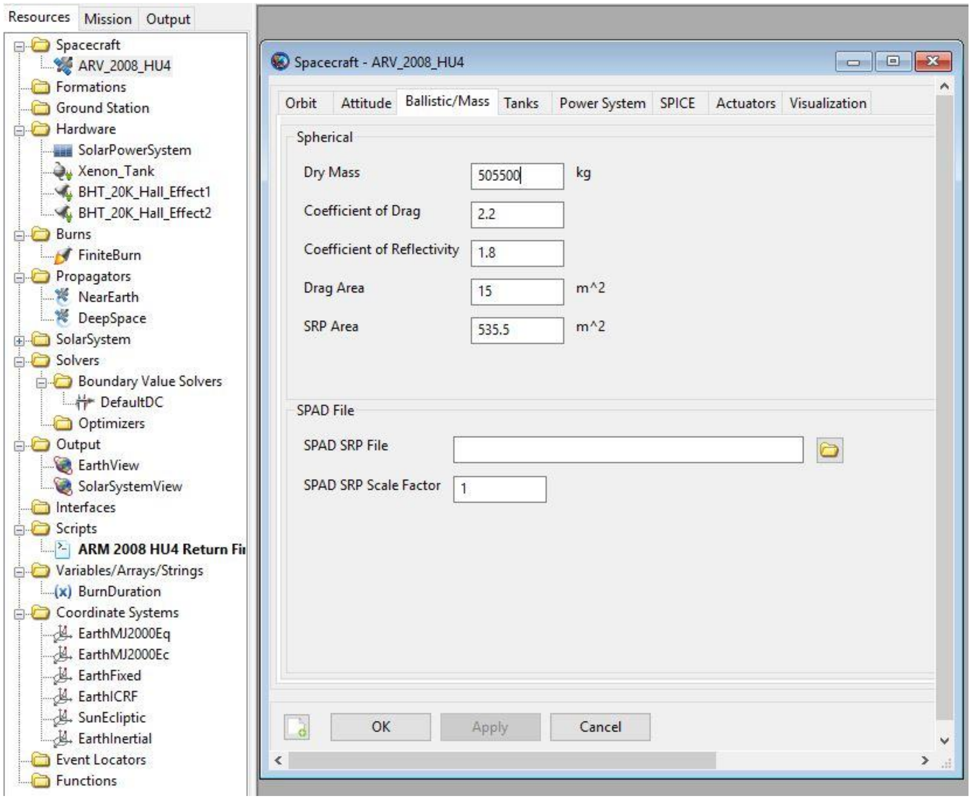

Figure 8.3 Dry Mass of the ARV with 2008 HU4

In this study, asteroid 2008 HU4 is assumed to be approximately 500 metric tons, or 500,000 kg. Therefore its mass combined with the spacecraft's 5,500 kg dry mass make the ARV towing 2008 HU4 roughly 505,500 kg in total dry mass. With the capture bag fully deployed, the spacecraft then has a height of 15 m, and when combined with its maximum width of 35.7 m, its new SRP area becomes 535.5 m^2.

In addition to the spacecraft's and asteroid's combined mass of 505,500 kg, expended propellant is also taken into account. Approximately 384.867 kg of propellant is used to reach the asteroid [2] with the ARV's two-thruster configuration. [1]

The two remaining propagators are left mostly the same as they were configured for the ARV's approach. The NearEarth propagator's Accuracy value is changed to equal 1e-007 in order to speed up computational time, and the DeepSpace propagator's maximum step size is left at a much larger value of 86400. Maneuvering toward Earth does not require the much smaller iterations used in the approach as Earth is a much larger celestial body with enough gravity to aid in the return trajectory. The EarthView output is also updated to show Luna and collect data every 1 step instead of 50 to provide a smoother plot. The larger propagator steps allow for this without causing the program to exceed its maximum number of rendered points in the plot.

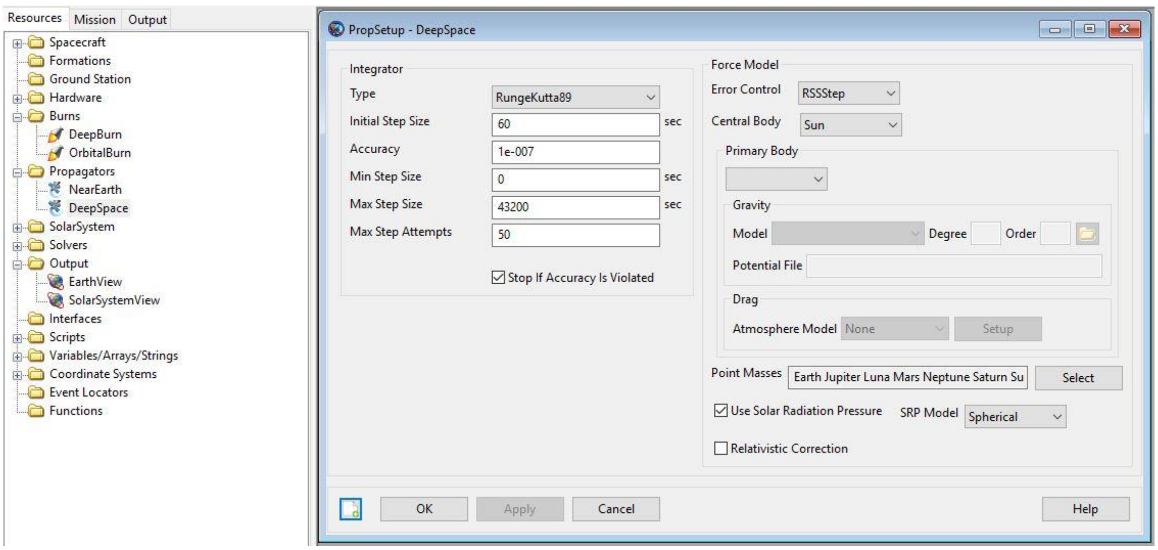

Figure 8.4 Deep Space Propagator for ARV Return

Here the Max Step Size value is set to 43200 to allow for faster computational time of the target trajectory. This value can be adjusted as discussed in Figures 7.29 and 7.30 as needed to expedite the computation process.

8.3 Return Mission Sequence

Much like the Resources tab, the Mission Sequence for the return has a very similar structure compared to the approach sequence. The initial Propagation in Deep Space command is set for 7 elapsed days this time, giving a week to run through tests and preparations to begin the thruster start-up and initiate the return maneuver.

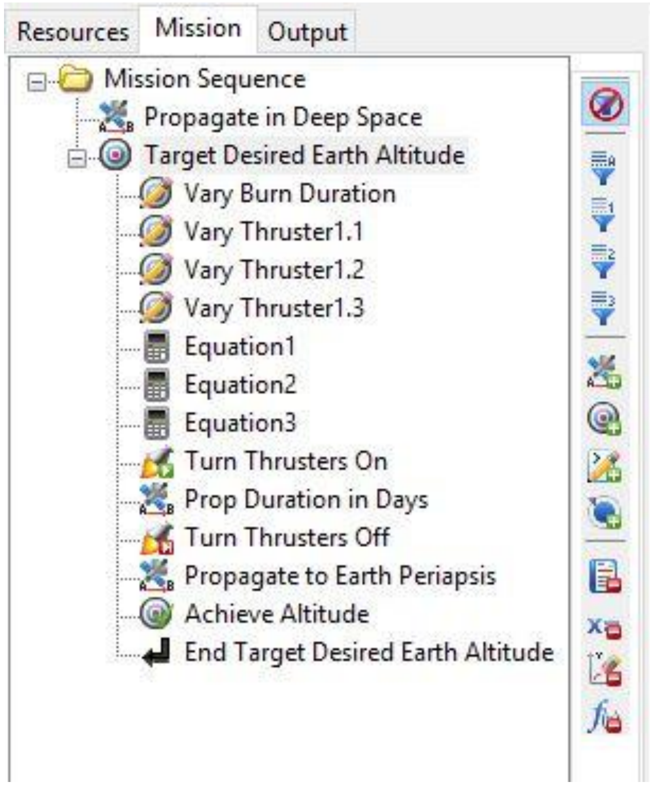

Figure 8.5 ARV Return Mission Sequence

The same finite burn variation commands remain for the return sequence.

It is assumed that moving a 500-metric-ton asteroid with two electric thrusters will take a considerable amount of time, around 9 years to return to the vicinity of Earth. [2] For this reason, an initial guess for the burn duration is set at 3400 days, or roughly 9.3 years, which is close to the estimate given in the ARM design summary by Wagner and Wie.

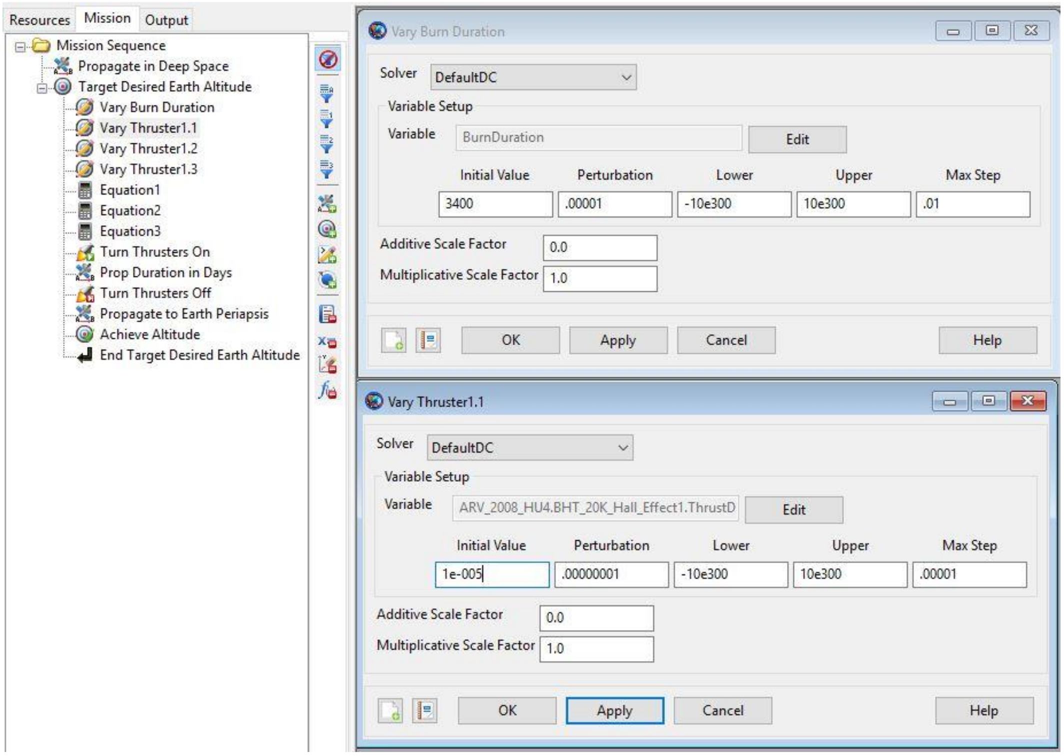

Figure 8.6 ARV Return Vary Commands

The Burn Duration Initial Value is set at 3050 days, while each Vary Thruster command's Perturbation and Max Step values are set to 0.00000001 and 0.00001, respectively. Larger maximum steps in the propagators call for smaller perturbations in maximum steps in the vary commands to have a reasonable level of accuracy.

Just like in the approach mission sequence, the burn duration variable is inserted in the Prop Duration in Days propagate command, controlling how long the thrusters are activated. The Propagate to Earth Periapsis command utilizes the NearEarth propagator and has the stopping condition parameter set as reaching periapsis for Earth orbit. Finally, the Achieve command is set for the spacecraft to reach 450,000 km in altitude above Earth.

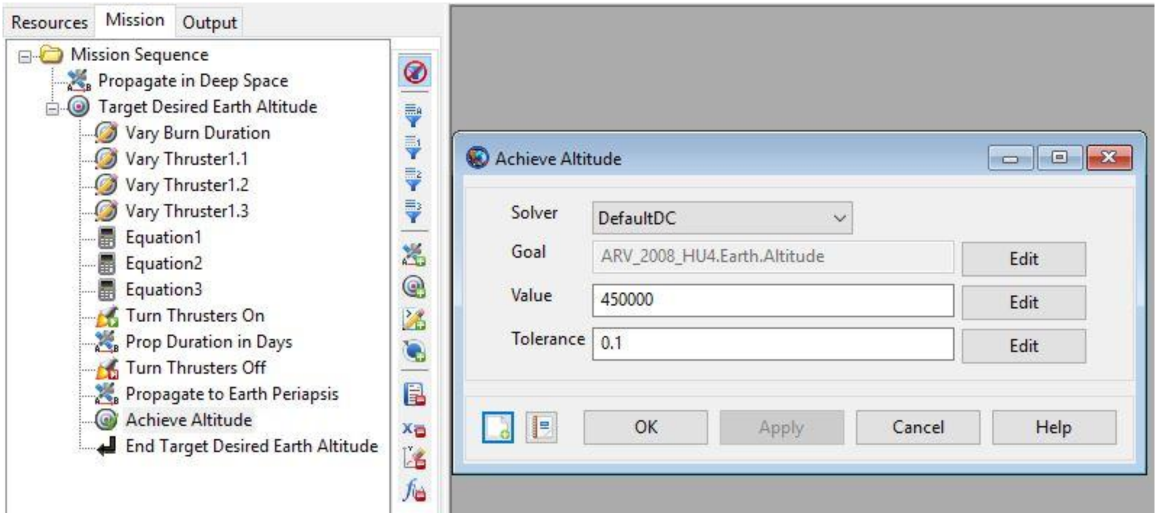

Figure 8.7 ARV Return Intended Altitude Goal

450,000 km is chosen as the intended altitude as that would place the ARV and 2008 HU4 in High Earth Orbit (HEO) just beyond the orbit of the moon, ideal for a subsequent lunar gravity assist to establish a High Lunar Orbit, which is not covered in this study.

8.4 ARV Return Results

The results for the ARV's return to Earth with 2008 HU4 are as follows, rounded to the nearest thousandths decimal place.

Table 8.1 Return Results

Burn Duration	3050.001 Days
Thruster V-Direction	3.530 m/s
Thruster N-Direction	-1.010 m/s
Thruster B-Direction	-44.488 m/s

Take note that in this table, the units for the thrust directions are in m/s, not km/s as was the case with the initial approach values in Table 7.1.

As with the initial approach results, the ΔV is calculated with each thruster direction value used as a component of a three-dimensional vector.

$$|\Delta V| = \sqrt{\Delta V_x^2 + \Delta V_y^2 + \Delta V_z^2} = \sqrt{(3.530)^2 + (-1.010)^2 + (-44.488)^2} \quad (8.1)$$

$$= 44.63925563895527 \text{ m/s}$$

Therefore ARV return ΔV = 44.639 m/s, rounded to the nearest thousandths place. With this ΔV, the spacecraft arrives within 450,000 km of Earth on 23 April 2026.

Table 8.2 ARV Return Results

Time to Return to Earth	3058 Days or 8.38 Years
ΔV Required	44.639 m/s
Propellant Remaining	5961.700 kg

This represents the return of the ARV with 2008 HU4 to within 450,000 km/s of Earth. The time to return to Earth is based on the 08 December 2017 starting epoch of the spacecraft and does not include the prior five months of evaluation and de-tumbling of the asteroid. The remaining propellant is shown in the Target Command Summary. Refer to Appendix A.2 for more information.

With the initial mission time of 24 May 2016 to the ARV with 2008 HU4 reaching an Earth altitude of 450,000 km on 23 April 2026, the total length of the Asteroid Redirect Mission is approximately 9.9 years.

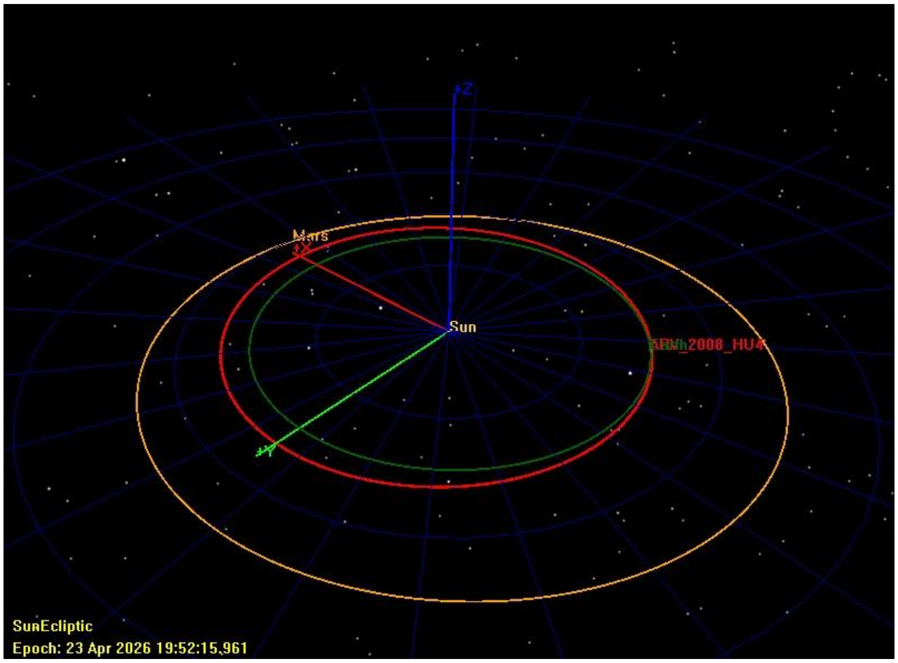

Figure 8.8 Heliocentric View of Asteroid Return

It takes approximately 9.36 years for the ARV to maneuver the asteroid to Earth, finally converging at a region where the orbits of 2008 HU4 and Earth are closest each decade.

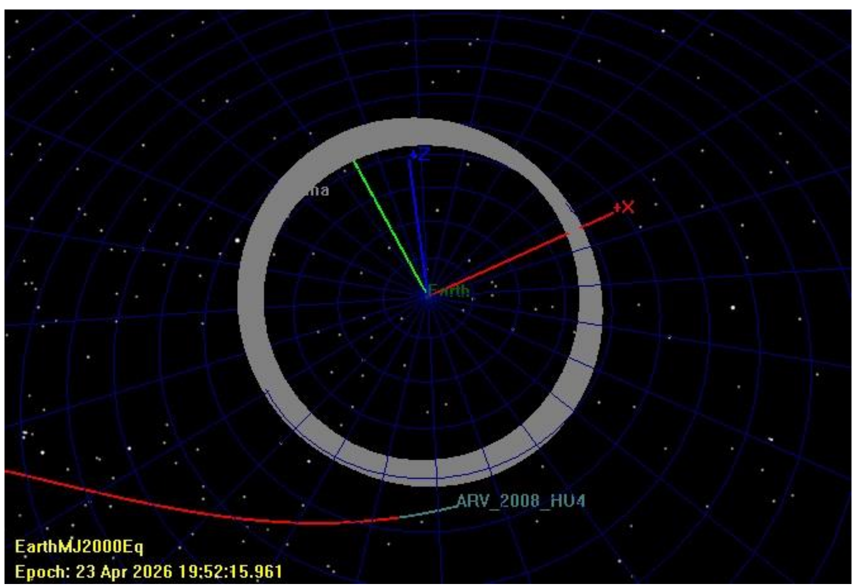

Figure 8.9 Earth View of ARV Return with Asteroid

The Earth Orbit View is set to include Luna, Earth's moon (represented by the gray orbit), to illustrate how close the ARV approaches lunar orbit.

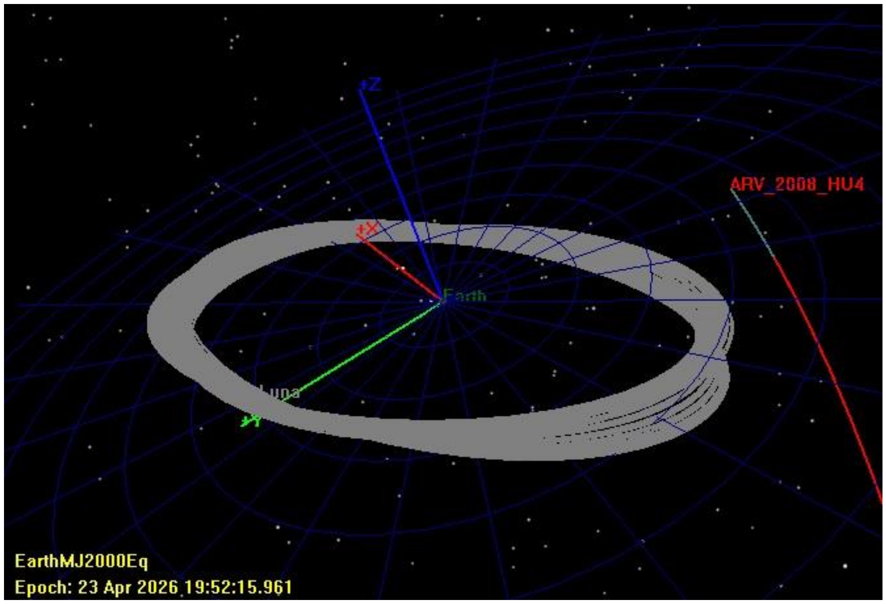

Figure 8.10 Alternate View of ARV Returning to Earth with Asteroid

The red lines in the background show the spacecraft's slow but steady orbit occurring over a period of 8.38 years before converging on Earth's orbit. The gray ellipses represent the moon's orbits over that same time period.

Solver Window - Target 'Target Desired Earth Altitude' DefaultDC {SolveMode = Solve, ExitMode = SaveAndContinue, ShowPro...

Control Variable	Current Value	Last Value	Difference
DeepBurnDuration	3050.000768456955	3050.000768456955	0
ARV_2008_HU4.BHT_20K_Hall_Effect1	0.003528666319497979	0.003528666319497979	-4.336808689942018e-19
ARV_2008_HU4.BHT_20K_Hall_Effect1	-0.00100586786760278	-0.00100586786760278	2.168404344971009e-19
ARV_2008_HU4.BHT_20K_Hall_Effect1	-0.04448794058342194	-0.04448794058342194	0
Constraints	**Desired**	**Achieved**	**Difference**
(==) ARV_2008_HU4.Earth.Altitude	450000	450000.0489651319	0.04896513186395168

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Figure 8.11 GMAT Solver Window for ARV Return

GMAT provides this solver window as a form of output, which is used to collect the final data for the ARV's return. As can be seen, the altitude constraint of 450,000 km is successfully reached.

CHAPTER IX

CONCLUSION

9.1 Comparison to Previous Research

Table 9.1 Comparison of Approach ΔV

This Study	ΔV = 0.406 km/s
Wagner and Wie	ΔV = 0.161 km/s
Brophy and Friedman	ΔV = 2.800 km/s

The approach ΔV for this study is 0.245 km/s higher than the estimate by Wagner and Wie. The Brophy and Friedman study has a significantly higher ΔV to account for its lunar gravity assist maneuver.

Table 9.2 Comparison of Return ΔV

This Study	ΔV = 44.639 m/s
Wagner and Wie	ΔV = 311.0 m/s
Brophy and Friedman	ΔV = 0.170 m/s

This time the Wagner and Wie study has a ΔV estimate 266.361 m/s greater than what was found in this study, while this study finds a ΔV value 44.469 m/s more than Brophy and Friedman's ΔV estimate.

There are several factors to account for the differences in ΔV requirements between these three studies. The mass and configuration of the spacecraft, the propellant usage, and the maneuvers executed during the mission all greatly affect the ΔV given in each study.

Table 9.3 Comparison of Approach Timeframe

This Study	400.0 Days or 1.10 Years
Wagner and Wie	277.8 Days or 0.76 Years
Brophy and Friedman	620.9 Days or 1.70 Years

As can be seen, this study falls roughly in the middle of the other two studies in terms of approach timeframe.

Table 9.4 Comparison of Return Timeframe

This Study	3058.0 Days or 8.38 Years
Wagner and Wie	3408.2 Days or 9.33Years
Brophy and Friedman	2191.5 Days or 6.00 Years

Again, this study falls between the timeframes of the other two studies, 350.2 days less than the Wagner and Wie return timeframe.

9.2 Suggestions for Improvement

While this orbital simulation works as a simplistic modeling of a very complex mission, there are several ways this study can be significantly improved to be more accurate. The first is to introduce lunar gravity assists to the mission sequence. These maneuvers were omitted from this study due to time constraints, but they would greatly benefit the overall simulation in terms of realism. Introducing lunar gravity assists could also help alleviate propellant consumption inaccuracies encountered in this study. Furthermore, the power system parameters were largely left at default settings for the sake of simplicity but could also improve propellant consumption if they were configured specifically to match the spacecraft's hardware parameters.

Beyond improving the orbital simulation itself, the study could be expanded to include other asteroid candidates as well, such as 2006 RH120, which is estimated to have a shorter total flight time and smaller ΔV requirements, although the optimal mission launch time would be six years later than that of 2008 HU4. [2] Having a larger variety of asteroid candidates would make this study much more thorough and aid in terms of deciding on future endeavors and narrowing down options for the Asteroid Redirect Mission.

Another phase of the Asteroid Redirect Mission that was omitted from this study due to complexity and time constraints is the asteroid's insertion into High Lunar Orbit and the subsequent station-keeping effort to maintain the asteroid's position for future missions. This study's simulation ends as the ARV reaches a distance set for High Earth Orbit, which can then lead to maneuvering into High Lunar Orbit. Including this aspect of the mission can help give a better perspective on the long term goals of what to do with the asteroid once it has been retrieved. It should be noted that the propellant required for station-keeping over a period of many years is not carried onboard the ARV and would require refueling upon returning to Earth orbit.

One particular concern with this mission is the apparent lifetime of the Busek Hall Effect Thrusters. The Asteroid Redirect Mission is a long-term mission requiring extended use of the thrusters, putting significant wear on them. The primary life-limiting process is the erosion of the acceleration channel caused by divergent accelerated ions. [14] For the BHT-20K, tests predicted it to have a lifetime of up to 33,000 hours, or 3.76 years. [10] Because of this, a spacecraft like the Asteroid Redirect Vehicle referenced in this study would most likely require additional spare thrusters onboard that could be powered on when the active thrusters wear out over time, which is a very likely possibility given that the total flight time for the Asteroid Redirect Mission is approximately 9.9 years based on results of the orbital simulation.

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APPENDIX A

ORBITAL PARAMETERS

This appendix contains the specific orbital parameters given in GMAT's mission command summary for both the ARV approach and the return of the Asteroid Redirect Mission.

A.1 ARV Approach to Asteroid 2008 HU4 (Asteroid-Centric Coordinates)

```
Target Command: Target Desired Bdot Trajectory
 Spacecraft       : ARV
 Coordinate System: AsteroidInertial

 Time System   Gregorian                     Modified Julian
 ----------------------------------------------------------------------
 UTC Epoch:    14 Jul 2017 22:39:35.819      27949.4441645715
 TAI Epoch:    14 Jul 2017 22:40:11.819      27949.4445812382
 TT  Epoch:    14 Jul 2017 22:40:44.003      27949.4449537382
 TDB Epoch:    14 Jul 2017 22:40:44.003      27949.4449537350

 Cartesian State                       Keplerian State
 ---------------------------           --------------------------------
 X  =   0.6886413842205 km             SMA  =  -0.0000000000000 km
 Y  =  -0.2057885801669 km             ECC  =  7.560984846e+015
 Z  =  -24.989640601000 km             INC  =   89.999983490326 deg
 VX =   3.0426031109889 km/sec         RAAN =   343.36270731565 deg
 VY =  -0.9091961410715 km/sec         AOP  =   271.64744491776 deg
 VZ =   0.0913326234805 km/sec         TA   =   359.99999669335 deg
                                       MA   =  -25001538560.912 deg
                                       HA   =   359.99999669335 deg

 Spherical State                       Other Orbit Data
 ---------------------------           --------------------------------
 RMAG =   24.999974265245 km           Mean Motion        =  9.608073e+014 deg/sec
 RA   =  -16.637866710195 deg          Orbit Energy       =   5.046206481 km^2/s^2
 DEC  =  -88.352558570187 deg          C3                 =   10.09241296 km^2/s^2
 VMAG =   3.1768558295979 km/s         Semilatus Rectum   =  9.575755376e+031 km
 AZI  =   0.0005742632127 deg          Angular Momentum   =   79.4213139843 km^2/s
 VFPA =   90.000003488026 deg          Beta Angle         =  -68.153846127858 deg
 RAV  =  -16.637292209514 deg          Periapsis Altitude =   24.995974265245 km
 DECV =   1.6474449177561 deg          VelPeriapsis       =   3.1768558295979 km/s

 Planetodetic Properties               Hyperbolic Parameters
 ---------------------------           --------------------------------
 LST       =   340.27633933323 deg     BdotT          =   0.0000072066758 km
 MHA       =   182.56747230934 deg     BdotR          =   24.999974265244 km
 Latitude  =  -88.322165990079 deg     B Vector Angle =   89.999983483499 deg
 Longitude =   157.70886702389 deg     B Vector Mag   =   24.999974265245 km
 Altitude  =   24.995987664529 km      DLA            =   1.6474449177561 deg
                                       RLA            =  -16.637292209514 deg
Spacecraft Properties
 ------------------------------
 Cd                    =   2.200000
 Drag area             =   15.00000 m^2
 Cr                    =   1.800000
 Reflective (SRP) area =   381.9900 m^2
 Dry mass              =   5500.0000000000 kg
 Total mass            =   14615.133124320 kg

 Tank masses:
   Xenon_Tank:            9115.1331243202 kg
```

A.2 ARV Return with 2008 HU4 (Earth Equatorial Coordinates)

```
Target Command: Target Desired Earth Altitude
 Spacecraft       : ARV_2008_HU4
 Coordinate System: EarthMJ2000Eq

 Time System   Gregorian                     Modified Julian
 ----------------------------------------------------------------------
 UTC Epoch:    23 Apr 2026 19:52:15.961      31154.3279625169
 TAI Epoch:    23 Apr 2026 19:52:51.961      31154.3283791836
 TT  Epoch:    23 Apr 2026 19:53:24.145      31154.3287516836
 TDB Epoch:    23 Apr 2026 19:53:24.147      31154.3287517017

 Cartesian State                       Keplerian State
 ---------------------------           --------------------------------
 X  =  -144466.29530651 km             SMA  =  -364433.26619709 km
 Y  =  -432907.31229029 km             ECC  =   2.2522956258464
 Z  =    1340.2934297991 km            INC  =   27.203846755105 deg
 VX =    1.4226042465980 km/sec        RAAN =   251.21817497767 deg
 VY =  -0.4723544670722 km/sec         AOP  =   0.3680735832080 deg
 VZ =    0.7704748685600 km/sec        TA   =   359.99999914623 deg
                                       MA   =  -0.0000006634498 deg
                                       HA   =   359.99999947021 deg

 Spherical State                       Other Orbit Data
 ---------------------------           --------------------------------
 RMAG =   456378.18517154 km           Mean Motion        =  2.869735e-006 deg/sec
 RA   =  -108.45446470627 deg          Orbit Energy       =   0.546877135 km^2/s^2
 DEC  =   0.1682667351771 deg          C3                 =   1.093754271 km^2/s^2
 VMAG =   1.6853940512783 km/s         Semilatus Rectum   =   1484276.7753651 km
 AZI  =   62.796633943836 deg          Angular Momentum   =   769177.078421 km^2/s
 VFPA =   90.000000000046 deg          Beta Angle         =  -5.3561937107381 deg
 RAV  =  -18.367975043973 deg          Periapsis Altitude =   450000.04887154 km
 DECV =   27.203239054259 deg          VelPeriapsis       =   1.6853940512783 km/s

 Planetodetic Properties               Hyperbolic Parameters
 ---------------------------           --------------------------------
 LST       =   251.88228833877 deg     BdotT          =   717494.99072466 km
 MHA       =   149.93705438171 deg     BdotR          =   161619.21752010 km
 Latitude  =   0.1199755856623 deg     B Vector Angle =   12.694280281157 deg
 Longitude =   101.94523395706 deg     B Vector Mag   =   735472.52374700 km
 Altitude  =   450000.04896513 km      DLA            =   24.099033315008 deg
                                       RLA            =   10.735050775627 deg


 Spacecraft Properties
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 Cd                    =   2.200000
 Drag area             =   15.00000 m^2
 Cr                    =   1.800000
 Reflective (SRP) area =   535.5000 m^2
 Dry mass              =   505500.00000000 kg
 Total mass            =   511461.69863565 kg

 Tank masses:
    Xenon_Tank:            5961.6986356463 kg
```