Investigating Various Propulsion Systems for an External Attachment for a Controlled-Manual De-orbit of the Hubble Space Telescope

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This reports explains the results for a proposed senior project. This project concerns the Hubble Space Telescope, and exploring the possibility of having an external propulsion attachment for a manual de-orbit. The Hubble Space Telescope was proposed to return to Earth via the Space Shuttle. Although, through the current U.S. Space Administration, the Space Shuttle has been retired before the Hubble Space Telescope was retrieved. By completing this project, the results could provide insight to what type of propulsion would best de-orbit the Hubble upon its retirement. Different propulsion systems were considered to attempt to determine an optimal attachment, varying different specific impulse values, propulsion burn times, preformed ΔV, and the direction of the thrust impulse. From the calculated results and after conducting a feasibility analysis, it was concluded that a ≥ 450 seconds Isp propulsion system burning for a maximum of 10 minutes proved to be an optimal choice from the simulated cases.

Nomenclature

A	=	cross-sectional area, [km^2]
C_D	=	coefficient of drag, []
D	=	drag, [kgkm/s^2]
F	=	thrust, [N]
H	=	scale height, [km]
Isp	=	specific impulse, [s]
J_2	=	second zonal harmonic value for Earth, [1.08263×10^{-3}]
P	=	perturbations acceleration (for J_2), [km/s^2]
R	=	radius of Earth, [6378 km]
T	=	orbital period, [s]
a	=	acceleration, [km/s^2]
dm	=	mass flow rate, [kg/s]
e	=	eccentricity, []
g	=	acceleration due to gravity, [9.81 m/s^2]
h	=	specific angular momentum, [km^2/s]
i	=	inclination, [deg]
m	=	mass, [kg]
r	=	radial distance, [km]
v	=	velocity, [km/s]
y	=	state vector
z	=	altitude, [km]
ΔV	=	change in velocity, [km/s]
α	=	semi-major axis, [km]
θ	=	true anomaly, [deg]
μ	=	gravitational parameter of Earth [398600 km^3/s^2]
ρ	=	atmospheric density, [kg/km^3]
ω	=	argument of perigee, [deg]

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Subscripts

0	=	initial, sea level
P	=	perturbations
$deorbit$	=	de-orbit
f	=	final
h	=	normal, specific angular momentum direction
pg	=	perigee
r	=	radial
x	=	x-coordinate
y	=	y-coordinate
z	=	z-coordinate
$\perp$	=	transverse

I. Introduction

The Hubble Space Telescope (HST) has been a crucial and extremely valuable tool for the science community for over two decades. This telescope alone has made hundreds of discoveries and breakthroughs about our solar system and even the universe. These discoveries have been able to provide humanity a better understand of the universe they live in. It has also helped answer some of the most compelling astronomical questions and uncovered mysteries that were never known to have been. Some of these discoveries and breakthroughs by the Hubble Space Telescope include providing detailed images of our planets in our solar system and other distant galaxies many light-years away, provide a better understanding of the formation of stars and planets, and even provided a better estimation of how old the universe is. The Hubble Space Telescope truly is a magnificent tool.

There is a long history of telescopes before the HST was even an idea. Galileo in the 1600's was the first person in recorded history to have observed the skies with a telescope[1]. Although his telescope was much smaller and primitive to today's standard, Galileo scanned the skies to simply learn what was out there. From that point on, telescopes kept improving by becoming more powerful and providing clearer images. Although, the Earth's atmosphere presented to be a limit to these ground-based systems. The idea of an on-orbit telescope was thought to be a potential solution, and the first approved program started in 1969 known as the Large Space Telescope project[1]. The United States then approved funding for their most complex and sophisticated space telescope, which they named after the astronomer Edwin Hubble, in 1977[2]. The HST launched aboard the Shuttle Discovery and was set into orbit on April 25, 1990. The telescope has continued its service even to this day, having had five previous service and maintenance missions[3].

The HST is currently still in orbit about 570 km above Earth's surface[3]. The Hubble page on the NASA website also provides daily activity reports summarizing what the HST is currently doing as well as the vitals of a few of the currently installed instruments and systems. It is only a matter of time when the HST becomes too old and outdated for continued use and therefore should be retired. There are already plans for a replacement in the coming future: the James Webb Space Telescope. With the retirement of the HST and its possible replacement, there will no longer be a need for the HST. It was originally proposed that the HST was to return to Earth via the Space Shuttle[3]. Unfortunately, through the current U.S. Space Administration, the Space Shuttle themselves have already been retired and therefore the HST has no way of returning to Earth by the original plan.

With the current situation of the HST, it has no other option than continue its services until its been decided what to do next. There could possibly be a next generation shuttle to retrieve the HST, but that is unknown when it is expected to happen. Another possible method of retiring the HST is to bring it into the Earth's atmosphere to re-enter. This method was rejected in an official trade study[4], but for retirement without the Space Shuttle, there aren't many other known alternatives than de-orbit. This senior project was performed to show the possibility and feasibility of having an external propulsion attachment connect to the HST for a manual de-orbit. The propulsion systems analyzed varied primarily in specific impulse, which in turn will determine the type of system (solid motor, bi-propellant/hybrid engines, or continuous thrusts), propulsion burn time, the amount of preformed ΔV, and the direction of the thrust impulse. Also included in this project is determining how much time it will take to de-orbit the HST. For the proposes and boundaries of this project, it is assumed that the attachment had successfully rendezvoused and attached with the HST after its launch.

II. Analysis

The following is the set of processes and equations used to simulate the maneuver and to acquire data in completing this project. The main sections of calculations contain propagating the orbit of the HST including perturbations and calculating the effects and properties of the theoretical propulsion system attachment.

A. Orbit Propagation

The first set of calculations needed for this project was to determine the initial orbit of the HST. The approach taken to determine the HST orbit required the six classical orbital elements (COE) of the orbit to calculate a position and velocity vector. With these two vectors, the orbit was propagated by time through a set of ordinary differential equations. The orbital elements for the HST were taken from the Heavens Above website[5] in the form of a two line element (TLE). Although, the TLE by itself doesn't provide all six of the required COE, but can help determine the remaining ones. The first missing COE is the true anomaly. The TLE provides a mean anomaly instead, which can be used to calculate the true anomaly. The method of determining the true anomaly from the mean anomaly used Kepler's equation and Newton's method of iteration[6]. The second and last COE needed was the specific angular momentum. This value was calculated with the following two equations[6]

$$\alpha = \left(\frac{T\sqrt{\mu}}{2\pi}\right)^{\frac{2}{3}} \tag{1}$$

$$h = \sqrt{\alpha\mu\left(1 - e^2\right)} \tag{2}$$

where α is the semi-major axis of the orbit [km], T is the orbital period [s], μ is the Earth's gravitational parameter [398600 km^3/s^2], h is the specific angular momentum [km^2/s], and e is the eccentricity of the orbit. Now that all of the six classical COE are obtained, the position and velocity vectors can be determined in the Earth Centered Inertial frame (ECI)[6].

With the position and velocity vectors determined, the orbital accelerations can be calculated. A simplified orbital acceleration expression exists for a perfect spherical Earth, but in reality there are perturbations that should be included. The primary perturbations of concern for this project are atmospheric drag and the oblateness of the Earth. With the effects of perturbations, the orbital acceleration was determined to be[6]

$$\vec{a} = -\frac{\mu}{r^3}\vec{r} + \vec{a}_P \tag{3}$$

where $\vec{a}$ is the acceleration vector [km/s^2], $\vec{r}$ is the position vector (while r is its magnitude) [km], and $\vec{a}_P$ is the accelerations vector in ECI due to the perturbations [km/s^2]. Each of these perturbations will be discussed separately in the following.

The first perturbation calculated was from atmospheric drag. The magnitude of the drag force is calculated by[6]

$$D = \frac{1}{2}\rho v^2 A C_D \tag{4}$$

where D is the drag force [$kgkm/s^2$], ρ is the atmospheric density [kg/km^3], v is the magnitude of the relative velocity to the rotating atmosphere [km/s], A is the cross-sectional area of the HST [km^2], and C_D is the coefficient of drag. Please note that for the velocity, the orbital velocity was assumed to be the same as the relative velocity of the HST to the rotating atmosphere. Also, there are a couple unorthodox units for the drag force and atmospheric density. These values have these units to simplify the final calculation of orbital acceleration, which involve kilometers instead of the standard meter. The cross-sectional area of the HST was determined to be 63.94 m^2 from its geometry[7]; two cylinder shapes on top of one another will project two rectangular shapes and a set of two rectangular solar panels make up the cross-sectional area. A C_D of 2.2 was used as a nominal value. The other two values, v and ρ, change with time and depend on altitude. The velocity was calculated through an ordinary

differential equation (ODE) solver, but that will be discussed later. The atmospheric density was calculated with the expression[8]

$$\rho = \rho_0 e^{\frac{-(z-z_0)}{H}} \tag{5}$$

where ρ_0 is a reference density [kg/m^3], z is the altitude [km], z_0 is a reference altitude [km], and H is a scale height [km]. The ρ_0, z_0, and H depend on the current z, and these values are provided in Table 7-4 in Vallado[8]. Note that "*e*" here is the mathematical exponential expression, not eccentricity. With these values, the drag force was then calculated. Drag force always opposes the direction of motion, so the drag vector was simply the opposite direction of the velocity vector and shown as

$$\vec{D} = -D\frac{\vec{v}}{v}. \tag{6}$$

According to Newton's 2nd Law, force is mass multiplied by acceleration. So the acceleration due to drag is simply the drag force divided by the mass of the HST. This acceleration term is then included in the $\vec{a}_P$ term for Eq. 3.

The next perturbation to account for was for the oblateness of the Earth. This is more commonly known as the second zonal harmonic, or J_2, perturbation. This perturbation can be expressed as the following acceleration vector[6]

$$\vec{P} = p_r\hat{u}_r + p_\perp\hat{u}_\perp + p_h\hat{h} \tag{7}$$

where $\vec{P}$ is the perturbation acceleration vector in ECI [km/s^2], $\hat{u}_r$, $\hat{u}_\perp$, and $\hat{h}$ are the radial, transverse, and normal unit vectors attached to the HST, and p_r, $p_\perp$, and p_h are the magnitudes of each of these accelerations for each respective vector. The radial and transverse unit vectors are simply the same direction as the position and velocity vectors, respectively, while the normal unit vector completes a right-hand triad. The magnitudes of these vectors are determined by[6]

$$\begin{aligned} p_r &= -\frac{3\mu}{2r^2}J_2\left(\frac{R}{r}\right)^2\left[1-3\sin^2 i\sin^2(\omega+\theta)\right] \\ p_\perp &= -\frac{3\mu}{2r^2}J_2\left(\frac{R}{r}\right)^2\sin^2 i\sin\left[2(\omega+\theta)\right] \\ p_h &= -\frac{3\mu}{2r^2}J_2\left(\frac{R}{r}\right)^2\sin 2i\sin(\omega+\theta) \end{aligned} \tag{8}$$

where J_2 is the second zonal harmonic value for Earth (1.08263×10^{-3}), R is the radius of the Earth (6378 km), *i* is the inclination of the orbit [deg], ω is the argument of perigee [deg], and θ is the true anomaly [deg]. The latter three values are three of the classical COE determined earlier, but need to be updated for every new position and velocity vector. With the completed $\vec{P}$ vector, it was also added into the $\vec{a}_P$ term for Eq. 3 above.

Now with all the perturbations included, the orbit was then propagated by means of the differential equations of orbital motion. As stated above, an initial position and velocity vector were determined through the COE. These two vectors become the state vector for the following system of equations

$$y = \begin{bmatrix} r_x \\ r_y \\ r_z \\ v_x \\ v_y \\ v_z \end{bmatrix} \qquad \dot{y} = \begin{bmatrix} v_x \\ v_y \\ v_z \\ a_x \\ a_y \\ a_z \end{bmatrix} \tag{9}$$

where y is the state vector, $r_{x,y,z}$ are the components of the position vector, $v_{x,y,z}$ are the components of the velocity, $\dot{y}$ is the rates vector, and $a_{x,y,z}$ are the components of the acceleration vector determined in Eq. 3. The rates vector is made as a MATLAB[9] function to be the evaluated through an ODE solver. This set of differential equations are evaluated over a time array to output the positions $r_{x,y,z}$ for analysis. Note that the velocity outputs are also used to calculate the drag force as mentioned earlier.

B. Propulsion System Characteristics

The next major part of this project was to determine the propulsion system characteristics. The main concept of de-orbiting the HST in this project was to perform a propulsion burn to decrease the velocity, and so this burn was treated as a ΔV maneuver. An appropriate ΔV value was approximated with the expression[10]

$$\Delta V_{deorbit} \approx v\left[\frac{z_0 - z_{pg}}{4\left(R + z_{pg}\right)}\right] \tag{10}$$

where $\Delta V_{deorbit}$ is the change in velocity to de-orbit [km/s], z_0 is the altitude of the initial orbit [km], and z_{pg} is the altitude of the desired perigee at the end of the burn [km]. Note also that this approximation is for circular orbits, which does apply for the HST. The z_{pg} value doesn't necessarily have to be 0 km[10], but it is projected to be anywhere from 50 to 100 km. An object would be in the atmosphere by 100 km, therefore re-entry by drag and gravity alone is certain. The z_{pg} value was decided to be 50 km. From the ΔV value, the propellant mass for the propulsion system was determined. The mass was calculated by the equation[6]

$$m_f = m \cdot e^{\left(\frac{1000 \cdot \Delta V}{Isp \cdot g_0}\right)} \tag{11}$$

where m_f is the final mass (with propellant mass) [kg], m is the dry mass [kg], Isp is specific impulse [s], and g_0 is the acceleration of gravity at sea level [9.81 m/s^2]. As mentioned in the introduction, the Isp was varied to determine what effects it would have from the propulsion system and the type. From the final mass and the dry mass, the propellant mass was determined by the subtraction of the two values. With the propellant mass, a mass flow rate can also be calculated. This value can vary just based on how long the propulsion system would burn. The mass flow rate is calculated by dividing the propellant mass by a specified burn time [kg/s]. Next, the magnitude for the produced thrust can be calculated by[11]

$$F = Isp \cdot dm \cdot g_0 \tag{12}$$

where F is the thrust produced [N] and dm is the mass flow rate [kg/s]. As stated before, the propulsion burn is to decrease the velocity, so the direction of the thrust would be

$$\vec{F} = -F\frac{\vec{v}}{v}. \tag{13}$$

Similar to the drag term from the previous section, dividing this force by the mass of the HST would yield an acceleration; this was added as another term to Eq. 3. Although, this value was only added when the propulsion system was used. Note that this acceleration from thrust should also be in km/s^2 to be consistent with the other acceleration terms in Eq. 3.

A radial propulsion burn was also simulated. This option was considered to provide another aspect to consider in the feasibility analysis and to compare these results with the tangential burn simulation. This radial propulsion burn was characterized by having one difference in the simulation. The radial burn simulation used the same process described in this section, except that Eq. 13 used the position vector ($\vec{r}$) and its magnitude (r) instead of velocity.

III. Results and Discussion

This section describes the results from the all the simulations conducted for this project. These include the results

from a tangential ΔV maneuver and a radial ΔV maneuvers. Each respective simulation will have different variables, which will be properly described in the appropriate section. The main purpose of this section is to display the results and provide some initial analysis. The more valuable feasibility analysis section later in the report provides more analysis and a deeper explanation of the results from all of the simulations.

A. Propulsion System Trade Study

There are differing types of propulsion systems available for spacecraft impulse maneuvers, each having their own capabilities. Therefore, a simple trade study was conducted to compare the systems to validate which type to truly consider and use for the de-orbit maneuver. A score of 1-3 was utilized for this trade to simplify the process, showing if desired (score of 3) or if not (1) while a score of 2 shows that the category can go either way. A deeper explanation for the scoring will follow the table below. A weighted system for each metric of the trade study was not considered; as stated this was a simplified trade and the purpose was just to validate an appropriate system. The following table shows the results and scores for each type of propulsion system and categories considered for the trade.

Table 1. Propulsion System Type Trade Study

	Multi-Use	Simplicity	Power Required	Decay Time	Controllable	TOTAL
Cold Gas	3	3	3	3	3	15
Solid	1	3	3	3	1	11
Liquid	3	1	2	3	3	12
Electric	2	1	1	1	3	8

As shown in the table, four primary types of propulsion systems were considered. The apparent winner from this trade study seems to be cold gas propulsion, receiving the high score in all categories. Although this system did have the highest score, later it will be described how this propulsion type was infeasible for this maneuver in the feasibility analysis section due to the calculated propellant mass. An added category of mass in the trade study was not believed to be useful. The resulting scoring from a mass category would still have led to cold gas propulsion having the highest score. Also, there wasn't enough knowledge of how massive each of the other systems (solid, liquid and electric) would compare due to their varying components. Therefore, mass was thought to be more of a feasibility criterion for the feasibility analysis later in this report.

A description of the trade study scoring will be explained as follows. The first category in the trade study was multi-use; being straightforward, this category suggests if the propulsion could be used more than once. This category was considered for the ability to possibly use multiple burn de-orbit scenarios. Solid propellants are single-use, therefore the low score was given while liquid propellants can be turned off and on, resulting in the high score. Electric propulsion (EP) was given a middle score because it can turn off and on, but the purpose for EP is for continuous thrust. The simplicity category was also straightforward. This category was considered because this theoretical attachment does a rather simple task of slowing down an object to fall into the atmosphere. An overly complex system to complete this task was thought to be unnecessary. Solid propellant systems received the high score for its general simplicity while both liquid and EP are rather complex, having multiple parts and components just to make each respective system work. The power category implies the use of electric power that each system uses. A solid propellant system wouldn't require a lot of electricity to start, thus received a high score. A liquid propellant system would require more electricity in comparison to a solid propellant, having other components to control (pumps, valve controls, etc.), so this system received the middle score. The EP system received the low score because it requires an extreme amount of power to work. Also, despite the very low mass of the propellant used by various EP systems, the mass of the other required components, such as batteries and large solar panels, might upset the mass savings from the propellant.

The next category is decay time, referring to how much time it would take to de-orbit. This category could potentially be advantageous for long or short decay times, but that truly depends on the mission. Since there isn't any real set mission for the HST de-orbit, this is more of a result for this project. This category was considered in the hopes of having the HST de-orbit in a relatively reasonable time (on the order of days). With that decided, solid and liquid propellant systems could de-orbit the HST in the target time range, and so these systems received the high score. Electric propulsion systems on the other hand take a long time to complete the maneuver, thus receiving the low score. The controllable category refers to the ability to control the impulse, in magnitude (throttle), in one way or another. Solids are one time use and once they burn, they continue until burnout. Liquid and EP systems can be controlled in the aforementioned sense, thus justifying the scores above. With all the scores added, the result shows

that liquid propellant systems would be the most effective propulsion system to use. Since the solid propellant system scored relatively well, this system was also considered in analysis for this project.

The cold gas option was considered up until calculated mass and mass flow rate values were determined. Their results justified that this propulsion system in particular is not ideal. Again, a further explanation will be discussed later in the feasibility analysis section. The EP option was completely removed from this project due to having the lowest overall score and a couple of practicality reasons. To accurately simulate an EP maneuver, the angle in which the thrust acts on must always be aligned, and thus corrected through time. This will make the computations significantly more difficult, unless an assumption(s) is made. Optimization of EP maneuvers also makes the simulation for this project overly complicated. Other assumptions and lack of optimization also led to the decision of excluding EP for consideration for this project.

B. Simulation Fidelity Check

In order to have any set of believable data, a check of the MATLAB[9] code produced for the simulations was conducted for fidelity. It is known that the International Space Station (ISS) drops about 100 meters in altitude per day due to atmospheric drag[12]. Therefore, the orbital and physical information for the ISS was implemented into the simulation. A simple orbit propagation of a day was applied to the ISS to the MATLAB[9] simulation. With only the acceleration terms described in Eq. 3 above, without the thrust term, the resulting altitude drop was about 60 meters. This result was thought to be within an acceptable range of the true altitude drop and therefore the perturbation calculations seemed to be verified. Another check implemented for the code was for an actual de-orbit of the ISS. A similar scenario intended for the HST for this project was conducted for the ISS, both with a tangential and radial propulsion burn. The following figures are a plot of the ISS altitude experienced in the decaying orbit after a single de-orbit propulsion burn for each burn type.

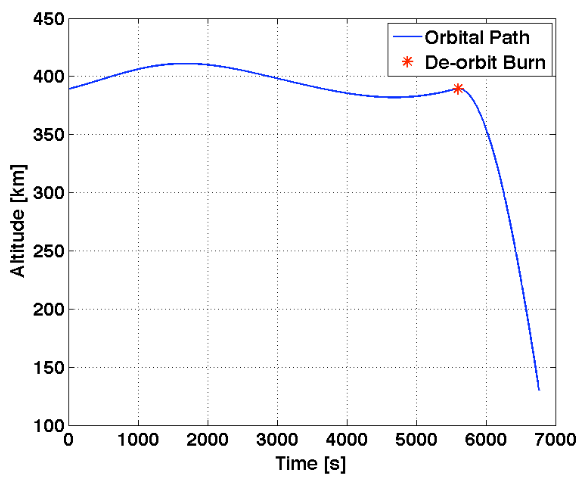

Figure 1. ISS Tangential Burn Simulation Fidelity Check

Figure 1 shows the tangential burn simulation. As shown in this figure, after the burn the altitude immediately drops signifying the re-entry. This method shows that this burn type provides a very quick re-entry. The other method (radial propulsion burn) was also checked if the simulation would work. The drag force was also plotted to see if the drag was being calculated properly. This is shown in the following figure.

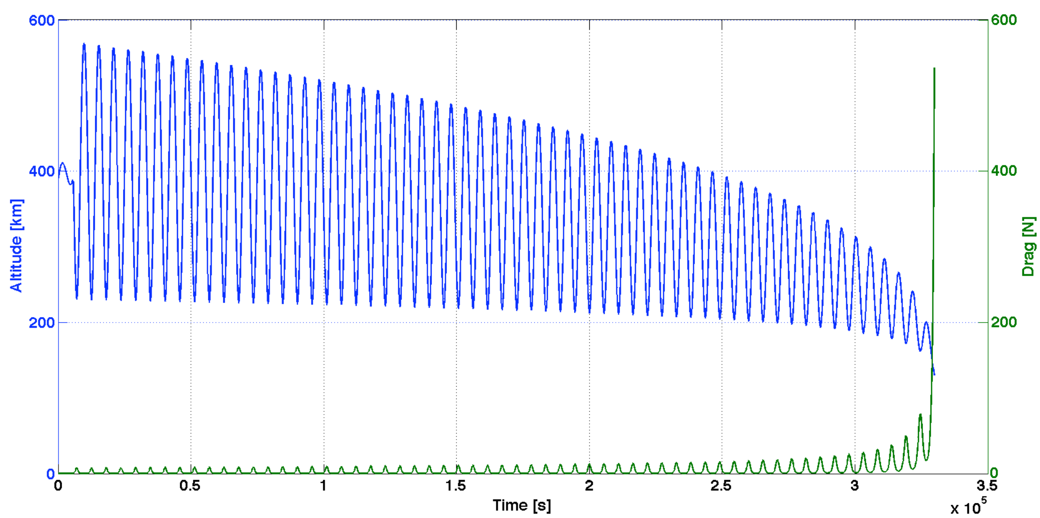

Figure 2. ISS Radial Burn Simulation Fidelity Check. *Specifically, this fidelity check simulation used an Isp of 450 seconds and a 0.2 km/s ΔV maneuver.*

As shown in this figure, the drag peaks at every perigee of the decaying orbit. Also, there is a slow yet gradual decrease in the perigee altitude, but more noticeably is the decrease in the apogee altitude after each period. The orbit decays, as shown by the altitude decrease, until it eventually drops rapidly signifying a re-entry. Due to the shear size and mass of the ISS, the aforementioned de-orbit scenario decay time in particular only took just under 4 days to de-orbit. With the ISS drop test result and an actual sample de-orbit, the simulation fidelity proved to be satisfactory and therefore the project continued for the HST simulations.

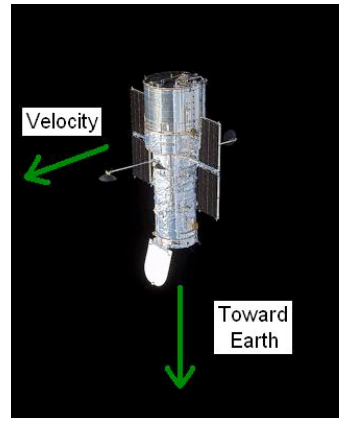

Figure 3. Proposed HST Orientation for Maneuver

C. Simulation Results

The following section explains the results for both the tangential and radial propulsion burn simulations. To also clarify, there were a few assumptions made for each of the simulations conducted. It was assumed that this theoretical attachment had successfully launched and rendezvoused with the HST. Also, the orientation for the HST would be such that the most cross-sectional area would face the ram direction as shown in Fig. 3. Also, the attachment connected to the HST such that the thrust vector passed through the center of mass of the HST as to not produce any unnecessary torques while thrusting. The main results taken to use for the feasibility analysis are the calculated propellant mass and the decay/re-entry time. Plots also were provided when necessary. There are also some results that were specific to the simulation, being either for the tangential burn or the radial burn.

1) Tangential Burn

For the tangential burn scenario, the calculated propellant mass, the produced thrust, the resulting altitude over time, and the re-entry time were thought to be the main results. To view these results, please refer to Table 2 and Fig. 7 in the Appendix. To help illustrate the simulation, an altitude plot for one of the scenarios was also generated.

For this simulation, there were two values that were varied to see their results. These variables were the Isp values and burn time for the propulsion system. The varied Isp categorizes what type of propulsion system was used. A cold gas propulsion system was described to be the 30-70 seconds Isp values, 250-350 seconds Isp values are solid propulsion systems, and 400-450 seconds Isp values are the liquid propulsion systems. The burn time was varied to see if it affected the re-entry time. The process of this simulation allowed for the HST to orbit once, just to simply show the original orbit and then the propulsion burn, illustrated as the red point in the plot, signifies the de-orbit maneuver.

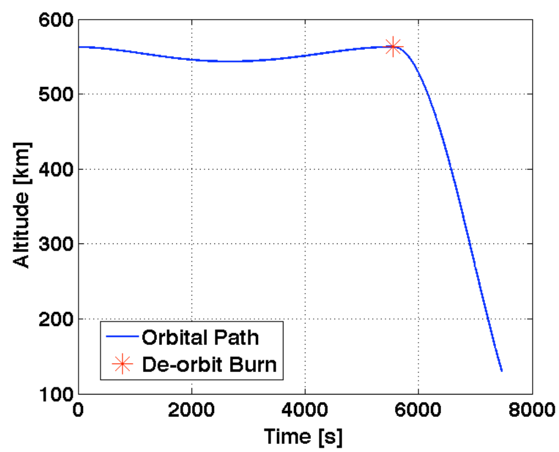

Figure 4. Tangential Burn Simulation Result for the HST. *Specifically, this simulation is the result from a propulsion burn of 10 seconds using an Isp of 300 seconds.*

Figure 4 here is like a template for all the other scenarios; each scenario from the tangential burn simulation did successfully de-orbit and all yielded a very similar plot. The propulsion burn caused the altitude to drop dramatically and in a very short amount of time. As Table 2 shows, the re-entry times for each of the varied propulsion systems are all around 30 minutes. This is considerably fast; this re-entry time along with the calculated propellant mass, produced thrust, and propellant mass flow rate will all be further analyzed and discussed in the feasibility analysis section later in this report.

2) Radial Burn

The main results for this simulation are similar to the tangential burn. There was one different variable for this simulation. Specific impulse was still varied to describe the type of propulsion system, but the ΔV was varied instead of the burn time. This simulation initially ran with the calculated ΔV from Eq. 10, but the results weren't satisfactory, so the ΔV was varied. This variable then became an appropriate category for the feasibility analysis

discussed later in this report. The burn time was kept at 5 minutes for each iteration. This burn time was decided on to try and show a significant effect in the resulting orbit. Also, it was proven that the shorter burn times were actually disadvantageous during the tangential burn simulation, so the burn time was increased; this will be further discussed in the feasibility analysis section. The computation time it took to run each iteration was too long, so a third variable was thought to be unreasonable. Please refer to Table 3 and Fig. 8 in the Appendix to view the results for this simulation.

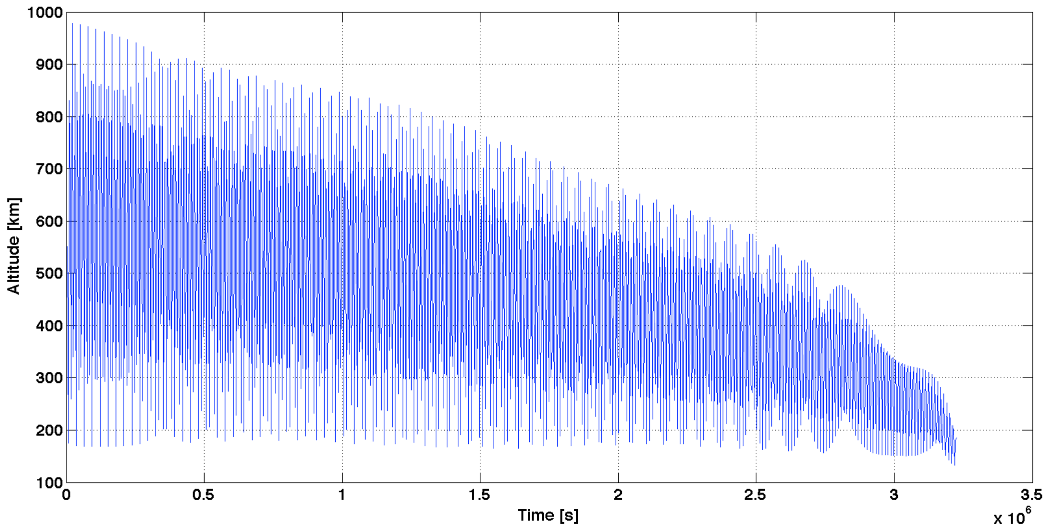

Figure 5. Radial Burn Simulation with an Apparent De-orbit. *Specifically, this simulation used an Isp of 400 seconds and a 0.3 km/s ΔV maneuver. Other radial burn simulations that did successfully de-orbit had a similar altitude plot.*

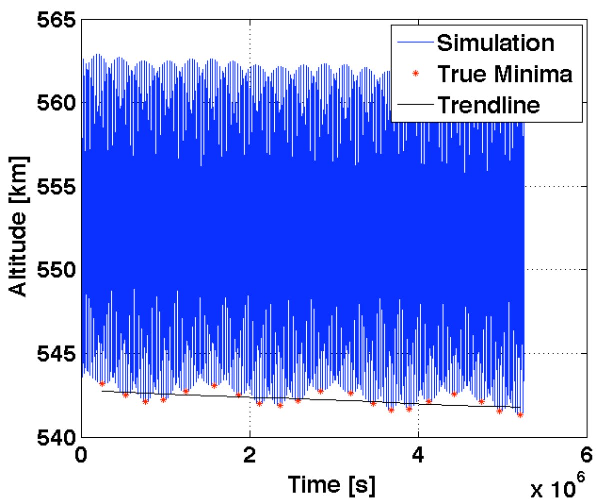

Figure 6. Radial Burn Simulation, Without De-orbit. *Specifically, this simulation used an Isp of 350 seconds and a 0.2 km/s ΔV maneuver. Since this simulation didn't de-orbit, the trend line estimation method is also shown.*

This simulation proved to be significantly more difficult to deal with and obtain data. In this simulation, the HST was again allowed to orbit one period before the propulsion burn was initiated and then allowed to coast until de-orbit. Although the process was the same, the data acquisition time was very long. To try and reduce the amount of time taken for the computations, multiple computers were used in obtaining data and also a forced time step through the ODE solver was implemented. This did help obtain the data relatively quicker, but at the cost of lower fidelity of the data. This can be seen in either Fig. 5 or 6. The forced time step allowed the ODE solver to only save data at the specified times. Without the time step, the ODE solver would've recorded data at even fractional seconds, taking more time to calculate as well as store more data for each scenario, but in return would provide a smooth curve for a plot over time. Since a time step was used, the recorded data did suffer from lower fidelity. As made apparent in the figure, there are multiple magnitudes in the local minima signifying the perigees. With higher fidelity data, this altitude plot over time would have looked more like Fig. 2 shown above with the ISS simulation check with the true perigees. Also, as shown in Table 3, most of the scenarios didn't have an apparent de-orbit. With a radial burn for the de-orbit maneuver, the actual de-orbit didn't occur immediately. Since the decay time was unknown, a reasonable simulation time was decided on. Various simulation times were experimented with, but as mentioned

before, the computation time was very long by itself. Adding more time in completing the computations by increasing the simulation time was not worth the extra delay, so a limit to 2 months simulation time was applied. With the varying magnitudes in perigees and for the scenarios without an obvious de-orbit, an estimation method was used to obtain a decay time. This estimation method will be described in the following.

The main idea behind this estimation method was to create a trend line of the minimum data points, technically the perigee points, and see where down in time the altitude will hit 100 km. The 100 km altitude was chosen to truly show that the HST has reached the atmosphere. After the data of each scenario had been saved, an analysis function was created in MATLAB[9] to generate altitude plots like Fig. 6. In the function, the local minima were extracted from the data. The local minimums were extracted several times to try and find the true minima out of all the varying minima. With the true minima found, this data was passed through the MATLAB[9] polyfit function to determine the coefficients for a first-degree polynomial fit, and thus a trend line was created. The true minima and resulting trend line were also illustrated in Fig. 6.

D. Feasibility Analysis

As previously mentioned several times, this section provides a deeper and more detailed analysis of the results from each simulation. The main criteria that will be studied and compared with for each simulation are the estimated propellant mass, the decay time to re-enter, and the varied ΔV (only for radial burn simulation) to perform the maneuver. Before anything else is mentioned, it would like to be noted that any additional mass for the attachment, considering this theoretical attachment itself is a spacecraft, such as structure and required subsystems for proper operation was not included. The additional mass could possibly alter the results for this project, but there wasn't any high enough fidelity mass estimation for a full spacecraft found. Therefore, only the addition of the propellant mass was considered for this project. Further analysis or future continuation of this project could possibly be included to determine any differences.

1) Tangential Propulsion Burn

Between the two methods studied for the de-orbit maneuver, the tangential burn was definitely the more efficient way. For each case simulated, with varied Isp and burn time, the HST did successfully de-orbit from a single burn. Therefore, it was appropriate to truly compare each case and the results recorded in Table 2 in the Appendix.

To start off, the calculated propellant mass will be analyzed. From an initial glance, the required propellant masses for the 30, 50, and 70 seconds Isp cases yielded a very high estimation. These Isp values were used to depict a cold gas propulsion system. To reiterate, the cold gas propulsion system was considered and included for analysis only because the trade study concluded that this system might be the best solution. All the categories included in the trade study were all in favor of the cold gas system. Through the estimated propellant mass alone, the use of this system is definitely infeasible. In order to perform the calculated de-orbit ΔV maneuver, more than 2000 kg of propellant is needed using the highest cold gas Isp considered; even more mass is required for the other two cold gas systems. The required tank size to accommodate that amount of mass would also be unreasonably large. The only real use for cold gas propulsions systems is for attitude control because of their simplicity and the small thrust it can provide[10]. It is this actual magnitude of thrust and low performance[10] that is simply to demanding for this system to provide. As shown by the tabulated thrust results, the use of this system is too unreasonable. Because of this conclusion, it was decided that the cold gas propulsion system would not be considered further.

The other Isp values, depicting either solid propellant systems (250-350 seconds) or liquid propellant systems (400-450 seconds), resulted in a more reasonable propellant mass estimation. Although they seem more reasonable in comparison to the cold gas propulsion systems, the stated masses are still rather massive. This is believed to be because the HST is massive itself. It has a total dry mass of 11,110 kg[3]; to perform the calculated ΔV maneuver on this massive spacecraft, a massive amount of propellant would be required. As initially thought, since the mission for this attachment is practically propulsion driven, most of the mass for this attachment would theoretically be allocated to the propulsion system. The results seem to be consistent with reason and do make sense. The masses don't seem to vary that much, but if mitigating mass is a priority, the 450 second Isp case would be the best choice. If lesser efficient propulsion systems are allowed, any of the solid or liquid propellant systems should be acceptable; this highly depends on a specified mission.

The next criterion for the feasibility analysis was the decay time. As mentioned before hand and as shown in the table, all the cases did successfully de-orbit and consistently only took around 30 minutes to re-enter. For desired missions requiring a quick re-entry, any of the propulsion systems simulated for this project would work. As a quick note, a small trend was noticed while examining the re-entry times. It appears that as the burn time increased, so did the re-entry time. A presumed explanation is that the longer burn time is preventing, or at least slowing down, the actual re-entry. The ΔV maneuver's main purpose is to slow down the HST, so a longer burn is believed to decrease

both velocity and, as a side effect, re-entry time. Although, there aren't any significant differences, as the burn time only affected the re-entry time by a few seconds.

The thrust produced from each case was also thought to be an important result. The calculated produced thrust magnitudes for each case were compared to preexisting motors/engines. From the scenarios completed for this project, most of the propulsion systems considered might not actually be very feasible to complete this maneuver. Both of the varied Isp values and burn times generated very large required thrust magnitudes. Large enough thrust magnitudes to be considered booster propulsion[#] for rocket motors or launch vehicles. For the solid and liquid propellant systems considered (250-450 seconds Isp values), the produced thrusts for burn times 10-60 seconds are considerably too large. The 10 second burn time generated about 172 kN of thrust. The Shuttle orbiter has two 27 kN thrusters as its primary propulsion system[11]. The magnitude of thrust is too large for practical and feasible reasons. Even the 60 second burn time scenarios produced thrusts were comparable to the Shuttle orbiter. Needless to say, a very large and powerful propulsion system would not be the most efficient or economical way to de-orbit the HST. The longer burn times, such as the 150 and 180 second scenarios, might be the relatively most feasible out of the considered because of their lower produced thrust. As made apparent by Fig. 7, the longer the burn time the lower the produced thrust. Based on the decreasing thrust magnitude trend, it would have been better to consider even longer burn times. After further research, it was found that spacecraft orbital maneuvers could typically last up to 10 minutes[11]. Therefore, future analysis or possible continuation of this project could and should examine the consequences of longer burn times. This realization also lead to the decision that a five minute burn time for the radial burn simulation might help increase the feasibility for its use.

For this simulation, the ΔV was not varied. This value was calculated and was kept constant throughout each scenario. This value, by is derivation and definition, is the required ΔV to de-orbit. Therefore, there was no need to vary this value. The ΔV was varied for the radial burn simulation instead because it seemed more necessary in order to lower the decay time.

For the tangential propulsion burn simulation, the feasible choice out of the considered systems to perform the maneuver is a liquid propulsion system capable of an Isp value of 450 seconds burning for at least 180 seconds. The Isp value produced the lowest amount of required propellant mass; mass savings is always a priority in spacecraft design. Also, the 180 second burn time generated a more feasible thrust. Although, theoretically by the equations described in the analysis section, an even higher Isp value would estimate a smaller amount of required propellant mass and as discussed earlier, a longer burn time would also help lower the thrust value to feasible and practical levels with preexisting systems. It is concluded that a liquid propulsion system capable of $\geq$ 450 seconds of Isp and burning for up to a maximum of 10 minutes would provide a possible optimum option. Further analysis would help validate this claim.

2) Radial Propulsion Burn

This radial burn option was also attempted to compare the results with the tangential burn simulation. Along with the unaccounted for mass of the actual attachment, it would also like to be noted that this simulation also suffered from unaccounted cross-sectional area. In order to perform the thrust maneuver towards Earth and have the line of action through the center of mass of the HST, the proposed location for the attachment would be connected "on top" of the HST, relative to the orientation of the HST shown in Fig. 3 above. With this orientation, the attachment itself would also provide cross-sectional area that would affect the drag calculation and ultimately the re-entry. Since the size of the attachment, and thus the cross-sectional area it would provide, is unknown, this area is neglected. Therefore, unaccounted for mass and area for this simulation also means the results aren't as high fidelity either. Future analysis and possible design of an attachment should help generate more accurate results.

As previously explained, the ΔV was varied instead of burn time for this simulation. The initially calculated ΔV defined in Eq. 10 proved to not be enough to cause a de-orbit in the simulated time of 2 months. Therefore the ΔV was varied to see if the effect would help decrease the decay time. The increased ΔV had a direct correlation to the propellant mass to perform the modified maneuver. The increased ΔV significantly increased the propellant mass required as shown in Table 3. The decay times did have a decreasing trend relative to the each of the other times, but this decreased time didn't matter due to how much more mass was added to complete the increased ΔV maneuver. It is understood that the majority of the mass for the attachment would be propellant, but the amounts estimated for this maneuver are extremely high. As a worst case, one of the simulated scenarios estimated about 1600 kg of propellant was required to perform a 0.4 km/s ΔV maneuver. Even though the decay times did decreased as hoped from the increased ΔV, the consequential increased propellant mass wouldn't make the increased ΔV options feasible. Like the tangential burn, the initially calculated ΔV and use of 450 seconds for Isp, provided the least amount of required propellant mass. Unfortunately the decay time from this scenario did suffer significantly and is just too long to even be considered as a viable option.

As determined by the tangential burn simulation, the produced thrust was a significant value to consider. The produced thrust values potentially show if feasible propulsion systems exist to perform the maneuver. Also, from the study conducted for the tangential burn simulation, a longer burn time was necessary to lower the thrust magnitude to more feasible levels. A five minute burn time was held constant for each scenario for this simulation. This burn time did help provide more reasonable thrust magnitudes. For the calculated ΔV, the produced thrusts were about 5.7 kN for each scenario; these thrusts could be more easily produced from existing propulsion systems[10]. Even the thrusts produced up to the 0.3 km/s ΔV option are on par with the best options from the tangential burn simulation. As expected, because of the increased ΔV, the thrust magnitude also increased. The higher ΔV options may be to powerful and considered as booster propulsion. These thrusts may not be feasible options.

From actually comparing the results, the radial burn option has no real benefit for its use at all. The main result to truly prove this claim is that the re-entry times for this simulation were significantly increased. As shown in Table 3 the decay times were estimated to be more on the order of hundreds of years. Even a few of the estimated decay times were negative, meaning the estimation method didn't work. This will be discussed further in the problems encountered section. Out of all the simulations run, only three scenarios had an apparent de-orbit within the simulated time. All other remaining times are completely infeasible in comparison to the tangential burn simulation. The tangential burn simulation consistently de-orbited around 30 minutes, while the radial burn caused a lot variation. Clearly, a radial burn is not the proper or best way to de-orbit an object. The extended decay times are of no benefit in general. If the de-orbit maneuver causes the decay time to increase, then it would've been better to leave the HST alone for its own natural decay and not spend the resources.

Overall, the radial burn simulation doesn't seem to be a feasible option from all variables considered. The additional propellant mass generated from the higher ΔV options proved to make the increased ΔV options infeasible. The produced thrusts are more manageable to produce in comparison to the tangential burn simulation, but the resulting re-entry times just make the radial burn option completely infeasible. Multiple radial burn simulations also proved to not help very much and therefore aren't worth mentioning further. Truth be told, this radial burn simulation was the actual first attempt in completing this project. There was a mistake found in the simulation code late in completing this project. The mistake was corrected, resulting in conducting the study for the tangential burn. It was realize that theoretically this radial propulsion burn shouldn't really cause a de-orbit. It is believed that since the HST is in a low Earth orbit (LEO), the new orbit caused from the radial burn would allow the HST to experience enough drag to slow down and decay into a re-entry. If it weren't for being in LEO and experiencing the drag, the HST should of just continue orbiting. Obviously, this wasn't the goal for this project.

E. Problems Encountered

In completing this project, there were several problems encountered in obtained viable data. There were two primary errors in completing this project, the first error not being as severe as the second. The second error was considered severe because it caused other setbacks. The first error encountered was due to the ODE solver in MATLAB[9]. While running the code for all the simulations that actually experienced de-orbit, MATLAB[9] kicked out of the ODE solver early. To clarify, "early" implies that the simulation did not output altitudes to 100 km. The 100 km limit was thought to be a reasonable altitude to clearly show that it had reached the atmosphere. The error displayed was claiming that the ODE solver could not meet its integration tolerances and each final altitude was about 130 km. Through some research done online, it appeared that the set of differential equations was too stiff and that a stiffer solver would help. All solvers, stiff and non-stiff solvers, provided by MATLAB[9] were tried, but unfortunately each solver arrived with the same result. Although, in a practical sense, the ~130 km final altitude still seems reasonable. Even the altitude plots show the final altitude around ~130 km doesn't really matter. In the plots shown for both tangential and radial burn simulations, the altitude dropped such that it realistically won't be able to gain altitude again through another orbital period (the tangential burn is more apparent of this than the radial burn). Therefore, the simulations were still thought to be successful.

The second and more severe error involved the thrust vector for the simulations. It wasn't until late into the second quarter in completing this project that the thrust vector for the maneuver was discovered to be in the wrong direction. Initially, the thrust vector was pointed towards the center of the Earth. During the first quarter of work, it was researched and understood that a de-orbit propulsion burn was to be conducted tangentially. For unknown reasons, "tangential" was interpreted to be tangential to the velocity vector. With knowledge gained from orbital mechanics classes and further validation in literature, the initial interpretation was incorrect. A main principle behind propulsion maneuvers in orbital maneuvers is to change the velocity by either accelerating or decelerating. Some directional thrust may also help or possibly even be necessary, but primarily the maneuvers are produced in the positive or negative velocity direction, depending on the type of maneuver. For a de-orbiting maneuver, it is best to decelerate (thrust toward the negative velocity direction) and allow the change in velocity de-orbit the object.

Again, initially this was not the case, as the thrust was produced toward the center of the Earth. As mentioned in the results section for the radial burn, due to this direction in thrust, the decay times of the orbits after the maneuver increased instead. From this one mistake, others ensued and will also be described in the following.

With the thrust vector pointed towards Earth, each simulation took a very long time to finish one iteration of the implemented scenarios. It seemed that this vector caused the simulation to be computationally more difficult. Also, the simulation kept running because it did not de-orbit. It is believed that if it did de-orbit, it would have ran into the first problem described above and kicked out of the ODE solver at around 130 km in altitude. As previously mentioned, the computations took a long time to complete that multiple computers were used to run different simulations just to obtain the simulation data. Even with multiple computers, the data acquisition time was still unsatisfactorily long. It was originally planned to complete other analyses for this project such as determining the re-entry footprint, a ground track plot, and possibly planning for a simple de-orbit mission. These aforementioned pieces of analyses were discarded due to the delay caused by the data acquisition.

Other problems caused by the radial thrust vector were that some of the scenarios didn't run properly, some scenarios didn't even provide enough data, and the decay time estimation described earlier didn't always work. As mentioned earlier, multiple computers were used in obtaining data. The technically superior computer could not run simulations at and lower than 0.2 km/s ΔV, while a laptop finished these simulations (although it took a significantly longer time to complete). It was believed that the differing versions of MATLAB[9] on each respective computer might have caused this. Some scenarios also didn't output enough data to provide an estimate in the decay time. In these instances, it is believed that the scenario did cause the HST to de-orbit, but because of the forced time step it only recorded data at specific times, and obviously not at enough times. Lastly, the estimation method didn't always work, as some of the estimated decay times were negative, indicating a positive slope in the trend line of the minimum points. This indicated that there weren't enough local minima data points to create a negative sloped trend line. A longer simulation time would've helped created better data, but as previously discussed, longer simulation times would have caused longer delay in data acquisition. Therefore, nothing else was really done about this result, as it further proved that the radial burn simulation was not the best way to perform a de-orbit maneuver. As a last remark, after the vector was changed to be in the negative velocity direction, it was realized that there would've been enough time to do all the discarded pieces of analyses if the vector was correct from the start. The data acquisition time decreased from three straight days to about 10 minutes. This was an unfortunate result, but it was an important lesson learned. Although, the radial burn did cause a de-orbit (some simulations more apparent than others). In retrospect, the radial propulsion burn could be thought of as the "not ideal" direction, rather than the wrong direction.

IV. Conclusion

This project was conducted to examine and investigate various propulsion systems for a theoretical external attachment for a controlled-manual de-orbit of the HST. The various propulsion systems initially considered were cold gas, solid, liquid, and EP systems, but through a basic trade study conducted, the EP option was eliminated. The cold gas system was also eliminated after considering how much propellant mass was required to perform the de-orbit maneuver. Two different types of simulations were conducted to determine the re-entry times from a ΔV maneuver. A tangential burn was conducted to decrease the velocity to allow drag and gravity to de-orbit the HST. A radial burn was also conducted to decrease the perigee of the orbit and also allow drag and gravity to de-orbit the HST. The final results concluded that the tangential burn was the more efficient way to de-orbit in comparison to the radial burn. The radial burn increased the decay times so much that, for feasibility purposes, it was concluded that this option shouldn't be considered. Also, from the cases conducted, a propulsion system capable of using propellant with an Isp of 450 seconds burning for 180 seconds was the most feasible option. The 450 seconds Isp provided the least amount of required propellant mass while the 180 seconds burn time generated a more reasonable thrust magnitude in comparison to the others. Ideally, a liquid propulsion system capable of $\geq$ 450 seconds of Isp and burning for up to a maximum of 10 minutes could provide a possible optimum propulsion system for this maneuver.

If future work were to be completed and continue this project, a few suggested topics to consider are as follows. All previously mentioned work that was initially thought to be included for this project, such as determining the re-entry footprint, a ground track plot, and possibly planning for a simple de-orbit mission, could be completed. Further analysis and calculations to include an estimate for the size and mass of a complete spacecraft attachment could also possibly be conducted to produce higher fidelity de-orbit data. Other additional research could be done on the actual method of rendezvous and attaching to the HST. It could possibly be completed autonomously or even manned like previous service missions on the HST.

Appendix

A. Raw Data

TLE for the ISS[5] used for the code Fidelity Check

```
1 25544U 98067A   12038.99170544  .00009608  00000-0  12760-3 0  9670
2 25544 051.6419 055.1125 0020295 018.7741 080.7026 15.58746633757711
```

Epoch (UTC):	23:48:03, Tuesday, February 7, 2012
Eccentricity:	0.0020295
Inclination:	51.6419°
Perigee height:	377 km
Apogee height:	405 km
Right Ascension of ascending node:	55.1125°
Argument of perigee:	18.7741°
Revolutions per day:	15.58746633
Mean anomaly at epoch:	80.7026°
Orbit number at epoch:	75771

TLE for the HST[5] used for all of the HST simulations

```
1 20580U 90037B   11304.18123544 +.00002224 +00000-0 +15437-3 0 08623
2 20580 028.4698 207.9317 0003445 130.5397 229.5258 15.01712402980112
```

Epoch (UTC):	04:20:58, Monday, October 31, 2011
Eccentricity:	0.0003445
Inclination:	28.4698°
Perigee height:	559 km
Apogee height:	564 km
Right Ascension of ascending node:	207.9317°
Argument of perigee:	130.5397°
Revolutions per day:	15.01712402
Mean anomaly at epoch:	229.5258°
Orbit number at epoch:	98011

B. Tabulated Results from Each Simulation

Table 2. Tangential Burn Simulation Results

	Burn Time [s] ↓	Isp [s] →	30	50	70	250	300	350	400	450
Propellant Mass [kg]*			7456.08	4008.81	2734.85	706.12	585.39	499.91	436.21	386.91
Re-entry Time [min]	10		32.02	32.02	32.02	32.03	32.03	32.03	32.03	32.03
	30		32.12	32.13	32.12	32.12	32.12	32.12	32.12	32.12
	60		32.32	32.32	32.30	32.30	32.30	32.30	32.30	32.30
	90		32.53	32.52	32.52	32.52	32.50	32.50	32.50	32.50
	120		32.77	32.75	32.73	32.72	32.72	32.72	32.72	32.72
	150		33.00	32.97	32.95	32.93	32.93	32.93	32.93	32.93
	180		33.23	33.20	33.20	33.15	33.15	33.15	33.15	33.15

*Varied Isp only affected the propellant mass calculation, not burn time

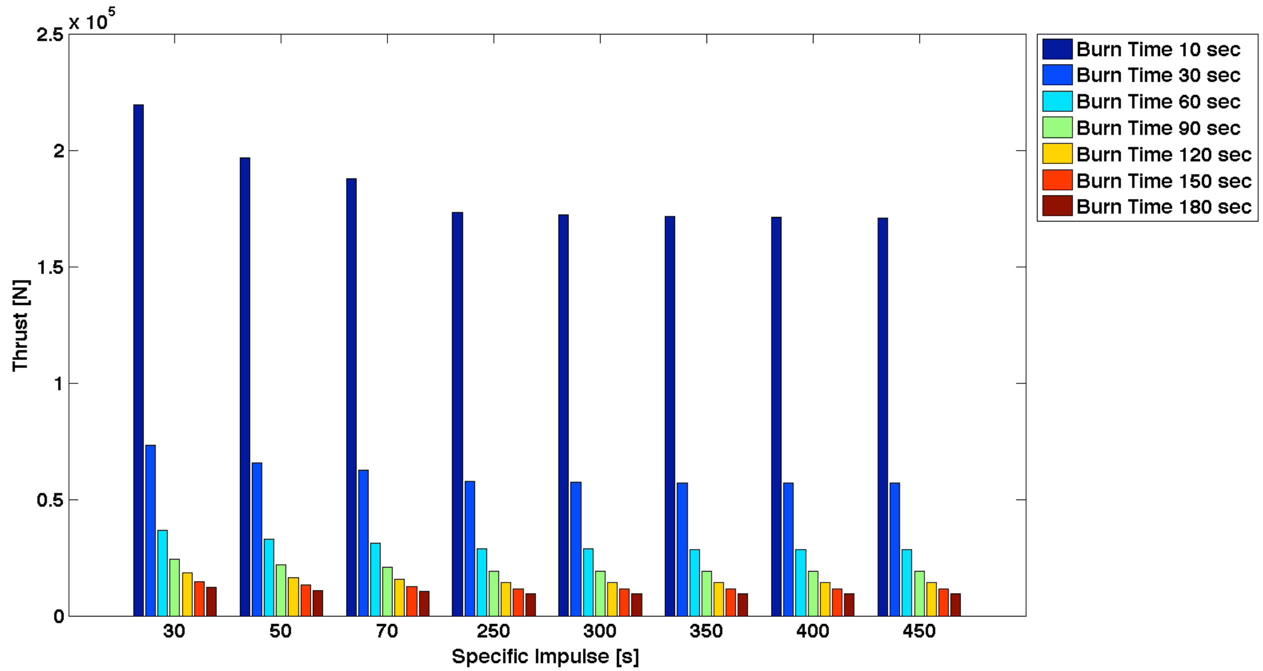

Figure 7. Produced Thrust to Perform the Calculated ΔV Maneuver for Each Varied Isp

Table 3. Radial Burn Simulation Results

	ΔV [km/s] ↓	Isp [s] →	300	350	400	450
Propellant Mass [kg]	0.1511*		585.389	499.910	436.212	386.910
	0.2		781.258	666.373	580.938	514.917
	0.25		985.010	839.120	730.858	647.333
	0.3		1192.253	1014.401	882.701	781.258
	0.35		1403.047	1192.253	1036.491	916.707
	0.4		1617.453	1372.714	1192.253	1053.700
Re-entry Time [weeks]	0.1511*		-21898.636	6653.128	3905.578	141388.361
	0.2		2946.537	3726.701	3726.701	3726.701
	0.25		1257.608	4464.203	-50150.113	90.099
	0.3		-1389.812	300.667	5.328†	**
	0.35		87.271	3.096†	**	**
	0.4		3.071†	**	**	**

*Calculated ΔV from Eq. 10
**Didn't provide enough data to create trend line estimation
†Apparent re-entry through altitude plot, therefore the decay time estimation wasn't needed

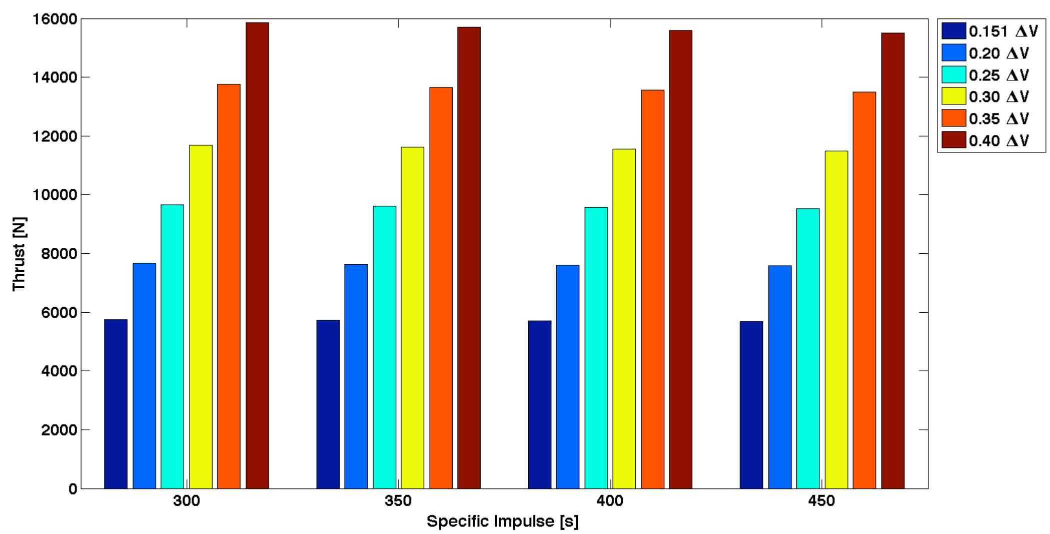

Figure 8. Produced Thrust to Perform the Varied ΔV Maneuver for Each Varied Isp

C. MATLAB[9] code

Available and provided upon request from the Aerospace Engineering Department office.

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