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

By

Torrey Harriel

A Thesis

Submitted to the Faculty of Mississippi State University in Partial Fulfillment of the Requirements for the Degree of Master of Science in Aerospace Engineering in the Department of Aerospace Engineering

Mississippi State, Mississippi

August 2016

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2016

Asteroid Redirect Mission (ARM) using Solar Electric Propulsion (SEP) for research,

mining, and exploration endeavors of Near-Earth Objects (NEOs)

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Pages in Study 75

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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
- Isp 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

1

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-meterdiameter 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 fullcycle 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.

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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 (v) 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

$$\Delta V = g \cdot I_{sp} \cdot \ln\left(\frac{m_{init}}{m_{init} - m_{prop}}\right)$$
(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. $B \cdot T$ (BdotT) and $B \cdot R$ (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/s² 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 lifecycle 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 everexpanding framework of planetary defense as well, as it would serve as a proof-ofconcept 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. H₂SO4 or HF dissolution
	Metal Oxides (for oxygen)	 A. Ionic Liquid Acid dissolution B. H₂SO4 or HF dissolution C. Hydrogen Reduction D. Carbothermal Reduction E. Molten Oxide Electrolysis
	Metal Oxides (for metal)	A. Molten Oxide ElectrolysisB. 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 ElectrolysisB. 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 proofof-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 precapture 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.

Ground Station		
Hardware	Steroid - Asteroid_2008_HU4	
Dorns Dorns Dorns Dorns Dorns Dorns Dorns SolarSystem SolarSystemBarycenter Sun Venus Venus Venus Venus Uranus Uranus Uranus Venue Neptune Venue Neptune Venus Neptune Venus Solvers Solvers Solvers Solvers Solvers Cordinate Systems Coordinate Systems	Properties Orbit Orientation Visualization Ephemeris Data Central Body Sun Ephemeris Source SPICE NAIF ID 3409707 SPK Files AGMATI SPKwid31615.bsp Add Remove	
Functions	d OK Apply Cancel	Help

Figure 7.1 GMAT Custom Celestial Body Interface

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

Generations	🛞 Asteroid - Asteroid	d_2008_HU4		3
Ground Station Ground S	Properties Orbit Options Mu Equatorial Radius Flattening PCK Files	Orientation Visualization	km^3/sec^2 km	
Asteroid_2008_HU4 Asteroid_2008_HU4 Divers Output Interfaces CoordinateSystems Event Locators Functions	ОК	Apply	Cancel Hel	

Figure 7.2 Asteroid Properties

For 2008 HU4's properties, its approximate mass of 500,000 kg means it has a gravitational parameter $\mu = 3.337e-014 \text{ km}^3/\text{s}^2$ (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.

E-Co Spacecraft	Spacecraft - ARV						
	Orbit Attitude E	allistic/Mass Tanks	Power Syster	n SPICE Actuators	Visualization		
Hardware		Parameter and a second		Elements			
	Epoch Format	UTCGregorian	~	SMA	-32593.2159927297	km	
	Epoch	24 May 2016 11:59:28	.000	FCC	1 202072540116102	_	
BHT_20K_Hall_Effect2	Coordinate System	EarthMJ2000Eq	~	INC	1.202872348116182		
Burns	State Type	Keplerian	~	INC .	28.80241266404147	deg	
Propagators				RAAN	25.2	deg	
Solvers				AOP	239.9999999999999	deg	
				TA	350.000000000003	deg	
Gripts Variables/Arrays/Strings							
Coordinate Systems							
Functions							
		Apply	Cancel				Help
	d UK	Арріу	Cancel				пер

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.

The Attitude Dunistic/1	Mass Tanks	Power System	SPICE Actuators	Visualization	
Spherical					
Dry Mass	5500	kg			
Coefficient of Drag	2.2]			
Coefficient of Reflectivity	1.8]			
Drag Area	15	m^2			
SRP Area	381.99	m^2			
SPAD SRP File SPAD SRP Scale Factor	1				

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]

Resources Mission Output Spacecraft ARV Formations	
Ground Station Hardware SolarPowerSystem Zenon_Tank SHT_20K_Hall_Effect1 BHT_20K_Hall_Effect2 Burns Propagators Propagators	Properties Allow Negative Fuel Mass Fuel Mass 9500
SolarSystem SolarSystem Solvers Output Interfaces Scripts Coordinate Systems Event Locators	OK Apply Cancel Help Spacecraft - ARV Image: Concel Help Orbit Attitude Ballistic/Mass Tanks Power System SPICE Actuators Visualization Available Selected
	-> Xenon_Tank

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.

Spacecraft	SolarPowerSystem -	SolarPowerSystem				
 Formations Ground Station Hardware SolarPowerSystem Xenon_Tank 	General Epoch Format Initial Epoch	UTCGregorian 24 May 2016 11:59:28.000]	Coefficients Bus Coeff1 Bus Coeff2	0.3	kW kW*AU
BHT_20K_Hall_Effect1 BHT_20K_Hall_Effect2 BHT_20K_Hall_Effect2 BHTs Propagators	Initial Max Power 40 Decay Rate 5 Margin 5		kW percent/year percent	Bus Coeff3 Solar Coeff1 Solar Coeff2	0 1.32077 -0.10848	kW*AU^2
Solarsystem Solvers Output Interfaces Solvers	Shadow Model Shadow Bodies	DualCone ~	Select	Solar Coeff3 Solar Coeff4 Solar Coeff5	-0.11665 0.10843 -0.01279	
Gripts Gripts	ОК	Apply Cancel				Help
	Spacecraft - ARV	listic/Mass Tanks Power Syste	m SPICE Actuator	s Visualization		
	Power System SolarP	owerSystem ~	Actuator	3 113001201011		

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.

Resources Mission Output	ElectricThruster - BHT 20K Hall I	iffect1		Space	cecraft - ARV	/		0	
Spacecraft	Coordinate System		_	:/Mass	Tanks Po	wer System SPIC	E Actuators V	isualization	Ŷ
- Ground Station	Coordinate System	Local	\sim						
😑 🗀 Hardware	Origin	Earth	~	N N	Available	Thrusters		Selected Thrusters	
SolarPowerSystem	Axes	VNB	~					BHT_20K_Hall_Effect1	
BHT_20K_Hall_Effect1	Thrust Vector						->		
BHT_20K_Hall_Effect2	ThrustDirection1	1					<-		
Propagators	ThrustDirection2	0	=						
🗈 🫅 SolarSystem	ThrustDirection3		=						
Solvers	Dute Curls	0.35	_				=>		
interfaces	They Cycle	0.25	=						
Scripts	I nrust scale Factor						<=		
Coordinate Systems	Mass Change			1					
- Event Locators	Decrement Mass								J
	lank	Xenon_lank	×	<					х
	GravitationalAccel	9.81	m/s^2						
	Thrust Config.								
	Thrust Model	FixedEfficiency	~						
	Minimum Usable Power	5	kW						
	Maximum Usable Power	20	kW						
	Fixed Efficiency	0.7							
	lsp	2750	s						
	Constant Thrust	1.08	N						
	Conf	igure Polynomials							
	ок Арр	ly Cancel	Help						
		2.							

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]

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

$$\overline{T} = f_d * f_s * \frac{(2\eta P)}{(I_{sp}g_0)} * \overline{R_{\iota T}} * \widehat{T}$$
(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 $\overline{R_{tT}}$ = 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:

$$f_d = 0.25$$

 $\eta = 0.70$
 $I_{sp} = 2750 \text{ s}$
 $g_0 = 9.81 \text{ m/s}^2$
 $f_s = 1$

- - -

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.

	S FiniteBurn - FiniteBurn
Ground Station	Thruster BHT_20K_Hall_Effect1 V
FiniteBurn	OK Apply Cancel Help
- Solvers - Output	C:\Users\Torrey\Documents\Thesis\Results\ARM to 2008 HU4 Periapsis Bdot Finite Burn Non-Fuel 25 km.script
Interfaces ARM to 2008 HU4 Periaps ARM to 2008 HU4 Periaps Variables/Arrays/Strings Coordinate Systems Event Locators Functions	<pre>292 % 293 % 294 % 295 296 Create FiniteBurn FiniteBurn; 297 GMAT FiniteBurn.Thrusters = {BHT_20K_Hall_Effect1, BHT_20K_Hall_Effect2}; 298 GMAT FiniteBurn.ThrottleLogicAlgorithm = 'MaxNumberOfThrusters'; 299</pre>
	Active Script Save Sume Save Sume Dun Save Save Art Close

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.

> Formations				Force Model
Ground Station	Integrator			
Hardware	Туре	RungeKutta89 ~		Error Control RSSStep ~
Burns	Initial Step Size	600	sec	Central Body Earth V
Propagators NearEarth	Accuracy	1e-013	1	Primary Body
	Min Step Size	0	sec	Earth V
NearAsteroid	Max Step Size	600	sec	Gravity
Solvers	Max Step Attempts	50	í	Model JGM-2 V Degree 8 Order 8
Output Interfaces Carpits ARM to 2008 HU4 Periaps Variables/Arrays/Strings Carpits Contents		Stop If Accuracy Is Violated	_	Potential File JGM2.cof Drag Atmosphere Model None V Setup
 Event Locators Functions 				Point Masses Luna Sun Select
				Use Solar Radiation Pressure SRP Model Spherical V

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.

Formations Ground Station	Integrator			Force Model		
Hardware	Туре	RungeKutta89	~	Error Control RSSStep ~		
Burns Propagators	Initial Step Size	600	sec	Central Body Sun 🗸		
NearEarth	Accuracy	1e-007		Primary Body		
CheepSpace	Min Step Size	0	sec	~		
SolarSystem	Max Step Size	10800	sec	Gravity		
Solvers	Max Step Attempts	50	_	Model 🗸 Degree Order 🔄		
Interfaces Scripts	Stop If Accuracy Is Violated			Potential File		
Coordinate Systems				Drag Atmosphere Model None Setup		
Event Locators Functions				Point Masses Earth Jupiter Luna Mars Neptune Saturn Su Select		
				Use Solar Radiation Pressure SRP Model Spherical		
				Relativistic Correction		

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.

- Contractions				Force Model
Ground Station	Integrator			
🛅 Hardware	Туре	RungeKutta89	~	Error Control RSSStep ~
🛅 Burns	Initial Step Size	1	sec	Central Body Sun 🗸
Propagators	Accuracy	1e-007	-	Primary Body
2 DeepSpace	Mar Char Clas	10 007	_	i iiiiii yoody
NearAsteroid	win step size	0	sec	· · · · · · · · · · · · · · · · · · ·
🚞 SolarSystem	Max Step Size	20	sec	Gravity
Solvers	Max Step Attempts	50		Model Degree Order 🦲
- Cutput - Ci Interfaces - Ci Scripts		Stop If Accuracy Is Viola	ted	Potential File
ARM to 2008 HU4 Periaps Variables/Arrays/Strings Coordinate Systems				Drag Atmosphere Model None - Setup
Event Locators Functions				Point Masses Asteroid_2008_HU4 Sun Select
				Use Solar Radiation Pressure SRP Model Spherical Relativistic Correction
	ОК	Apply Cancel		Heip

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.

Resources Mission Output	S DifferentialCorrec	tor - DefaultDC	
Ground Station Ground Station	Options Algorithm Max Iterations Derivative Method	NewtonRaphson v 1000 ForwardDifference v	
Solary Sterring Solvers 	Output Show Progress Report Style	Normal	 1
Output Interfaces	Report File	DifferentialCorrectorDC1.data	
Scripts Variables/Arrays/Strings Coordinate Systems Event Locators Functions	ОК	Apply Cancel	Help

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.

Percurrent Mission October		
Green Spacecraft Formations		
🤤 Ground Station	S CoordinateSystem - SunEcliptic	
⊖- 🗁 Hardware 	Origin Sun 🗸	
BHT_20K_Hall_Effect1	Axes	
BH1_20K_Hall_Effect2	Type MJ2000Ec 👻	
Propagators	Primary Secondary	
🗊 🛅 SolarSystem		
Solvers	Ativitie chock	
🗄 🧰 Output	X: 🔻 Y: 👻 Z: 💌	
Interfaces		
B Scripts		
Coordinate Systems		
EarthMJ2000Eq		
EarthMJ2000Ec		
EarthFixed		
AsteroidInertial		
Asteroid_2008_HU4Fixed		
Functions	Ld UK Apply Cancel	Help

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]

Gracecraft Gracecraft Formations	
Ground Station	CoordinateSystem - Asteroidinertial
Hardware SolarPowerSystem Xenon_Tank MHT_20K_Hall_Effect1 BHT_20K_Hall_Effect2 Burns	Origin Asteroid_2008_HU4 Axes Type BodyInertial Primary Secondary
Propagators	
SolarSystem	ATMOD Ebocu
Output Output Output Output Output Output Output Output Output Output	X: Y: Z: •
Variables/Arrays/Strings	
는 Coordinate Systems - 섎. EarthMJ2000Eq - 섎. EarthMJ2000Ec - 섎. EarthFixed	
Event Locators	OK Apply Cancel Help

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.

📄 🧰 Spacecraft	Contraction - Califinitiem			
Formations Foround Station Hardware Burns SolarSystem SolarSystem SolarSystem Output	Plot Option Collect data every 50 step Update plot every 50 cycle Enable Stars Number of stars 7000	View Object Spacecraft Celestial Object	Selected Spacecraft	Draw Object
Coordinate Systems Coordinate Systems	Number of points to redraw 0 (Enter 0 to redraw whole plot) Show Plot Show Labels Drawing Option	View Definition		
C Event Locators	Draw WireFrame Draw Ecliptic Plane Oraw Aces Draw Grid Draw Grid Draw Grid Solver Iterations Current	Coordinate System EarthMI20001 View Point Reference Earth View Point Vector Vector View Scale Factor 4 View Direction Earth	Eq ~ ~ 10000 -30000	30000 km
	View Option	View Up Definition Coordinate System EarthMJ2000Eq	✓ Axis Z ✓	
	OK Apply	Cancel		Help

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.

Ground Station Ground Station		Black Carling	Africa Oblicat			
Solurystem Solurystem Chable Stars ∑ Chable Stars Solurystem Solurystem Science Constellations Selected Celestial Object Output Safrystem View Solarsystem View Selected Celestial Object Jupiter Solarsystem View Safrystem View Solarsystem View Selected Celestial Object Jupiter Solarsystem View Safrystem View Solarsystem View Selected Celestial Object Jupiter Solarsystem View Safrystem View Solarsystem View Selected Celestial Object Jupiter Solarsystem View Solarsystem View View Points to redraw whole plot) Usa Mercury Number of tars Solarsystem View Show Plot Solarsystem Trawing Option View Definition Oraw Kales Draw WireFrame Oraw Solar System View Point Reference Solar Oraw Xaes Oraw Xaes View Point Reference Solar View View View View View View View View	Ground Station Hardware Burns Propagators	Collect data every 100 step Update plot every 50 cycle	Spacecraft	> AR	Selected Spacecraft V	
Vanable/Arzy/String: Coordinate/Systems Drawing Option Drawing Option Draw View Print Reference Draw XY Plane View Point Vector Vector Draw Sun Line Solver Iterations View Option View Up Efinition View Option View Up Efinition	SolarSystem SolarSystem Columnation Content Columnation Columnati	C Enable Stars Enable Constellations Number of stars 7000 Number of points to redraw (Enter 0 to redraw whole plot) Show Plot Show Labels	Celestial Object Jupiter Luna Mercury Neptune	<pre> Set</pre>	elected Celestial Object th teroid_2008_HU4 n rs	Draw Object
View Option View Up Definition	Devent Locators Event Locators Functions	Drawing Option Draw WireFrame Draw Schptic Plane Draw XY Plane Draw Axes Draw Axes Draw Grid Draw Sun Line Solver Iterations Current V	View Definition Coordinate System S View Point Reference S View Point Vector V View Scale Factor View Direction S	iunEcliptic v iun v /ector v 1	< 0 0	50000000 km
Coordinate system Sunectiplic V Axis Z V		View Option ☑ Use Initial View Def.	View Up Definition Coordinate System Sur	nEcliptic v	Axis Z 🗸	

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.

🖅 🛅 Spacecraft	S OrbitView - AsteroidView					
Formations Formations Formations Formations Hardware Propagators SolarSystem SolarSystem SolarSystemView Satarbifview Satarbifview	Plot Option Collect data every 10 step Update plot every 50 cycle ☑ Enable Stars ☑ Enable Constellations Number of stars 7000 Number of points to redraw (Enter 0 to redraw whole plot)	View Object Spacecraft Celestial Object Earth Jupiter Luna Mars v) > <=	Selected S ARV Selected Cel Asteroid_2008	estial Object	☑ Draw Object
Scripts Scripts Orables/Arrays/Strings Orables/Arrays/Strings Orables/Arrays/Strings Orables/Strens Orables/Strens Functions Functions	Show Plot Show Labels Drawing Option Draw WireFrame Draw Celiptic Plane Draw XP Iane Draw Aves Draw Grid Draw Grid Solver Iterations Current Solver Iterations Current	View Definition Coordinate System Acter View Point Reference View Point Vector View Scale Factor View Scale Factor View Direction Acter	oidInertial oid_2008_H r oid_2008_H	~ U ~ _ U ~	0] [100000] km
	View Option 🗹 Use Initial View Def.	View Up Definition Coordinate System Asteroid	dinertial	✓ Axis Z	· ·	
	ОК Арріу	Cancel				Help

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.

Resources Mission Output Spacecraft Formations Ground Station Hardware Burns Propagators SolarSystem		
Solvers Output Output Scripts Variables/Arrays/Strings	Variable - BurnDuration Variable Name	Value
Coordinate Systems Event Locators Functions	BurnDuration = 0	Cancel Help

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.

Resources Mission Output	[Propag	ate 3 Days Earth				
Mission Sequence Mission Sequence May Propagate 3 Days Earth May Propagate 12 Days DeepSpace May Target Desired Bdot Trajectory	@	Propaga Propaga	ators and Spacecraft ste Mode: None -		🗖 Backwa	ards Propagation 🔲 Propagate STM	Compute A-Matrix
Vary Burn Duration	2		Propagator			Spacecraft List	<u> </u>
Vary Thruster1.2	Ş.		NearEarth		ARV		
	悉		· · · · · · · · · · · · · · · · · · ·				
Equation3 	×	Stopping Stop Tol	g Conditions lerance: 1e-007				
Turn Thrusters Off			Parameter			Condition	*
	×a Ľe		ARV.ElapsedDays	=	3		
@ Achieve BdotR @ Achieve Altitude	<u>fa</u>						
End Target Desired Bdot Trajectory		Colors	🖾 Override Color For	This Segment	Orbit Color		
			B OK Apply Cancel]			Help

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.

Resources Mission Output	S Prop	agate 12 Days DeepSpace			
Mission Sequence K Propagate 3 Days Earth K Propagate 12 Days DeepSpace Ø Target Desired Bdot Trajectory	Propa Propa Propa	gators and Spacecraft gate Mode: None 💌	E	Backwards Propagation 📄 Propagate STM 🛛	Compute A-Matrix
- 🥥 Vary Burn Duration	Y	Propagator		Spacecraft List	<u>^</u>
- 9 Vary Ihruster1.1 - 9 Vary Thruster1.2 - 9 Vary Thruster1.3 - 8 Equation1	<u>₹</u>	DeepSpace	ARV 		
- I Equation2 - Equation3 - K Turn Thrusters On - K Prop Duration in Days	Stopp	ing Conditions folerance: 1e-007			
Turn Thrusters Off	B	Parameter		Condition	Â
- 25, Prop to Asteroid - @ Achieve BdotT - @ Achieve BdotR - @ Achieve Altitude	xa 16 <i>f</i> a	ARV.ElapsedDays	= 12		
🛛 📣 End Target Desired Bdot Trajectory	Color	s 🔲 Override Color	For This Segment Orbit Co	slor	
		OK Apply Cancel			Help

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.

esources Mission Output		
Propagate 3 Days Ear	S Target Desired Bdot Trajectory	
	Solver Name DefaultDC	~
Vary Thruster 1.3	Solver Mode Solve	~
Equation2	Exit Mode SaveAndContin	nue 🗸
	Apply Corrections	ss Window
Achieve BdotT	ОК Арі	ply Cancel Help
End Target Desire		

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.

Resources Mission Output	S Vary Burn Duration			
- 💥 Propagate 3 Days Ear 🥨 - 🎉 Propagate 12 Days D	Solver DefaultDC	~		
Target Desired Bdot	Variable Setup Variable BurnDuration Initial Value P 280	returbation Lower 001 -10e300	Edit Upper	Max Step
Achieve Bodr Achieve Adar Achieve Athitude	ParameterSelectDialog Object Type Variable Object List BurnDuration	Object Properties	UP DN	Selected Value(s)
			-> <- => <=	
		OK Cancel	Help	

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.

Resources Mission Output	Vary Thruster1.1				
🖃 🤭 Mission Sequence 💦 👝					
Propagate 3 Days Ear	Solver DefaultDC	~			
Target Desired Bdot	Variable Setup				
- 🥥 Vary Burn Duratic 🍸	Variable ADV DUT 20K LI-II	Effected Theorem Discosting 1	- 10		
🧭 Vary Thruster 1.1 🚏	ARV.DHT_20K_Hall_		Edit		
- 🥥 Vary Thruster 1.2 🔤	Initial Value	Perturbation Lower	Upper	Max Step	
- 🧭 Vary Thruster1.3	1e-005 .00	-10e300	10e300	.001	
Equation2	A LEVIN Confer Frankers	,			
	Additive Scale Factor 0,0				
Prop Duration in	Multiplicative Scale Factor 1.0				
Turn Thrusters OI					
- 🧏 Prop to Asteroid	ОК	Apply Cancel		Help	
- @ Achieve BdotT					
Achieve Altitude	ParameterSelectDialog			- T	×
End Target Desire	• Fundameterselectionalog			5000 D	~
NO 5563					
	Object Type	Object Properties		Selected Value(s)	
	Spacecraft ~	GravitationalAccel		UK_Hall_Effect1.Infustbire	ctionT
	Object List	ThrustDirection1 ThrustDirection2	UP		
	ARV	ThrustDirection3	DN		
		ThrustScaleFactor			
			->		
			<-		
	Attached Hardware List				
	Xenon Tank		=>		
	BHT_20K_Hall_Effect1		<=		
	SolarPowerSystem	-			
				_	
			C		/
			1		
		OK Cancel	Help		

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.

Resources Mission Output	Sequation 1	
Mission Sequence Massion Sequence Mass	Left-Hand Side Right-Hand Side ARV.BHT_20K_Hall_Effect2. = ARV.BHT_20K_Hall_Effect1.ThrustDirection1	
	OK Apply Cancel	Help
Equation1	Sequation2	
- Turn Thrusters OI	Left-Hand Side Right-Hand Side ARV.BHT_20K_Hall_Effect2. = ARV.BHT_20K_Hall_Effect1.ThrustDirection2	
Achieve BdotT Achieve BdotR	d UK Apply Cancel	Help
	© Equation3	
	Left-Hand Side Right-Hand Side ARV.BHT_20K_Hall_Effect2. = ARV.BHT_20K_Hall_Effect1.ThrustDirection3	
	OK Apply Cancel	Help

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.

Barrowsee Mission Output				
Mission Sequence				
Propagate 3 Days Earth	S Turn Thrusters On			
₩ Propagate 12 Days DeepSpan ● Target Desired Bdot Trajecto ● Vary Burn Duration ● Vary Thruster1.1 ● Vary Thruster1.2 ● ¥ary Thruster1.3 ● Equation1	Options Burn FiniteBurn ✓ Spacecraft ARV Edit			
Equation2 Equation3 Zero Turn Thrusters On Were Duration in Days	Cancel	Help		
Turn Thrusters Off	Duran Durantian in Duran			
Prop to Asteroid	w Prop Duration in Days			
- @ Achieve BdotT	Propagators and Spacecraft			
Achieve Altitude	Propagate Mode: None ~		Backwards Propagation Propagate STM [Compute A-Matrix
End Target Desired Bdot	Propagator		Spacecraft List	^
	DeepSpace		AP//	
	in the second se			
				×
	Stopping Conditions			
	Stop Tolerance: 1e-007			
	Parameter		Condition	^
	ARV.ElapsedDavs	=	BurnDuration	
	Celorr	1 I		
	Override Color For Th	is Segment	Orbit Color	
	OK Apply Cancel			Help 🗸
	<			ان <

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.

Resources Mission Output			
🖃 🤭 Mission Sequence	Urn Thrusters Off		
💥 Propagate 3 Days Earth 🦉	Options		
- Repagate 12 Days DeepSpa			
Vary Burn Duration	Burn FiniteBurn V		
- 🧭 Vary Thruster1.1 🛛 🐺	Spacecraf ARV Edit		
- 🥥 Vary Thruster1.2 🛛 🐺			
- Ø Vary Ihruster1.3			
Equation2	OK Apply Cancel	Help	
Equation3			
Prop Duration in Davr	Drop to Acternid		
Turn Thrusters Off	The to Ascend		
Prop to Asteroid	Propagators and Spacecraft		
- @ Achieve BdotT	Propagate Mode: None ~	Backwards Propagation	Propagate STM Compute A-Matrix
Achieve Altitude	Propagtor	Enver	roft list
End Target Desired Bdot	Machanal	ADV	
	NearAsteroid	AKV	
			· ·
	Stopping Conditions		
	Stop Tolerance: 1e-007		
	Parameter	Co	adition ^
	ARV.Asteroid_2008_HU4.Periapsis		
			v
	Colors	21 2000	
	Qverride Color Fr	r This Segment Orbit Color	
	Cancel		Help

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.

Resources Mission Output	😨 Achieve Bdo	tT	
Mission Sequence Mission Sequence Aropagate 3 Days Earth Propagate 12 Days DeepSpa Vary Druster Dearbox Vary Thruster 1.2 Vary Thruster 1.3 Vary Thruster 1.3	Solver Goal Value Tolerance	DefaultDC V ARV.AsteroidInertial.BdotT 0 0.0001	Edit Edit Edit
- Figuration 1 Control		OK Apply Cancel	Help
Prop Duration in Days	S Achieve Bdo	tR	
Prop to Asteroid Achieve BdotT G Achieve BdotT G Achieve Altitude Fachieve Altitude Fachieve Altitude Fachieve Altitude Fachieve Altitude Fachieve Altitude	Solver Goal Value Tolerance	DefaultDC ARV.AsteroidInertial.BdotR 25 0.0001 OK Amply Carcel	Edit Edit Edit
		on oppy cance	Theip
	🛞 Achieve Altit	tude	
	Solver Goal	DefaultDC ~ ARV.Asteroid_2008_HU4.Altitude	Edit
	Value	25	Edit
	Tolerance	0.01	Edit
		OK Apply Cancel	Help

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	Approac	h 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.

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

= 0.4056574909945581 km/s

Therefore ARV approach $\Delta V = 0.406$ 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 7.2Initial 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.

Control Variable	Current Value	Last Value	Difference
BurnDuration	277.7695459128955	277.7695459128955	0
ARV.BHT_20K_Hall_Effect1.ThrustDir	e -0.00195771946442989	-0.00195771946442989	0
ARV.BHT_20K_Hall_Effect1.ThrustDir	e -0.4047622072971175	-0.4047622072971175	0
ARV.BHT_20K_Hall_Effect1.ThrustDir	e 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.Altitud	e 25	24.97322967479275	-0.02677032520724865

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.

hormations	Orbit Attitude	Ballistic/Mass Tanks	Power Syst	em SPICE Actuators	Visualization	
Ground Station	Enoch Format	UTCC		Elements		
🗃 Hardware	Epoch ronnac	UTCGregorian	~	SMA	160287785.6175935	km
SolarPowerSystem	Epoch	08 Dec 2017 22:39:51.	566	ECC	0.05555352157796046	
BHT 20K Hall Effect1	Coordinate System	ⁿ SunEcliptic	~	INC	1 391470451432671	deg
BHT_20K_Hall_Effect2	State Type	Keplerian	~	PAAN	245 2040250427422	dea
Durns				NAAN	215.2919269127133	deg
FiniteBurn				AOP	350.5524977034908	deg
VearFarth				TA	176.4185547257664	deg
boundary value solvers boundary value solvers Output Output SolarSystemView SolarSystemView Interfaces Scripts ARM 2008 HU4 Return Variables/Arrays/Strings (x) BurnDuration Coordinate Systems - 24. EarthMJ2000Ec M. EarthFind						
- A. EarthICRF - A. SunEcliptic - A. EarthInertial	ОК	Apply	Cancel			

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.

E-Compared Spacecraft	Space	craft - ARV_2	2008_HU4					
Formations Cound Station	Orbit	Attitude	Ballistic/Ma	ss Tanks	Power System	SPICE	Actuators	Visualization
Hardware	Spher	rical						
Xenon_Tank	Dry	Mass		505500	kg			
BHT_20K_Hall_Effect2	Coe	fficient of Dr	ag [2.2				
E-O Burns	Coe	fficient of Re	flectivity	1.8				
- Dropagators	Drag	g Area		15	m^2			
DeepSpace	SRP	Area		535.5	m^2			
Solvers								
E	SPAD	File						
Optimizers Output	SPAI	D SRP File	[
	SPA	D SRP Scale F	actor 1		1			
	5.7.4							
ARM 2008 HU4 Return Fir								
Variables/Arrays/Strings (x) BurnDuration								
Coordinate Systems								
∰- SunEcliptic ∰- EarthInertial	d	ОК	A	pply	Cancel			
Event Locators	<							>.

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 m².

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.

Resources Mission Output	PropSetup - DeepSpace	2		
Spacecraft Spacecraft Formations Ground Station Hardware DeepBurn OrbitalBurn Propagators NearEarth WeepSpace SolarSystem Solvers Cutput KertNiew	Integrator Type Initial Step Size Accuracy Min Step Size Max Step Size Max Step Attempts	RungeKutta89 60 1e-007 0 43200 50 Stop If Accuracy Is Violated	sec sec sec	Force Model Error Control RSSStep Central Body Sun Primary Body Gravity Model Degree Order Potential File
Control Systems Subsystems Subsystems Control Systems Event Locators Functions	ОК	Apply Cancel		Drag Atmosphere Model None Setup Point Masses Earth Jupiter Luna Mars Neptune Saturn Su Select Use Solar Radiation Pressure SRP Model Spherical ~ Relativistic Correction Help

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.

Resources Mission Output	Vary Burn Duration	
→ 25 Propagate in Deep Space	Solver DefaultDC V	
Vary Burn Duration	Variable SturnDuration Edit	
🥥 Vary Thruster1.3	Initial Value Perturbation Lower Upper	Max Step
Equation1	3400 .00001 -10e300 10e300 .0	1
Turn Thrusters On Zim Thrusters On Zim Prop Duration in Days Zim Thrusters Off Zim Propagate to Earth Periapsis Achieve Altitude	Additive Scale Factor 0.0 Multiplicative Scale Factor 1.0	1124
End Target Desired Earth Altitude	d E OK Apply Cancel	нер
	W Vary Thruster1.1	
	Solver DefaultDC ~	
	Variable Setup	
	Variable ARV_2008_HU4.BHT_20K_Hall_Effect1.ThrustD Edit	
	Initial Value Perturbation Lower Upper	Max Step
	1e-005 .00000001 -10e300 .00	001
	Additive Scale Factor 0.0 Multiplicative Scale Factor 1.0	
	OK Apply Cancel	Help

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.

	 ⊘ № 	Achieve Altiti	ıde	
	· 🕪 🕅	Solver Goal	DefaultDC ~ ARV_2008_HU4.Earth.Altitude	Edit
		Value Tolerance	450000	Edit Edit
- 🧏 Propagate to Earth Periapsis - 🎯 Achieve Altitude - 4 End Target Desired Earth Altitude			OK Apply Cancel	Help

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.

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

Table 8.1Return Results

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.

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

Therefore ARV return $\Delta V = 44.639$ 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	Result
--	-----------	-----	--------	--------

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.

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

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	Com	narison	of A	pproach	ΛV
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This Study	$\Delta V = 0.406 \text{ km/s}$
Wagner and Wie	$\Delta V = 0.161 \text{ km/s}$
Brophy and Friedman	$\Delta V = 2.800 \text{ km/s}$

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.

This Study	$\Delta V = 44.639 \text{ m/s}$
Wagner and Wie	$\Delta V = 311.0 \text{ m/s}$
Brophy and Friedman	$\Delta V = 0.170 \text{ m/s}$

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 9.3Comparison 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 9.4Comparison 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.

REFERENCES

- J. R. Brophy and L. Friedman, "Returning an Entire Near-Earth Asteroid in Support of Human Exploration Beyond Low-Earth Orbit," 2012. [Online]. Available: http://www.kiss.caltech.edu/study/asteroid/papers/returning.pdf. [Accessed 9 December 2015].
- [2] S. Wagner and B. Wie, "Low-Thrust Trajectory Optimization for Asteroid Exploration, Redirect, and Deflection Missions," 2011. [Online]. Available: https://www.adrc.iastate.edu/files/2011/07/AAS-14-283.pdf. [Accessed 7 October 2015].
- M. Gates, S. Stich, M. McDonald, B. Muirhead, D. Mazanek, P. Abell and P. Lopez, "The Asteroid Redirect Mission and Sustainable Human Exploration," 2014. [Online]. Available: https://www.nasa.gov/sites/default/files/files/IAC-14-A5_1_1-Gates.pdf. [Accessed 7 October 2015].
- [4] D. A. Vallado, Fundamentals of Astrodynamics and Applications, Hawthorne, CA: Microcosm Press, 2013.
- [5] General Mission Analysis Tool Development Team, "General Mission Analysis Tool (GMAT) User Guide," 2 November 2015. [Online]. Available: http://gmat.sourceforge.net/doc/nightly/help-letter.pdf. [Accessed 9 April 2016].
- [6] G. P. Sutton and O. Biblarz, Rocket Propulsion Elements, Hoboken, New Jersey: John Wiley & Sons, 2010.
- [7] National Aeronautics and Space Administration, "HORIZONS System," Jet Propulsion Laboratory, 9 April 2016. [Online]. Available: http://ssd.jpl.nasa.gov/?horizons. [Accessed 11 December 2015].
- [8] D. D. Mazanek, R. G. Merrill, J. R. Brophy and R. P. Mueller, "Asteroid Redirect Mission Concept: A Bold Approach for Utilizing Space Resources," 2014. [Online]. Available: https://www.nasa.gov/sites/default/files/files/IAC-14-D3-Mazanek.pdf. [Accessed 7 October 2015].
- [9] National Aeronautics and Space Administration, "Asteroid Redirect Mission,"
 2013. [Online]. Available: https://www.nasa.gov/sites/default/files/files/AsteroidRedirectMission_FS_508_2
 .pdf. [Accessed 7 October 2015].

- [10] J. Szabo, B. Pote, V. Hruby, L. Byrne, R. Tedrake, G. Kolencik, H. Kamhawi and T. W. Haag, "A Commercial One Newton Hall Effect Thruster for High Power In-Space Missions," 2011. [Online]. Available: http://enu.kz/repository/2011/AIAA-2011-6152.pdf. [Accessed 17 March 2016].
- Busek Space Propulsion and Systems, "Busek High Power Hall Thrusters," 2013.
 [Online]. Available: http://www.busek.com/index_htm_files/70008511B.pdf.
 [Accessed 17 March 2016].
- [12] D. C. Folta, N. Bosanac, D. Guzzetti and K. C. Howell, "An Earth-Moon System Trajectory Design Reference Catalog," 2014. [Online]. Available: https://engineering.purdue.edu/people/kathleen.howell.1/Publications/Conference s/2014_IAA_FolBosGuzHow.pdf. [Accessed 2 November 2015].
- [13] D. Chavali and S. Hughes, "GMAT Central," 29 March 2016. [Online]. Available: http://gmatcentral.org/display/GW/GMAT+Wiki+Home. [Accessed 2 November 2015].
- [14] W. A. Hargus Jr. and M. R. Nakles, "Hall Effect Thruster Ground Testing Challenges," Air Force Research Laboratory, Edwards AFB, 2009.

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 22:39:35.81927949.4441645715TAI Epoch:14 Jul 2017 22:40:11.81927949.4445812382TT Epoch:14 Jul 2017 22:40:44.00327949.4449537382TDB Epoch:14 Jul 2017 22:40:44.00327949.4449537350

 Cartesian State
 INFIRING

 X = 0.6886413842205 km
 SMA = -0.000000000000 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

 Cartesian State Keplerian State -----

 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 = 90.00003488026 deg
 Beta Angle = -68.153846127858 deg

 RAV = -16.637292209514 deg
 Periapsis Altitude = 24.995974265245 km

 DECV = 1.6474449177561 deg
 VelPeriapsis = 3.1768558295979 km/s

 Spherical State Other Orbit Data

 Planetodetic Properties
 Hyperbolic Parameters

 LST
 = 340.2763393323 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
 =
 5500.000000000 kg

 Total mass
 =
 14615.133124320 kg

 Tank masses: 9115.1331243202 kg Xenon Tank:

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 19:52:15.96131154.3279625169TAI Epoch:23 Apr 2026 19:52:51.96131154.3283791836TT Epoch:23 Apr 2026 19:53:24.14531154.3287516836TDB Epoch:23 Apr 2026 19:53:24.14731154.3287517017 Cartesian State Keplerian State _____ -----HA = 359.99999947021 deg

 AVIAG =
 456378.18517154 km
 Mean Motion
 =
 2.869735e-006 deg/sec

 RA =
 -108.45446470627 deg
 Orbit Energy
 =
 0.546877135 km²2/s²2

 DEC =
 0.1682667351771 deg
 C3
 =
 1.093754271 km²2/s²2

 VMAG =
 1.6853940512783 km/s
 Semilatus Rectum
 =
 1484276.7753651 km

 AZI =
 62.796633943836 deg
 Angular Momentum
 =
 769177.078421 km²2/s

 VFPA =
 90.00000000046 deg
 Beta Angle
 =
 -5.3561937107381 deg

 RAV =
 -18.367975043973 deg
 Periapsis Altitude =
 450000.04887154 km

 DECV =
 27.203239054259 deg
 VelPeriapsis

 Spherical State Other Orbit Data 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

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