SEDS USA-SSPI 2017-18 Chapter Competition Report
"Designing the Robotic Space Tug"

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Table of Contents

Acknowledgments

We would like to thank the following individuals and groups for their assistance:

Ariane Cornell of Blue Origin, for her mentorship, help, and support throughout this project. She was invaluable in improving the quality of our presentation and we appreciate her efforts.

We would also like to thank SEDS and SSPI for sponsoring, hosting, and giving us the opportunity to participate in this competition.

Introduction

During the past decade there has been an increased focus on the development of a private space industry. The space industry represents a huge economic opportunity both as a new economic sector here on Earth and from the potential resources we can gather from space. Additionally, satellites are vital to the economy due to their multifaceted utility in communications, GPS capabilities, and research in many fields of science and technology. There are also new government funded programs such as the return to the Moon and human exploration of Mars, which will provide another opportunity for private industry to make a profit.

As space opens to new opportunities, the idea of a new multipurpose spacecraft has drawn the attention of governments and private space agencies. These entities desire a multifunctional spacecraft that could act as a workhorse in and around Low Earth Orbit (LEO). This conceptual spacecraft has been referred to as a “tug” due to a functional similarity to tugboats used at sea. The role of a space tug would be vital to this opening of space as a commercial frontier. There are several capabilities that would be incredibly beneficial to governments, private space organizations, and the commercial sector. These include the relocation of spacecraft, clearing of space debris, provision of maintenance to space stations, and material transport. A tug with this functionality would save time and money for all involved, and in most cases, these missions would be operated from the ground, without risk to human life.

It has been previously documented that space debris is a large threat to current space missions. This is due to the existence of over 500,000 known objects of debris greater than the size of a marble in addition to millions of untracked particles. With travel speeds up to 17,500 miles per hour, debris of a small size can do large damage to a spacecraft and create even more debris [1]. This shows the critical need to not only clean up space debris, but also to move those spacecraft that are in a direct path of collision with current debris.

Our team designed a system that we call the Sisyphus Tug-Station System (STSS). The name is derived from the figure Sisyphus in Greek mythology, who was punished by the gods to endlessly roll a large boulder to the top of a hill. We believed this constant act of pushing is a close analogy to our system, as a tug’s primary purpose is to move objects from one orbit to another or removing it from orbit entirely.

The design of this system focuses on a spacecraft that would emphasize cost efficiency and flexibility for future development. This ensures that the product is economically viable and will make the largest possible impact soon. This design features the ability to deorbit and maneuver satellites operating from an orbital station located in LEO. Additionally, the Sisyphus Tug can target larger pieces of space junk and remove them from heavily populated orbital radii. Finally, the Sisyphus Tug can be outfitted with a module designed to move supply capsules to the Lunar

Orbital Platform-Gateway (LOP-G). This Lunar space station is planned to be constructed within the decade and would be a lucrative mission objective. This period is believed to be appropriate for the development and implementation of the STSS.

The following report will further discuss the details and applications of the proposed design, the results of the technical analysis conducted, and the reasons why this design could be widely used and feasible in the current economic and technological environment.

Design Objectives and System Requirements

The primary design consideration when choosing objectives and functionality was the question of what would appeal most to the commercial and government sectors. This economic viability also had to be balanced with what can be accomplished technically. For instance, initially the idea of having a remote maintenance capability seemed highly economical. For a comparatively low cost, a satellite costing hundreds of millions of dollars could be repaired, which is an opportunity many commercial sectors would love to use. However, upon further research we discovered the immense variety of technical failures that impact spacecraft, and how most current orbital objects do not have replaceable parts. This type of mission objective would require very specialized satellites and would greatly increase the upfront costs in our system implementation. Thus, we decided to focus on missions that would be widely used and highly profitable. We also created the concept with flexibility in mind to allow the system to be open to future endeavors, such as growing into having a range of repair capabilities.

As we mentioned in the introduction, the system that we designed consists of two different orbital elements. The first is the Sisyphus Tug satellite which will carry out the mission objectives. The second is the Sisyphus Station which is designed as a docking and resupply location for the Tug. This Station will drastically improve the efficiency of our system because it will allow for orbital refueling and act as a meeting point for cargo shuttling. It will also give us a greater variety of functionalities and will provide the freedom to expand operations as the need arises. This project will serve as a proof of concept, describing the use of a single Station and Tug and with initial mission objectives that were chosen to maximize marketability. This concept could be expanded on with specialized STSS units in a variety of orbits.

The primary objective of the Sisyphus Tug will be removing debris and adjusting satellite orbits. The base orbit for the system will be at an 800-kilometer (km) altitude, inclined at 28.5 degrees. From here, the Tug will make use of a claw system and will rendezvous with orbital objects to either move or deorbit them. The secondary mission of our Sisyphus Tug will be to deliver payload to the proposed lunar space station (LOP-G) that is in development. This array of functionality can be utilized by a variety of industries where they will benefit from the high efficiency of the ion engines on the Tug. This will give more freedom to future projects around Earth and could be vital in extending the lives of various ongoing missions. The existence of our system will also allow the possibility of launch vehicles to get smaller, as satellites destined for high altitude orbits would have the option of entering a lower parking orbit where the Sisyphus Tug could rendezvous and maneuver them. More in-depth definitions of the mission parameters for both objectives will be defined later in the report.

Spacecraft Design

Initial Launch

The Sisyphus Tug/Satellite System is designed to be fully operational after a single launch. This launch is intended to be on SpaceX's Falcon Heavy rocket. Due to its successful test launch in February 2018, the team judged the use of the Falcon Heavy to be feasible within a few years' time. Although other rockets are smaller and more widely used, the Falcon Heavy currently provides the largest mass to orbit capabilities, a desirable feature for our design due to our Stations fuel repository functionality.

The launch will occur from Kennedy Space Center in Cape Canaveral, Florida. This is due to a variety of factors: additional ΔV is obtained from launching closer to the equator and due east; lower initial orbit inclination; and many American aerospace companies and defense contractors have a friendly relationship with the U.S. Government, which owns and operates all facilities at Kennedy Space Center. Additionally, we desire an orbital altitude for both the Tug and the Station of 800 km due to much orbital debris laying in and around this orbit. This is demonstrated at the first peak in Figure 1 below:

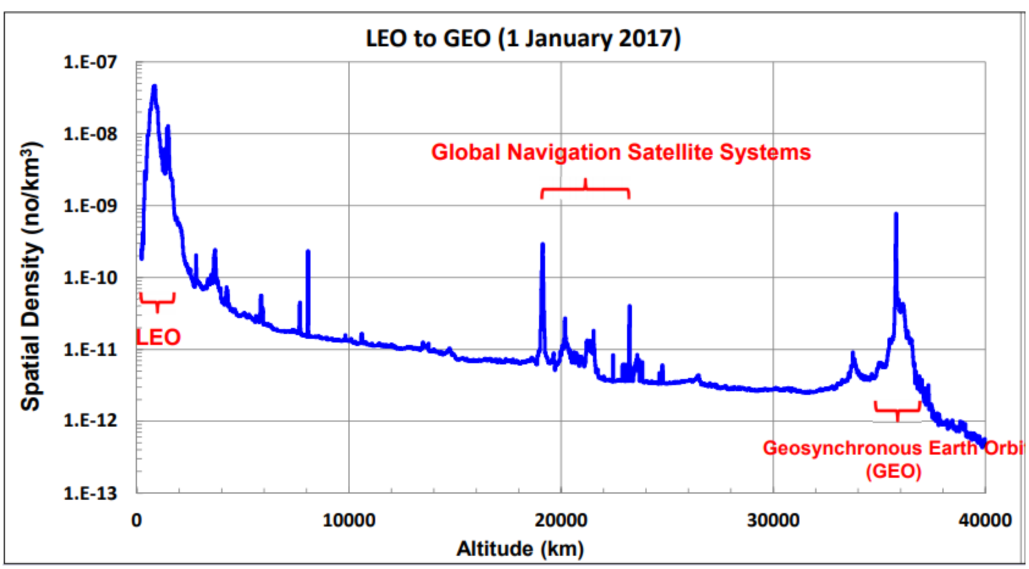

Figure 1: Spatial Density in Earth Orbit vs Altitude [2]

To determine the maximum mass of the STSS, we first need to determine the total ΔV required for the 800 km altitude orbit, then back out the mass using the ΔV equation modified for a 3-stage rocket. First, we had to find the required ΔV to get to orbit. Our first step was to find the velocity at a parking orbit altitude of 300km using the circular orbital velocity equation (all universally constant variable values are defined in the appendix):

$$v_c = \sqrt{\frac{\mu_{Earth}}{r_{orb}}} \tag{1}$$

Where r_{orb} is 6678 km (300 km altitude). After this maneuver, a Hohmann Transfer can be utilized to approximate the ΔV from the launch site to the final orbit altitude of 800 kilometers:

$$\Delta v_1 = \sqrt{\frac{\mu_{Earth}}{r_1}}\left(\sqrt{\frac{2r_2}{r_1 + r_2}} - 1\right) \tag{2}$$

$$\Delta v_2 = \sqrt{\frac{\mu_{Earth}}{r_2}}\left(1 - \sqrt{\frac{2r_1}{r_1 + r_2}}\right) \tag{3}$$

$$\Delta v_{HT} = \Delta v_1 + \Delta v_2 \tag{4}$$

Where r_1 and r_2 are 6678 km and 7178 km, respectively. After this value was found, the velocity assist from Earth was calculated using the latitude of Kennedy Space Center, an Earth sidereal day, and a due-east launch azimuth:

$$v_e = \frac{2\pi}{23.9345hr}\frac{3600s}{hr}\cos(\phi_{KSC})\sin(Az) \tag{5}$$

For this system, drag and gravity losses for ΔV were approximated as 0.15 km/s and 1.5 km/s, respectively. Summing the circular velocity at 300 km altitude and the Hohmann transfer ΔV, then subtracting the ΔV from the Earth gives us the total ΔV requirement (ΔV_{req}) of 9.2422 km/s for our rocket to provide.

Next, the mass of the STSS was iteratively reduced from the max value of 63,800 kg [3] using the following stage setup and values to find the maximum mass four our desired altitude:

Figure 2: Staging Configuration for Calculations

Table 1: Initial Values for Iteration

Variable	Description	Value	Units
m_{FH}	Total mass of launch vehicle (LV) including payload	1,420,788 [3]	kg
$m_{pay,i}$	Initial payload mass	63,800 [3]	kg
f_{split}	Fuel split between booster and main engines, determined by booster engine cutoff (BECO) and main engine cutoff (MECO) times	0.1902	--
m_{p1}	Mass of propellant in stage 1	1,111,831	kg
m_{p2}	Mass of propellant in stage 2 (left at BECO)	75,269	kg
m_{p3}	Mass of propellant in stage 3	92,670	kg
$I_{sp,1}$	Specific impulse for stage 1	282	s
$I_{sp,2}$	Specific impulse for stage 2	311	s
$I_{sp,3}$	Specific impulse for stage 3	348	s
ΔV_{FH}	Total ΔV the Falcon Heavy can provide with this payload	8.3934	km/s

Assumptions made for the above constant values are the following:

1. Constant flow rate of propellant
2. The specific impulse for stage 1 is approximately close to the sea level value
3. The stage 2 specific impulse is approximately the vacuum value

The constant flow rate of propellant assumption was utilized to determine the fuel left in Stage 2. Additionally, the booster engine cutoff and main engine cutoff times are 149 seconds and 184 seconds, respectively [4]. Using these values as a baseline, the mass of the payload was iteratively reduced until the ΔV_{FH} was greater than ΔV_{req}. The following table indicates the final iterated values:

Table 2: Final ΔV and Mass Values for Launch

Variable	Description	Value	Units
$m_{tot,1}$	Stage 1 total mass of LV and system	1,406,720	kg
ΔV_1	ΔV provided by stage 1	4.5445	km/s
$m_{tot,2}$	Stage 2 total mass of LV and system	243,689	kg
ΔV_2	ΔV provided by stage 2	1.1271	km/s
$m_{tot,3}$	Stage 3 total mass of LV and system	142,820	kg
ΔV_3	ΔV provided by stage 3	3.5729	km/s
ΔV_{tot}	Total ΔV provided	9.2445	km/s
m_{sys}	Mass of STSS delivered to 800 km altitude circular orbit	46,250	kg
$m_{tot,f}$	Final total mass of system and LV	1,406,720	kg

The payload included in this launch of the system will be comprised of both the robotic space tug, the claw system and the station module. After the ΔV from the launch vehicle has been applied, the Sisyphus Tug/Station System will have the following orbital parameters:

Table 3: Orbital Characteristics Post-Launch

Variable	**Description**	**Value**	**Units**
a	Semi-major axis	7178	km
e	Eccentricity of the orbit	1E-6	--
i	Inclination of the orbit	28.5	deg
RAAN	Right Ascension of the Ascending Node	0	deg
ϖ	Argument of perigee	0	deg

Launch Configuration

Figure 3: Launch configuration of the System

As previously mentioned, we designed our system to be capable of getting into orbit in one launch. One of the main features for this orientation to be possible is a coupling structure that is androgynous, meaning that any docking port can pair with any other available. To fit inside a standard fairing, the arms of the Station were retracted to fit flush against the hull. The solar panels are retracted into deployment boxes attached to the hull of both the tug and the Station. Once the system reaches orbit, the tug will detach and use its onboard attitude control system and thrusters to navigate to the opposite end of the Station. Once there, the claw system will grab onto the nose of the Tug, decouple from the station, and onboard monopropellant thrusters on the Tug will “reverse” both the Tug and claw away from the station to a safe distance. Station arms will then deploy, and the Tug will rendezvous with one of the arms to park the claw system, where the Latching End Effector (LEE) on the arm will mate with the claw system. The claw will then release the tug, and the tug will rendezvous with the opposite side of the claw system, where it will finally dock and become fully operational.

Robotic Space Tug

Figures 4 and 5: Visual Representation of the Sisyphus Tug

To further define our Tug's mission, we have additional performance parameters we wanted to meet. We desired it to be capable of deorbiting debris from a range of orbits, be capable of moving geostationary satellites to a dead orbit that follows the proposed guidelines [5], and finally be capable of acting as a cargo runner to and from the proposed Lunar Orbital Platform-Gateway. We also utilize this spacecraft as a technology demonstration platform for the Variable Specific Impulse Magnetoplasma Rocket (VASIMR) VX-200 engine with a total of four onboard. Each engine produces 5 N of thrust and draws 200 kJ/s of energy [6], a large requirement for the solar panels and batteries.

Deorbiting Debris

We first required the Tug to be able to deorbit satellites at an orbital altitude of 300 km, as drag effects will lower the orbit of satellites. A simulation was run in FreeFlyer, a spacecraft dynamics software, to examine area-to-mass ratios to help determine the lifetime of satellites in this orbit:

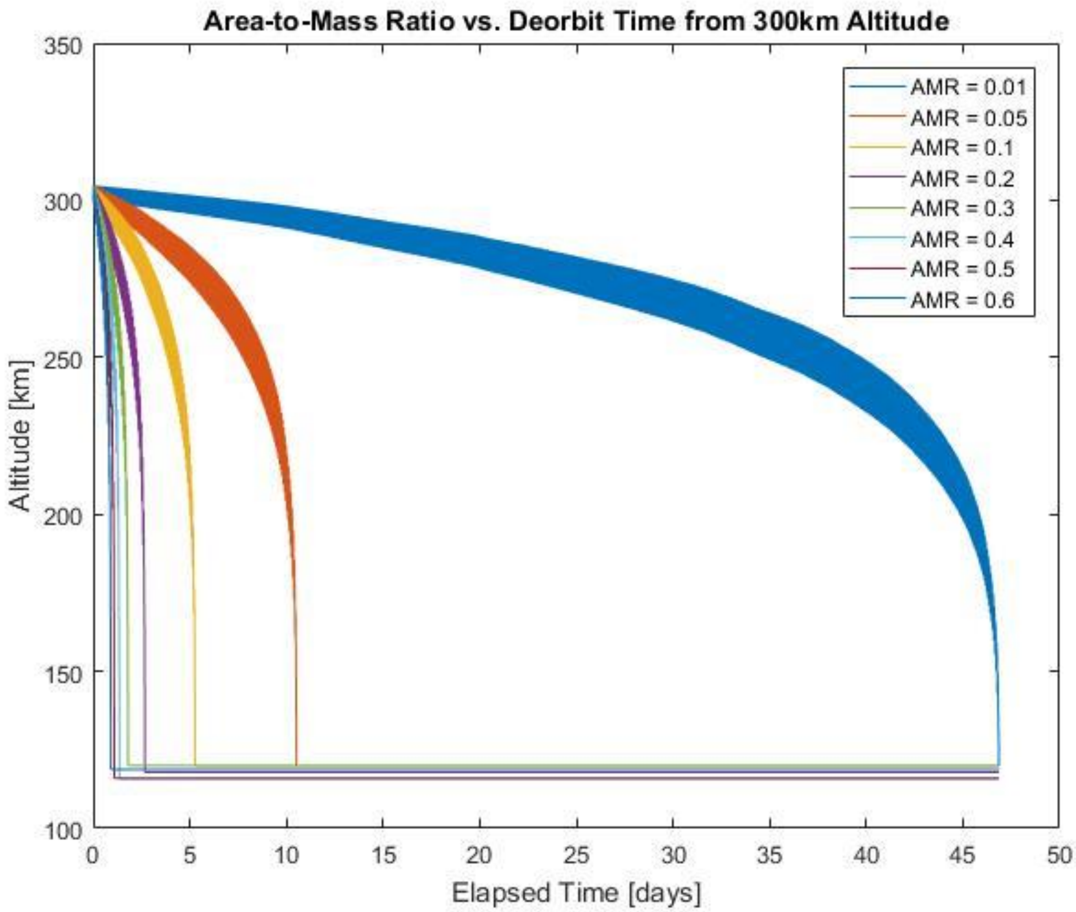

Figure 6: Area-to-Mass Ratio Effects on Deorbit Time

To conduct the above analysis, debris items were simulated using the point-mass Earth force model and drag effects were simulated using the Jacchia-Roberts atmospheric model. A debris item was defined as "deorbited" if the altitude reached below 120 km to allow FreeFlyer to propagate all debris items. From the above graph, area-to-mass ratio (AMR) controls the amount of time to deorbit for these items. The simulation was run for AMR values of 0.01, 0.05, 0.1, 0.2, 0.3, 0.4, 0.5, and 0.6. Any debris with an AMR higher than 0.6 is expected to deorbit in less than a day.

From the above simulation, the following time to deorbit for our debris was generated:

Table 4: Debris Time to Deorbit Dependent on Area to Mass Ratio (AMR)

AMR [-]	Time to Deorbit [days]
0.01	46.8675
0.05	10.5214
0.10	5.2708
0.20	2.6788
0.30	1.8084
0.40	1.3588
0.50	1.0943
0.60	0.9144

Propellant Mass Calculation

Next, we needed to determine the maximum amount of propellant that the Tug would carry. This is where our second requirement of resupplying the Lunar Orbital Platform-Gateway (LOP-G) is implemented. To determine the maximum amount of propellant, we investigated the case of resupplying the LOP-G due to the cargo mass and ΔV requirements being large. We utilized Cygnus Enhanced as our cargo module for resupply and used its fully-laden mass characteristics to size our tug appropriately. Additionally, the orbital case analyzed was when the Ascending Node (RAAN) of the orbit of the Moon and the Tug were in-line and the Moon was at its average radial distance of 384,400 km [7]. For the final orbit to determine the mass of propellant, we used one of the proposed orbital radii of 70,000 km from the Moon [8]. The following steps were analyzed to approximate the ΔV budget:

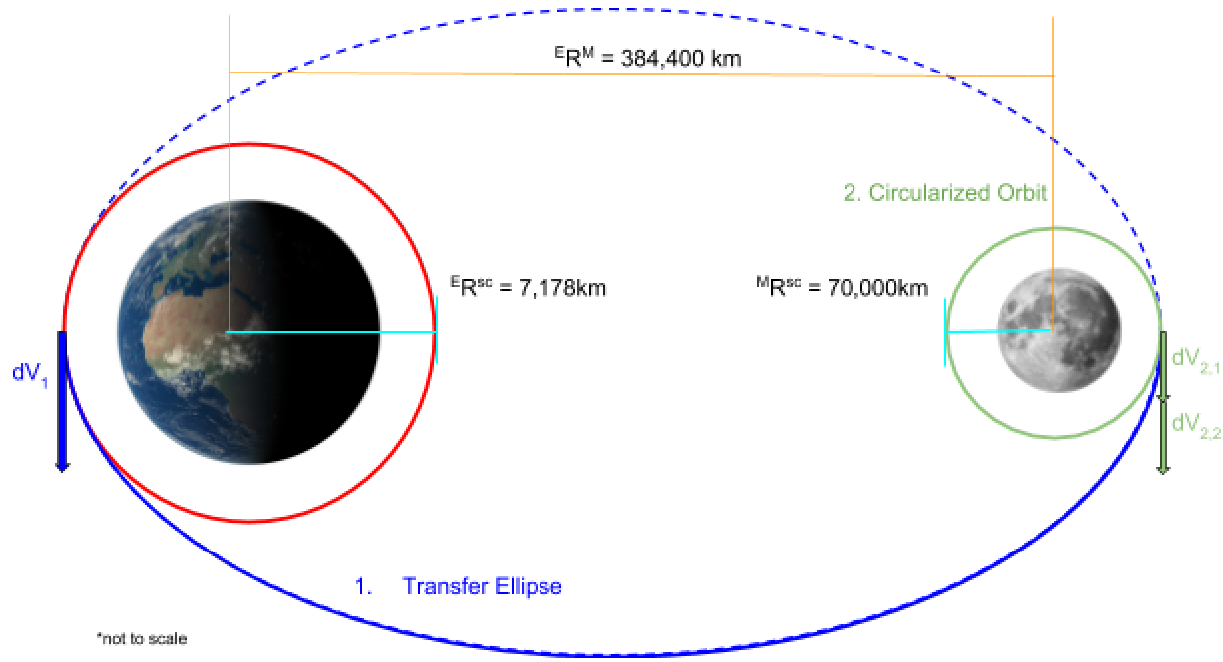

Figure 7: Orbit from Tug to LOP-G

The steps in the orbit are characterized as follows:

1. Transfer ellipse from 800 km altitude orbit (7178 km radial) to average Moon radial distance of 384,400 km plus desired 70,000 km orbit around the Moon (ΔV_1). ΔV_1 is determined by equation (2) with r_1 = 7178 km and r_2 = 454,400 km.

2. Perform circularizing burn at apoapsis to include both the Moon orbital velocity relative to Earth ($\Delta V_{2,1}$) and the circular orbital velocity about the Moon ($\Delta V_{2,2}$). This maneuver was determined with the following set of equations:

$$v_{c,m} = \sqrt{\frac{\mu_{Earth}}{r_{Moon}}} \tag{6}$$

$$\epsilon = -\frac{\mu_{Earth}}{2a_{SM}} \tag{7}$$

$$v_{ap} = \sqrt{2 * \left(\epsilon + \frac{\mu_{Earth}}{r_2}\right)} \tag{8}$$

$$v_{c,\frac{m}{sc}} = \sqrt{\frac{\mu_{Moon}}{r_{orb2}}} \tag{9}$$

$$\Delta v_{2,1} = v_{c,m} - v_{ap} \tag{10}$$

$$\Delta v_{2,2} = -v_{c,\frac{m}{sc}} \tag{11}$$

$$\Delta v_{tot} = 2(\Delta v_1 + \Delta v_{2,1} + \Delta v_{2,2}) \tag{12}$$

Where r_{Moon} is 384,400 km, a_{SM} is the semi-major axis value, v_{ap} is the velocity at apoapsis of the transfer orbit, $v_{c,\frac{m}{sc}}$ is the circular velocity of the spacecraft with respect to the moon with the desired 70,000 km orbit r_{orb2}. The total ΔV for this maneuver is multiplied by 2 as the return to the Sisyphus Station is expected to be the same as the initial maneuver. Thus, the total ΔV for this maneuver is 7.0442 km/s.

To determine an optimal mass for the total tug system, including propellant and the cargo module, an initial guess of 10,000 kg was used for the entire system, including the Tug, Cygnus Enhanced module, and propellant. Additionally, inert mass was assumed with a value of 2,500 kg. To optimize this system, the mass of Cygnus and the inert mass was removed from the initial estimate to define the initial propellant mass. Using the total mass, the fuel mass required to complete the maximum maneuver with the cargo module was calculated using the following equations:

$$m_e = m_0 * e^{-\frac{\Delta v}{g * I_{sp}}} \tag{13}$$

$$m_p = m_{sys} - m_e \tag{14}$$

If the mass of the fuel required was less than the difference between the entire system mass and the inert mass, then the entire system mass was reduced. This method repeated until the guessed mass and the calculated mass of propellant converged.

Additional Mass Considerations and Final Values

Additional important mass characteristics for this spacecraft are the battery and solar panel masses, which is why we opted for a large initial guess for the inert mass. To determine the mass of the batteries, we identified a few key factors in determining the requirements for minimum battery life. We determined the following requirements must be met:

1. The spacecraft must survive the maximum time in the Earth's shadow that can be experienced through an Earth year at the Station radius of 7178 km
2. The batteries must optimize specific energy density
3. Engines must be able to be on at max power (or approximately close) during the duration of the time in shadow

To determine the maximum time in shadow, the Tug's orbital parameters were inputted into FreeFlyer. The simulation was propagated using the Point Mass model and the J2 perturbation model over 1 full Earth year. Over this period, the maximum time in shadow recorded was 35 minutes. Knowing the time in shadow will determine the maximum capacity of the batteries and the total area of solar panels we require.

According to research, current technology for batteries that was both rechargeable and optimized specific energy density. Specific energy density refers to a battery's ability of storing energy per unit mass, typically reported in Megajoules per kilogram (MJ/kg). Considering both variables, the best current technology is provided by Panasonic with the 2170 Model [9]. This is a lithium-ion battery with a specific energy density of 0.9 MJ/kg. Given that our energy requirements for the engines is 800 kJ/s and knowing the maximum time in shadow, we found that the mass of our battery system is 1,866.67 kg.

To recharge the batteries, we compared the advantages and disadvantages between solar power and nuclear power. Because of launch concerns, fallout, and the possibility of radiating cargo, we determined that solar power was the best feasible option for spacecraft power generation. Due to our low thrust engines, we desired to minimize the weight of the solar panels and sought the solar panels with the highest specific power density. We determined that the best material for this mission is ultra-thin polymeric substrate film LaRCtrade-CP1, having a specific power density of 3,200 W/kg [10]. The minimum power requirement is determined from the rate at which the engines consume energy of 800 kJ/s (kW). Dividing the minimum power requirement by the rate at which engines consume energy, we get 250 kg of solar panels required for this mission. Since solar panel performance decreases over time, we will include an additional 50 kg of solar panels for an additional 150 kW of power, prolonging the mission lifetime of this system.

Finally, we looked at existing docking ports for connection between spacecraft. Using APAS as an example [11], the system has an approximate mass of 300 kg. However, since our system does not require an airtight seal between artifacts, and there is no transfer between modules, we estimated our soft docking system to be around 100 kg. These mass values are included in the structural mass of the tug, and the estimated mass of the claw system is defined in the "Robotic Arms Modification" subsection.

The final mass values of the various system components are in the table below:

Table 5: Mass Specifications for the Tug System

Variable	Description	Value	Units
$m_{i,cyg}$	Inert mass, Cygnus Enhanced	1800.00	kg
$m_{c,cyg}$	Cargo mass, Cygnus Enhanced	3500.00	kg
m_{tug}	Total tug mass, no modifications	3704.50	kg
$m_{i,tug}$	Inert mass, tug	2500.00	kg
$m_{p,tug}$	Argon propellant mass	1204.61	kg
$m_{s,tug}$	Solar panel mass	300.00	kg
$m_{b,tug}$	Battery mass	1866.67	kg
$m_{a,tug}^{*}$	Claw system mass	250.00	kg
$m_{t,tug}^{**}$	Structure inert mass	333.33	kg

*The robotic arm system designed for this project is evaluated under the "Robotic Arms Modification" section.
**Leftover inert mass after removing calculated solar panel and battery masses, used for structure, fuel tank, and docking port.

Capabilities Analysis

Now that the masses of the spacecraft, claw system, and propellent are known, analysis on the system capabilities can be performed. To deorbit debris, the spacecraft must use the claw modification, making the fully-laden mass of the system to be 3,954.50 kg. To analyze the maximum mass of a debris object that the Tug would be able to deorbit, we analyzed the required change in the orbital inclination and orbital radius to achieve a rendezvous with debris. The order of operations and mass changes for our orbital analysis is indicated below:

1. Hohmann Transfer to debris orbit, changing mass of propellant
2. Incline up/down to target, changing mass of propellant
3. Incline up/down to Sisyphus Station plane, changing mass of propellant and including debris mass
4. Hohmann Transfer to deorbit altitude of 150 km, changing mass of propellant and including debris mass
5. Return to Sisyphus Station for refueling, changing mass of propellant and no debris mass

These steps were analyzed to find the ΔV required for the maneuver with the goal of determining the maximum mass of debris that can be deorbited at any given radius/inclination value. We assume that all the propellant mass is used as efficiently as the engine design allocates. We also assume that the spacecraft is pointed correctly for the entire burning maneuver, and that the maneuver is timed to where the inclination change occurs at matching RAAN values between the Tug and debris. Additionally, no argument of perigee change was accounted for, as our spacecraft and station are in a circular orbit initially, allowing argument of perigee to be determined by the application of the first radial burn.

To calculate the above procedure, equation (2) was used and the following equation for change in inclination:

$$\Delta v_{inc} = 2v_c \sin\left(\frac{\Delta i}{2}\right) \tag{15}$$

Where Δi is the change in inclination. Calculations were performed in the order given above and the results were summed. This returned an approximation for the total ΔV of a maneuver at any given inclination and orbital radius value. The ΔV is plotted below against inclination and orbital radius:

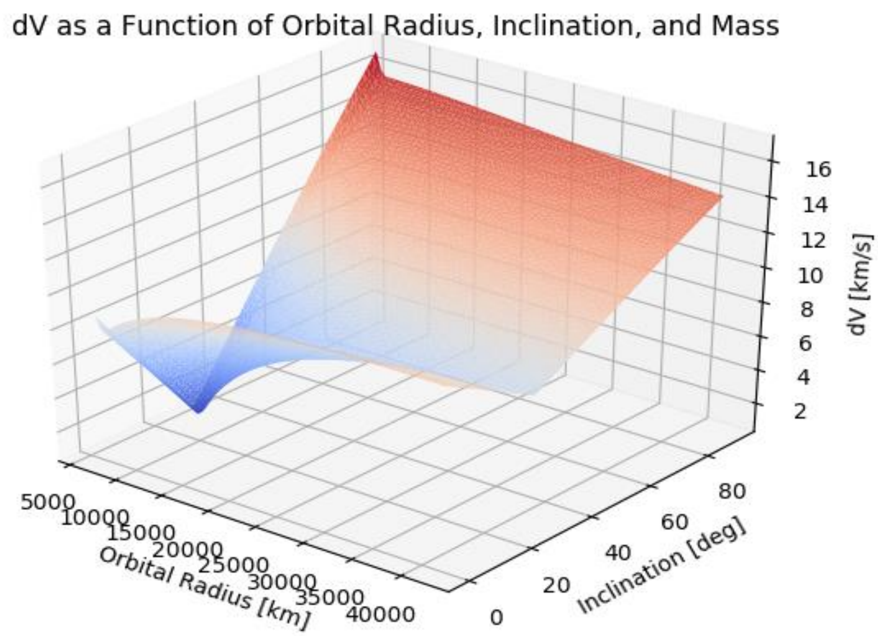

Figure 8: Plot of ΔV as a Function of Orbital Radius and Inclination

The ΔV cost displayed above follows conceptual knowledge of changes in inclination and change in radius maneuvers. As radius increases, the desired ΔV change increases at a lesser rate, which is visible at the inclination value of 28.5 degrees due to inclination change being zero. Additionally, the required amount of ΔV for an inclination change decreases as radius increases, which can be seen with the trace of the 90-degree case against the backplane of the graph.

To determine the maximum amount of mass that can be added to the spacecraft mass at steps 3 and 4, mass was added iteratively to the system, then the leftover propellant was calculated at the end of the maneuver. If the propellant mass was not equal to zero, the debris mass would increase until the mass of propellant was at or slightly above zero. To calculate the mass change of propellant, equation (13) and equation (14) were used.

After calculating each maximum mass value at each radius and inclination value, the following plot was generated to demonstrate the maximum mass with these two variables. We included a hard-upper limit of 14,911.8 kg as this is the maximum allowable additional mass on the arm structures discussed in the "Robotic Arms Modification" subsection:

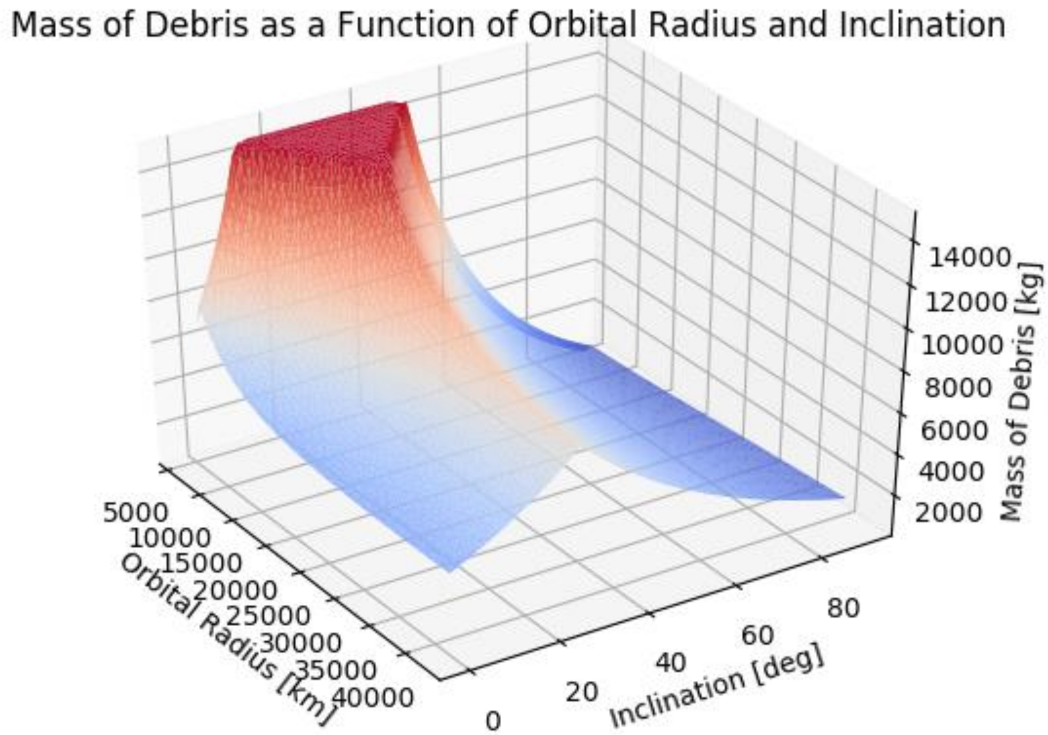

Figure 9: Mass of Debris as a Function of Orbital Radius and Inclination

The above plot demonstrates that our design is capable of deorbiting debris in GEO, but we expect most deorbiting missions to occur in LEO and the lower end of Medium Earth Orbit (MEO), defined as 80-2,000 km and 2,000-35,786 km, respectively.

Now that the maximum mass is known, and the maximum force of the engines is known to be 20 N (four engines with 5 N of thrust each), the acceleration for each maneuver can be determined using Newton's Second Law:

$$a = \frac{F}{m_{tug} + m_c + m_{deb}} \tag{16}$$

Where F = 20N and m_{deb} is the debris mass at the given orbital radius and inclination value. The maximum acceleration with the mass occurs when thrust is at its peak. These maximum acceleration values are demonstrated in the plot below, plotted against inclination and radius:

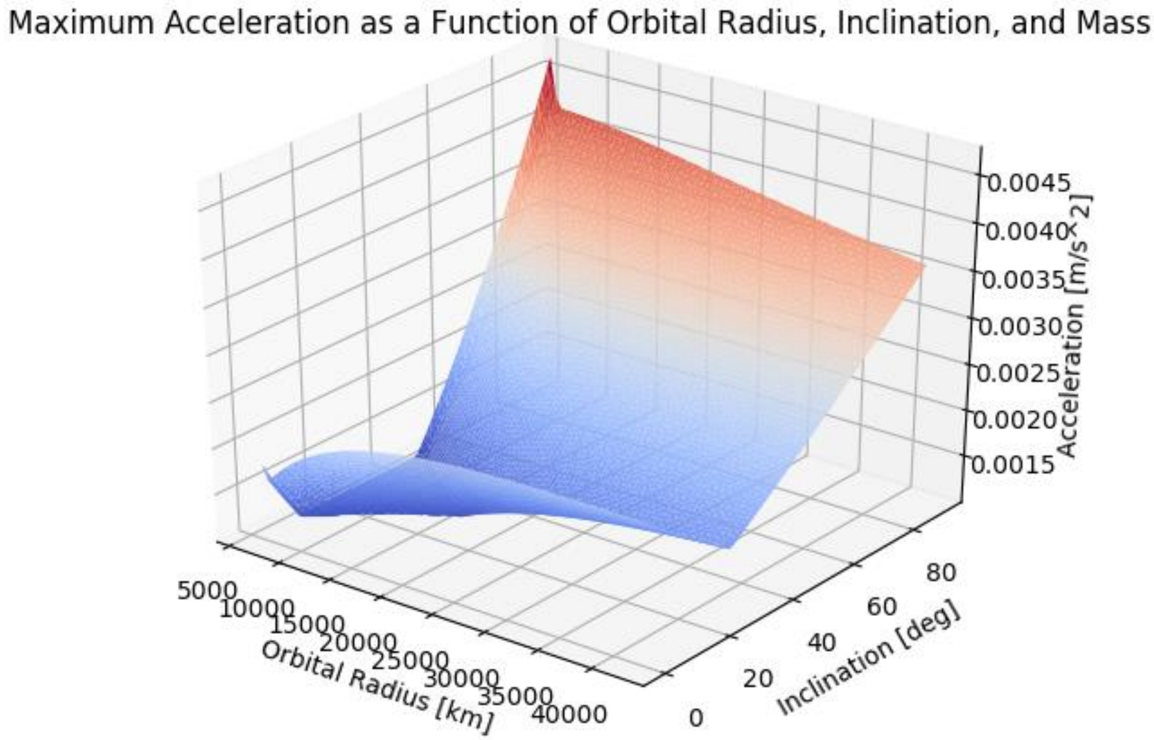

Figure 10: Plot of Max Acceleration as a Function of Orbital Radius, Inclination, and Mass

Since our force is assumed to be constant at 20 N, acceleration is a function of mass only in this simulation, indicating that at lower max mass values, the spacecraft can achieve higher acceleration.

Knowing the ΔV required and the acceleration at any point for a maneuver, the time to perform each maneuver can be found by performing the following calculation:

$$t = \frac{\Delta v}{a}\left(\frac{hr}{3600s}\frac{days}{24hr}\right) \tag{17}$$

Finally, the total time to perform any maneuver with the maximum mass of debris can be visually represented using the previous operation:

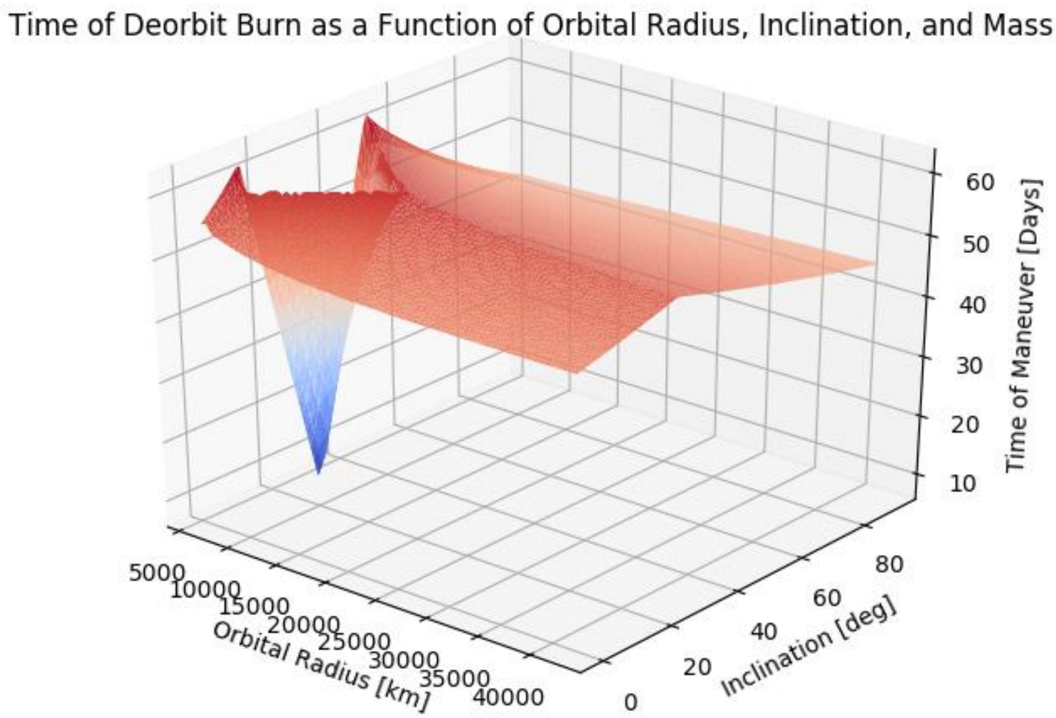

Figure 11: Time to Deorbit Maximum Mass Debris against Orbital Radius and Inclination

Since we are using low thrust ion engines, these thrust times are expected. These times define the maximum possible length of a mission at any given radius and inclination to deorbit debris, as a lower mass of debris will allow for a higher acceleration. Additionally, having a submaximal mass of debris would allow for less propellant to be loaded into the Tug, further decreasing the time to deorbit debris.

Examples of data pulled from the above plot are given in the table below to demonstrate inclination extrema along orbital radii change (note that the base orbital radius is 7178 km inclined at 28.5 degrees):

Table 6: Sample Data as a Function of Orbital Radius and Inclination

Orbital Radius [km]	Inclination [deg]	ΔV [km/s]	Max Mass [kg]	Max Acceleration [m/s^2]	Time to Deorbit [days]
7,400	28.31	0.8394	14,911.8	0.0011	9.2075
7,400	0	8.0175	7,810	0.0017	54.5841
7,400	90	15.7668	810	0.0042	43.4727
9,445	28.31	2.5041	14,911.8	0.0011	27.4681
9,445	0	8.8578	6,440	0.0019	53.2829
9,445	90	15.7172	830	0.0042	43.5179
24,271	28.31	6.7712	10,570	0.0014	56.9149
24,271	0	10.7349	4,180	0.0025	50.5340

24,271	90	15.0139	1,170	0.0039	44.5248
42,164	28.31	7.9004	8,110	0.0017	55.1587
42,164	0	10.9076	4,020	0.0025	50.3372
42,164	90	14.1542	1,620	0.0036	45.6611

LOP-G Resupply Mission Analysis

The ΔV for the mission to and from the LOP-G station has been determined to be 7.0442 km/s during the Tug's mass optimization. For this mission capability, we require our system to minimize cost. To assist with this goal, we chose to use the current International Space Station (ISS) cargo module Cygnus Enhanced from Orbital ATK as this system provides a large cargo mass of 3500 kg with a lower inert mass of 1800 kg [12]. Minor modifications in the base structure will need to be made to allow for coupling with our Tug. For this mission, cargo modules will be delivered to the Station from the Earth's surface, parking on one of the arms of the Station. The Tug will rendezvous with the cargo module and then deliver it to the LOP-G using the mission design outlined in the "Propellant Mass Calculation" subsection.

For resupplying the LOP-G, we determined that the maximum amount of mass that can be carried to the LOP-G is equal to the maximum mass that can be carried by the Cygnus Enhanced Cargo Module, 3500 kg, for a total initial mass of 9004.50 kg. We assumed that the cargo module will be filled with the same maximum amount of mass in refuse for the return journey to Earth. Finally, we gave the module 2 days' worth of parking at the LOP-G for movement of cargo and refuse. Performing the mission outlined in the "Propellant Mass Calculation" section and using a combined mass of the Tug and cargo module of 9004.50 kg, the total timeline for this maneuver is 38.7069 days including the parking at the LOP-G.

Graveyard Orbit Mission Analysis

For spacecraft that are placed in geostationary orbits, a more economical approach to removing them from their orbit is to move them into a specified graveyard orbit location. The graveyard orbit that was analyzed for maneuvering was an orbit that is located 300 km above geostationary orbits [13]. The following maneuvers were performed to efficiently deliver the geostationary debris to a graveyard orbit and return the Tug and claw system back to the Station:

1. Hohmann Transfer out to the Geostationary Orbit radial value, changing propellant mass
2. Inclination change to the target, changing propellant mass
3. Hohmann Transfer to the Graveyard orbit, changing propellant mass and including the target mass
4. Inclination change back to Sisyphus Station inclination, changing propellant mass
5. Hohmann Transfer back to the Sisyphus Station, changing propellant mass

Utilizing equations (2)-(4) and (15) from the “Capabilities Analysis” subsection and summing their values in the order presented above, the total ΔV for this maneuver is 10.5861 km/s. The mass of propellant is constant at each step, except for step 3. To determine the maximum amount of mass of propellant used, we used the maximum mass of a GEO object of 7000 kg [14]. The mass at each step is depicted in the table below:

Table 7: Mass analysis for graveyard orbit maneuver

Maneuver Step	**Mass before Maneuver [kg]**	**Propellant Mass Expended [kg]**	**Mass After Maneuver [kg]**	**With Target? (Y/N)**
1	3,954.50	287.79	3,666.71	N
2	3,666.71	111.70	3,555.01	N
3	10,555.01	2.34	10,552.67	Y
4	3,552.67	116.84	3435.83	N
5	3,435.83	250.37	3185.46	N

The total time to perform this maneuver was calculated by finding the average acceleration at each maneuver step, finding the time by dividing the step ΔV by the average acceleration at that step, and then summing the time values. The total timeline for a Tug with a full fuel tank to maneuver an object that is 7000 kg to a graveyard orbit is approximately 22.5 days. At the end of the maneuver, there is still 435.56 kg of fuel remaining. Optimizing the mass of fuel using a similar method to the one presented in the “Propellant Mass Calculation” section, we were able to optimize the initial fuel load to be 666.61 kg, with about 2 kg of fuel in reserve. This lowered our total maneuver time to approximately 19.5 days, a savings of 3 days.

Sisyphus Station Module

The proposed design includes the Sisyphus Tug and Station (Tug docked on the Station). It also includes one major “modification” that allows the Tug to move objects, sometimes referred to as the claw system, as well as the cargo modules. The Station is envisioned as the center of the operation to which the Tug can return between missions. The Station has three defining characteristics: solar panels for power production, internal fuel tanks for argon gas, and the external claw modification for the Tug. This claw modification aids in moving supplies to the LOP-G and moving objects to different orbits; they will be discussed in greater detail below. By using a single module that accommodates all aspects of the mission and does not require a human presence, the complexity and cost of the system decreases.

As mentioned above, the design calls for the Station to be located in a parking orbit altitude of 800 kilometers. As seen below in Figure 12, the outside of the Station has arms to hold the claw modification to be used by the robotic space Tug. Inside the Station, fuel tanks hold argon gas to refuel the Tug as needed. To mitigate further cost incursions, other existing spacecraft platforms will be slightly modified to deliver Argon to the Station, such as the Cygnus or Dragon resupply capsules. The capsules will only need minor modifications to succeed in their mission: gas pumps; our docking port instead of the Common Berthing Mechanism; fuel cross fed at the port; and ancillary power for the pump.

Figure 12: CAD Model Rendering of the Station

The Station also has two robotic arms that hold the claw mechanism when not in use and a cargo module for the LOP-G. Additionally, this design allows for the evolution of the system due to docking ports located on either end of the Station. Future updates can include multiple Stations linked together to create a single location for the storage of large amounts of fuel. Stations can also be located in various orbits around Earth to decrease the distance a Tug has to travel to its destination, therefore saving time and money. Lastly, the Station will be outfitted with solar panels that provide power for tasks such as internal diagnostics, reaction control gyros, and communications.

Technical Specifications

To determine technical specifications for our Station design, we first looked at the mass-to-orbit left over from our launch vehicle. Subtracting out the fully-fueled Tug mass and claw mass gave us approximately 42,295 kg of mass left to design the Station system.

Since the Station's main purpose is a repository for fuel, we desired to focus most of the mass budget for storing Argon gas. We defined the tank size on the station to be capable of storing enough Argon for at least 20 refuels of the Tug. Multiplying the mass of propellant by 20 gives a station propellant minimum mass of about 24,100 kg. While this is a heavy load of Argon, the unpressurized volume using a density of 1633 kg/m^3 [15] is 14.7533 m^3. To hold the Argon, we desire a multi-tank system so that in the event of tank failure, the Tug can continue to refuel.

Using an off-the-shelf tank capable of storing Argon from Orbital ATK [16] gives a maximum tank volume of .9376 m^3 per tank. To reach our minimum desired amount of fuel, we need approximately 16 tanks. Since these numbers do not divide into integers directly, the updated mass of fuel for our Station using the ATK tanks is about 25,000 kg.

In order to estimate the mass of the Station, a representative model was created in SolidWorks with the structure and skin made from 2024 Aluminum Alloy. The mass value gathered from the structure from SolidWorks mass analysis is 2017 kg. This is a rough estimate, as the structure would not be made entirely of an aluminum alloy in reality.

To estimate our battery and solar panel mass, we estimate that the max power draw for our Station would be about 400 kW of the Tug due to two large control moment gyroscopes and ancillary power requirements. Using the same methods and materials for the batteries and the solar panels as the Tug, we found the mass of each respectively to be 756 kg and 125 kg.

One final design that we need for our Station system is the arm devices. We utilize existing LEEs [19] from the Canadarm2 module to grab the claw system and incoming cargo modules. We also estimate that the mass for each arm is a function of its carrying capacity. For this approximation, we took the mass of the Canadarm2 defined as 1,800 kg and its max payload capabilities of 116,000 kg [19] to create a function. We expect our arm to handle 25,000 kg as a max payload capability. Sizing based off of Canadarm2 gives an arm mass value of about 388 kg per arm, for a total mass value of about 776 kg.

The following table represents all the mass values taken into consideration for the Station Module:

Table 8: Mass values of the Station

Variable	**Description**	**Value**	**Units**
$m_{p,hub}$	Propellant mass, Station	24,497.24	kg
$m_{t,hub}$	Total mass of the ATK tanks	508.02	kg
$m_{s,hub}$	Structure mass	2017.00	kg
$m_{b,hub}$	Battery mass	756.00	kg
$m_{sp,hub}$	Solar panel mass	125.00	kg
$m_{a,hub}$	Total mass of the robotic arms	775.86	kg
$m_{rcg,hub}$	Reaction control gyro mass, total	200	kg
$m_{tot,hub}$	Total mass of the Station	29,079.13	kg
m_{ex}	Excess mass from fully laden launch vehicle to achieve full propellant use to the required 800 km orbit	13,216.37	kg
$m_{tot,sys}$	System mass including tug, claw, and Station module	33,033.63	kg

*The nomenclature denotes the Sisyphus Station with the subscript "hub"

The excess mass from the launch vehicle is 13,216.37 kg, which takes into consideration the Tug, Station, and the claw system. Modifying steps found in the "Initial Launch" subsection can give the maximum altitude that a fully fueled configuration can reach for possible future system expansion. Modifying the payload total mass for the Falcon 9 Heavy to be 33,033.63 kg, backing out the ΔV, and solving for the orbital altitude gives a max altitude of about 2,900 km. This ability would allow for current launch vehicle technology to deliver a constellation of Sisyphus Station systems at varying radii up to 2,900 km, reducing the waiting time between missions and expanding the market capability.

Lunar Orbital Platform-Gateway Missions

The robotic space Tug is seen as an ideal solution to the issue of delivering supplies or even spacecraft to distant locations. Although this section refers to the planned Lunar Orbital Platform-Gateway, it could service any station located as far away as the Moon. As mentioned above, the Station will be refueled by capsules such as Cygnus or Dragon. However, these capsules could also be launched with supplies intended for the astronauts living aboard the LOP-G. Because the Tug has an androgynous docking mechanism, a secure connection could be established between the Tug and the capsule carrying supplies. Additionally, because standard capsules do not have the ability to travel great distances, the robotic Tug would immediately allow the delivery of thousands of kilograms of payload to the LOP-G after a routine docking procedure. This would help sustain a long-term presence on the station and mitigate the risk of human casualties because both the capsule and Tug would be controlled remotely from Earth.

Additionally, in unusual circumstances, the Tug could be used to connect to other spacecraft, whether humans are aboard or not, to push them towards the LOP-G or back to Earth. This assumes that the Tug docking mechanism is compatible to dock with the target spacecraft. It would be in this way that the robotic Tug acts as a true "tug boat" in space.

Communications

A major mission requirement for the Sisyphus Tug will be the necessity of a continual ability to communicate with the satellite. This is due to the long maneuvers that must be performed with the ion engines and the fact that the Tug will have to operate in Lunar orbit. Fortunately, with the continued evolution of the private space industry there will be an opportunity for commercial cooperation. In 2019, Vodafone and Nokia are expected to test one such network in support of a rover mission, and as NASA progresses with plans for the LOP-G, they recognize the need for such a system. This means that it will be inevitable that some such systems will be put into place. One of the most advanced commercial efforts come from the company Atlas Space Operations. We would pursue a partnership with Atlas Space Operations who are planning on having a 100 percent available communications link that would be available for both Earth and Lunar Operations within the timeframe we would expect to for the Tug project to be operational [17].

The Atlas Space Operations system will consist of ground-based sites and satellites that will make Lunar and Low-Earth communication possible. Figure 13 below shows a map of the Earth-based sites where Atlas Space Operations will function [17]. These sites vary in capability; however, the Atlas system will provide 100 percent coverage at all times for UHF, S, and X band communication.

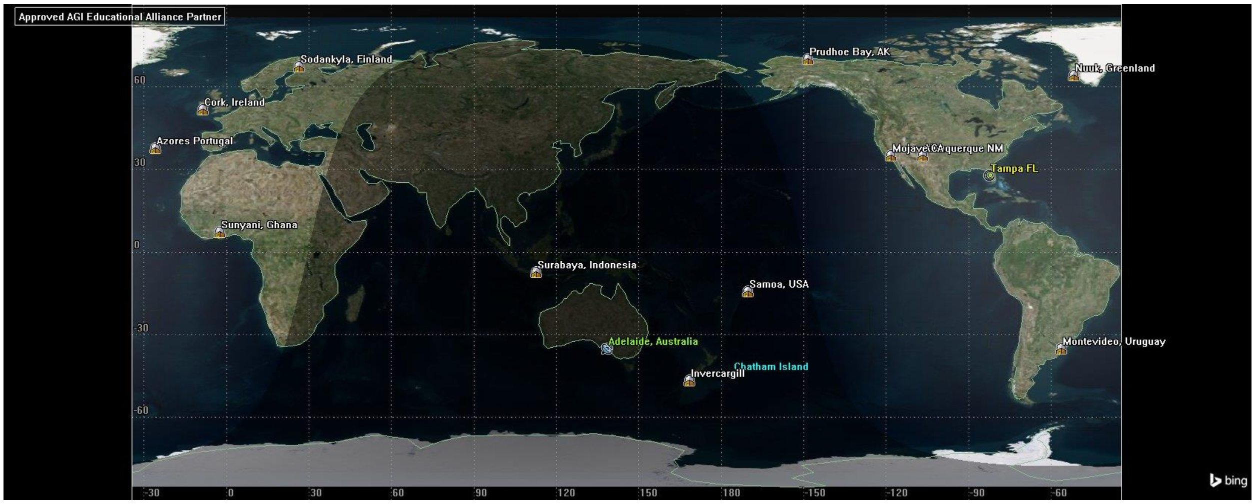

Figure 13: Locations of Atlas Space Operations Satellites

These Earth-based sites will communicate with the company's HALO satellite network, which includes low-Earth and Lunar communication satellites. As discussed above, there is a range of radio frequencies which would be available for communication and data transmission, including UHF, S, and X band communication, with plans to expand to more frequencies in the future. These frequencies are used to communicate from the ground station to ATLAS satellites in LEO. We could have an operating center which relays commands through this network for our satellite missions. The Tug will have very low operating requirements because the only information we need to relay to the satellite are maneuvers. There will be no need for the transmission of any science or other collected data. Additionally, ATLAS will offer launch assistance and management and control software (mission control assistance) if the STSS requires it. This may be useful, although we would not be able to tell at this preliminary stage of such a mission design. Following traditional protocols for satellite design, the Tug will be equipped with a primary high-gain antenna as well as a backup low-gain antenna for redundancy. This method of satellite communication will be in line with our mission priorities of minimizing cost and adding flexibility. It also provides us with necessary ground and space equipment with the added benefit of the flexibility of choosing to add additional services or build our own infrastructure at a later date. Lastly, it removes an enormous amount of up-front costs. We anticipate low operating costs with ATLAS due to our Tug requiring only very basic capabilities.

Robotic Arms Modification

The team cycled through several ideas regarding how the robotic Tug would control and move objects in orbit. Many of these methods were only theoretical or not yet widely used, including a net, tether, drag sail, and targeting the object with high-velocity gases. Ultimately, it was decided that robotic arms with a means to grip objects would be used as the primary means of moving objects.

Robotic arms were chosen because they are a successfully proven method, do not require refueling or reloading, and can be operated by professionals on the ground. Additionally, each robotic arm will be equipped with various tools, such as lights, cameras, and advanced sensors. This modification will have two robotic arms, and can therefore use both arms to grip the debris or satellite object.

When the Tug moves an object, there are two possible objectives: move the object to a different orbit for future use or move the object to prevent it from becoming space debris. To address the issue of space debris, the Tug can move the object to a “dead orbit” far outside the useful space around Earth, or it can grab the object and perform a burn into a circular orbit close to Earth’s atmosphere. The Tug can then release the object and allow Earth’s atmosphere to capture and deorbit debris much faster than if the object had not been moved. Unfortunately, a trade-off must be taken into account. The closer the Tug gets to Earth, the faster the objects deorbit time. However, this type of maneuver will consume more fuel when the Tug must circularize its orbit again. Additionally, the Tug does not have any type of heat shielding, so caution must be taken to not allow the Tug to fly too deep into the atmosphere.

The robotic Tug arms were based off the Canadarm 2 and Dextre models. Both are currently used on the International Space Station and fall in a range of size and performance requirements that were comparable to the objectives of this design competition. The Space Shuttle’s robotic arm, the original Canadarm, was too large to consider for this design.

Structural analysis was performed to determine the maximum weight that these robotic arms could carry. Both a two-beam and three-beam case were analyzed. The Canadarm 2 uses two beams, while Dextre uses multiple smaller arms comprised of both two and three beams. The approach was to model the situation as a simple beam bending problem. In Figures 14 and 16, a force P can be seen applied to the very end of the robotic arm system. This represents the weight of the object to be carried. To prevent an overly complex problem, it was assumed for this analysis that the angle between the robotic arms and the weight of the object is 90°, the cross-sectional area of the beams is uniform and circular, and the radius of each beam is constant. The Canadarm 2's radius of 0.175 meters was used in this analysis. A distributed load q was also incorporated to take the weight of the arms themselves into account. Figures 15 and 17 further show how the beam system is broken down to represent internal forces. The distributed loads q, in N/m, become a single point load qL when multiplied by the length of the beam. Their location is in the middle of each beam, at a distance of L/2. For Figures 14, 15, 16, and 17, note that red arrows denote external forces, while blue arrows denote internal reactionary forces.

Both the two-beam and three-beam cases are solved by breaking down each case by section, splitting them at the joint. Once the problem setup is complete, the following formulas are used:

$$\begin{gathered} \Sigma F_x = 0 \\ \Sigma F_y = 0 \\ \Sigma M = 0 \end{gathered} \tag{18}$$

The moment equation is only used in the first (left) section because it is the only section with a reactionary moment. After a few algebraic substitutions, the achieved result is an equation for the reactionary force at the wall, R_y. The equation for the two-beam case is

$$R_y = 2qL + P \tag{19}$$

while the equation for the three-beam case is

$$R_y = 3qL + P \tag{20}$$

By finding R_y, one can use the cross-sectional area of the beam to determine a wall stress. This stress can be compared to the yield stress of the robotic arm system material to determine when failure will occur.

Although not readily apparent, the equation for R_y contains variables that are easy to work with. The quantity of L is to be optimized, the quantity of q can be calculated as a constant, and P comes from the formula

$$F = ma \tag{21}$$

or

$$P = mg \tag{22}$$

The force F is represented by P and the acceleration term is g, the acceleration due to gravity. In the structural analysis code, g was taken as 90% of g at sea level, or approximately 8.83, due to the robotic Tug being located primarily in LEO.

The distributed load q is more complicated to determine, but the result works well for this analysis. Taking the equations for density and weight,

$$\rho = \frac{m}{V} \tag{23}$$
$$W = mg \tag{24}$$

one can substitute and rearrange to obtain

$$W = \rho V g \tag{25}$$

where V represents volume. Since the definition of a distributed load is weight per length, then q = $\frac{W}{L}$ and

$$q = \frac{\rho V g}{L} \tag{26}$$

by substitution. Lastly, because the beams were modeled as uniform cylinders, V = $\pi r^2 L$. Due to the L term in the denominator, the length cancels out. Therefore, because r is assumed to be constant in this analysis, q is constant for all cases and can be determined from the equation

$$q = \rho \pi r^2 g \tag{27}$$

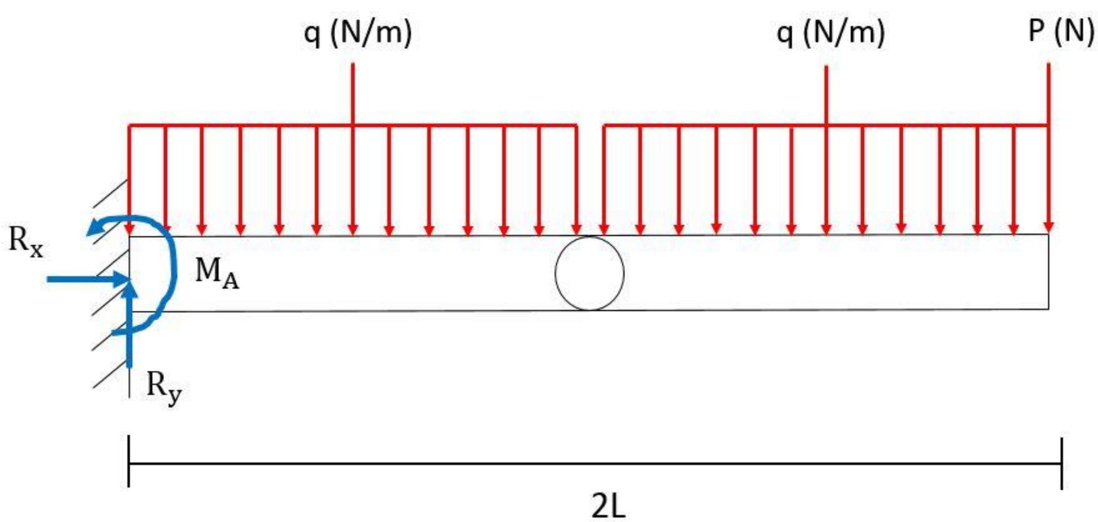

Figure 14: External and reaction forces for the robotic arm two-beam case

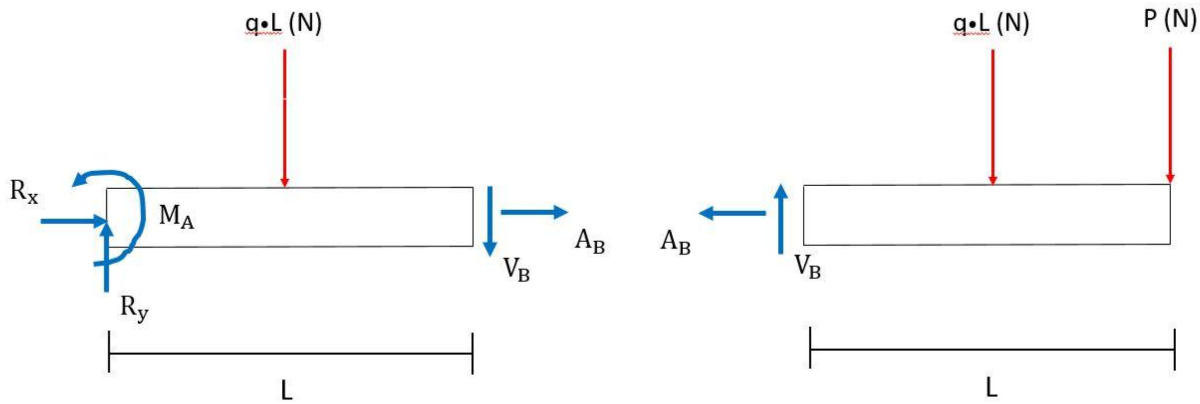

Figure 15. Section-by-section analysis of the robotic arm two-beam case

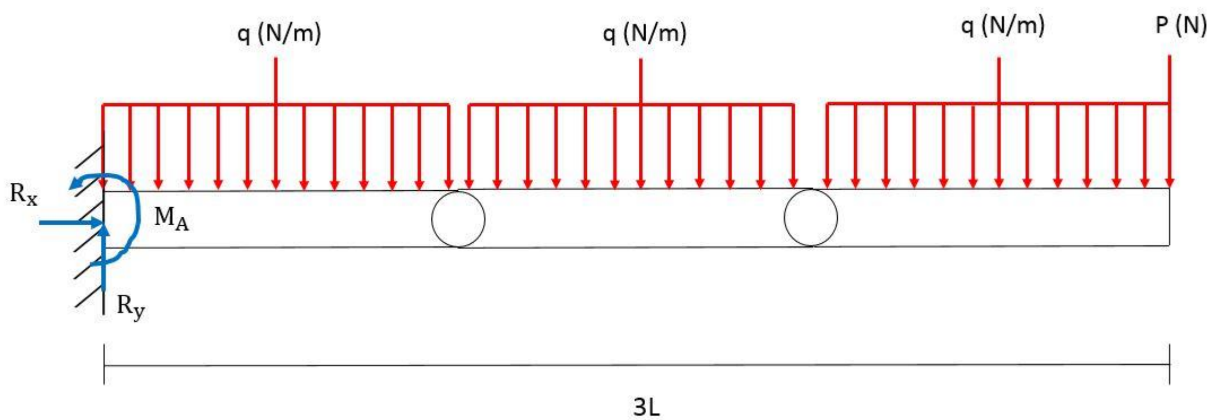

Figure 16. External and reaction forces for the robotic arm three-beam case

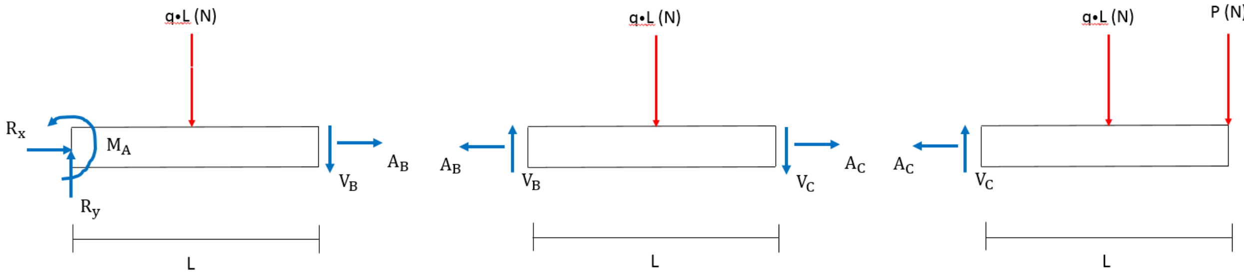

Figure 17. Section-by-section analysis of the robotic arm three-beam case

The final information needed before performing the structural analysis of the beams was the properties of the material. Like both Canadarm models, the team envisioned the Tug robotic arms being fabricated out of a carbon-fiber composite material. When performing analysis with metals, the yield strength is a known quantity and is generally constant for two or more sheets of the same metal. Carbon-fiber composites, however, have similar but varying material properties based on the number of ply sheets and direction of the fibers, among other factors. For this analysis, an approximate yield strength of 200 psi (1.38 MPa) and approximate density of 0.300 g/cc (300 kg/m^3) was used [20]. Because the cross-section of each beam is circular, and the radius is assumed to be constant at 0.175 meters, the yield stress at the wall was determined by dividing the results of R_y by πr^2. Recall that R_y changes as the application of the load P changes. This result was divided by a safety factor of 2.0 and compared to the 200-psi yield strength to determine the maximum allowable load. Although a safety factor of 2.0 can be considered unnecessarily high in many aerospace engineering cases, it was justifiable in this situation due to the uncertainty of the carbon-fiber yield strength and the mission criticality of the Tug being able to support the weight of the target object.

The results for this analysis can be observed in Figures 18 and 20. It can be seen in these figures, and inferred from the equations for R_y above, that a decreased beam length produces a decreased reaction force at the wall when the mass of the object is constant. Therefore, the minimum length of 1 meter per beam was chosen for the robotic arms modification. The team felt justified in this value because the modification would have at least two beams for each arm. A minimum distance of two meters between the object and the robotic Tug was judged to be a reasonable separation.

Figures 19 and 21 are cross-sections of Figures 18 and 20 at a length of 1 meter per beam. By finding the intersection between the stress at the wall for the two and three-beam cases and the yield strength of the carbon-fiber composite, the maximum allowable mass can be determined. These results are summarized below in Table 9.

Table 9: Maximum allowable object mass for the two and three-beam cases

	Maximum allowable mass to be carried
Two-beam case (single arm)	7,455.9 kg
Three-beam case (single arm)	7,427.0 kg
Two-beam case (two arms)	14,911.8 kg
Three-beam case (two arms)	14,854.0 kg

As shown in Table 9, the use of three beams on the robotic arm does not significantly decrease the maximum allowable mass. The margins between mission success and failure should not be 30 or so kilograms, meaning that both the two and three-beam cases are feasible in theory. However, the robotic Tug will use the two-beam case because it can carry slightly more mass and a robotic arm with three beams would be more expensive to manufacture. Like the Canadarm 2, each arm will have 7 degrees of freedom. The connection at the Tug will have 3, the elbow joint will have 1, and the joint that attaches to the claw will have 3.

Also shown in Table 9 are the maximum masses with two robotic arms. These numbers are simply doubled from the single arm analysis. Because the Tug will use two robotic arms with two beams on each arm, the maximum allowable object mass that can be carried is 14,911.8 kg, or approximately 14,900 kg. This covers a wide range of possible targets, including the Hubble Space Telescope at approximately 12,200 kg and the future James Webb Space Telescope at approximately 6,200 kg [21] [22]. Satellites such as these could be moved to different orbits to capture better images or establish more direct lines of communication. Of course, the more mass the Tug carries, the more fuel the Tug will consume as it makes a burn to its next destination. This would decrease the Tug's mission time away from the Station and require more frequent deliveries of fuel to the Station.

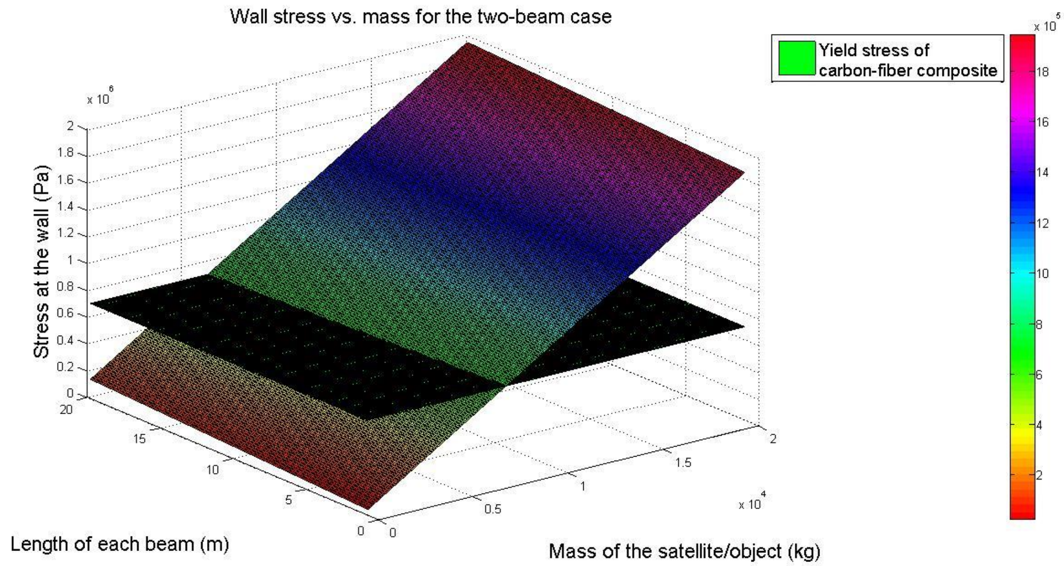

Figure 18: Wall stress vs. length vs. mass for the two-beam case

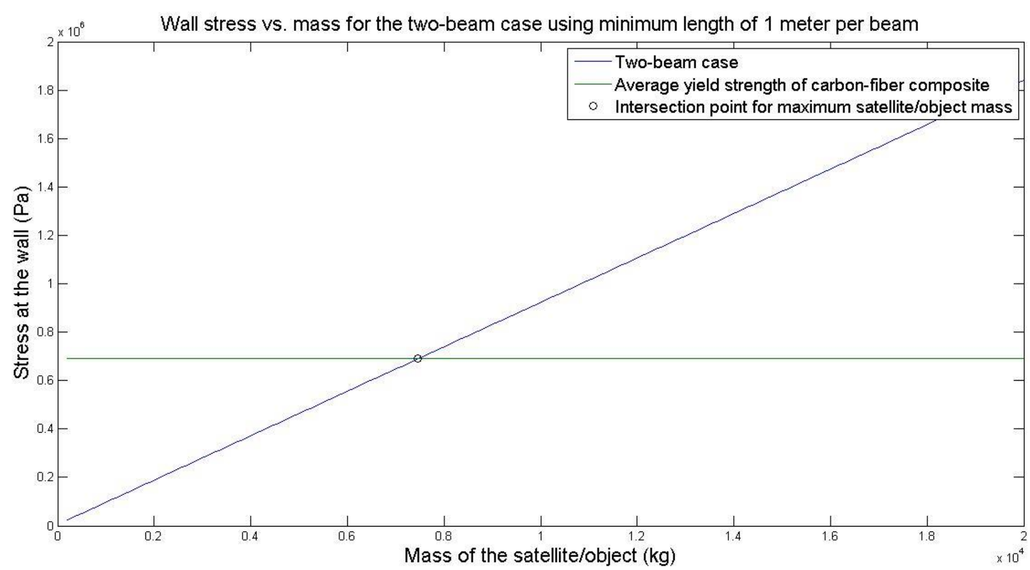

Figure 19: Wall stress vs. mass for the two-beam case using minimum length of 1 meter per beam

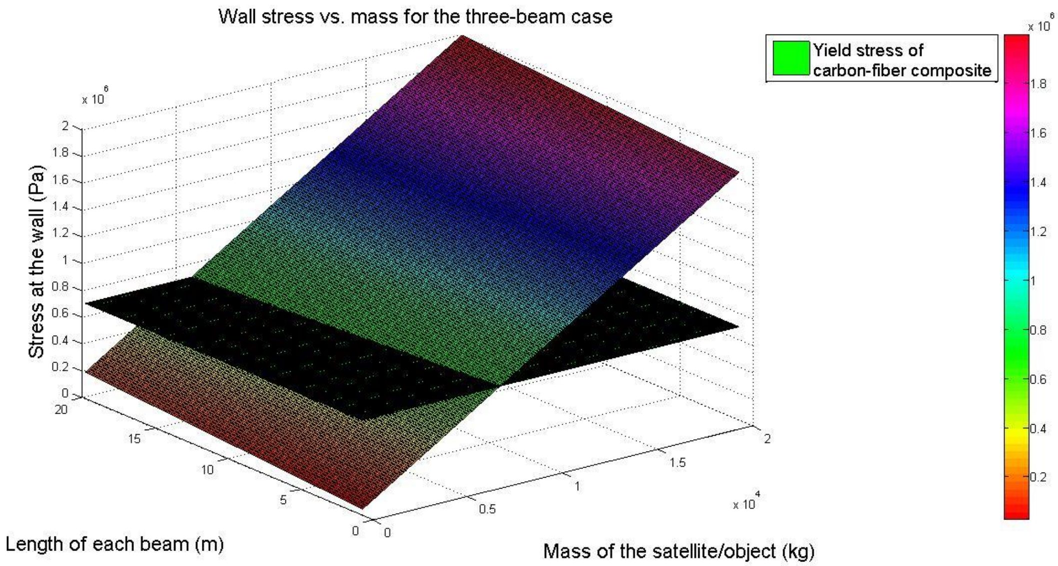

Figure 20: Wall stress vs. length vs. mass for the three-beam case

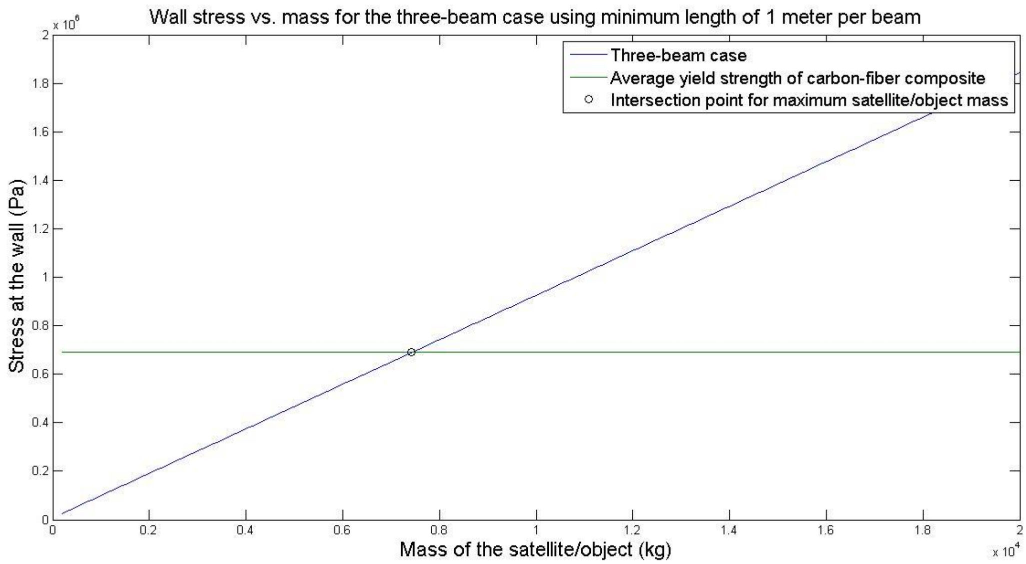

Figure 21: Wall stress vs. mass for the three-beam case using minimum length of 1 meter per beam

The final structural analysis completed for the robotic arms modification regarded buckling. As mentioned above, the beam bending analysis assumed that the satellite/object to be carried would be held perpendicular to the robotic arm, therefore creating no compressive force parallel to the beam. For the buckling analysis, it was believed the worst-case scenario would come from the possibility that the robotic Tug could approach an object traveling too fast, therefore impacting the object with enough force to cause buckling. Using the simple formula F = ma and Euler's critical load formula for buckling,

$$P_{crit} = \frac{\pi^2 EI}{L_{eff}^2} \tag{28}$$

where L_{eff} = 2L for the fixed-free case, one can determine the acceleration at which an impact would cause buckling of the robotic arms. The area moment of inertia, I, is based on the cross-section of the beam. Because the radius is constant at 0.175 meters, the term I is calculated from

$$I = \frac{\pi r^4}{4} \tag{29}$$

The term E is a material property and was taken as 14,200 psi [20]. The total mass of the Tug was taken as 3954.50 kg, which includes a worst-case scenario of the base Tug structure, solar panels, batteries, a full tank of argon gas, and the robotic arms modification attachment. An approximation of 250 kg was used for the robotic arms modification. This number is derived from a 115 kg total weight of the arms plus an estimate of the base structure to which they are attached. A full breakdown of the Tug mass can be seen in Table 5 in the "Additional Mass Considerations and Final Values" subsection. Additionally, a safety factor of 1.5 was used to account for assumptions in mass and elastic modulus, as well as ensure the robotic arms can survive most small impacts. The results can be seen below in Figure 22, where the maximum accelerations at impact to prevent buckling for the two and three-beam cases are approximately 7.5 m/s^2 and 3.3 m/s^2, respectively. This also lends support to the fact that the robotic arms modification uses the two-beam case due to its ability to withstand a greater impact before buckling.

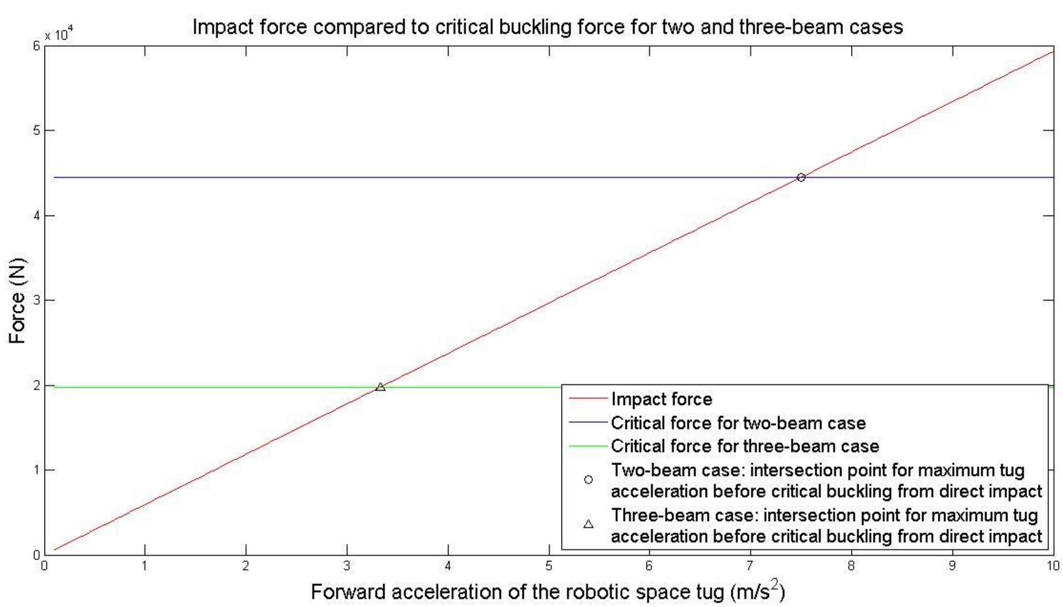

Figure 22: Impact force compared to critical buckling force for two and three-beam cases

It should be noted that the two and three-beam cases were analyzed as a single continuous beam for this approximation. Realistically, beams connected by joints require more complex analysis to determine buckling force.

Lastly, this approximate analysis can be compared to the Euler buckling formula for critical stress,

$$\sigma_{crit} = \frac{\pi^2 E}{\left(\frac{L_{eff}}{r}\right)^2} \tag{30}$$

where L_{eff}/r is the slenderness ratio. As described above, the effective length L_{eff} is 4 meters and the radius is 0.175 meters. The slenderness ratio of this two-beam system is 22.86. Plugging this number into the equation for critical buckling stress, the result is 1.85 x 10^6 Pa. However, the yield stress is approximated as 1.379 x 10^6 Pa. Therefore, the robotic arm column would yield before it buckles due to increasing stress. This aligns with the theoretical background of yield and buckling because shorter, wider columns generally yield first and longer, narrower columns generally buckle first. Since the beams used here in the robotic arms modification are approximately intermediate columns, it makes sense that the numbers for yield and buckling stress are relatively close.

Sample Mission Analysis

The STK models were designed to reflect some of the most extreme missions that the Sisyphus Tug might undertake. These simulations were primarily designed to confirm that the calculations done in Python approximately correspond with the simulation results. If these two methods of calculating the Tug's performance are reasonably similar, then we can be confident that we have good estimates of the system's performance. The first computer simulation reflected the case where the Tug must rendezvous and deorbit the satellite similar to the size of the GOES-15 satellite. This satellite was chosen because it represents the upper limit of what our system would try to target. The GOES-15 satellite is a large satellite with a dry mass of 1540 kg and it is a geostationary satellite. In our scenario we are assuming that the intercept with the target satellite is coplanar, so we positioned the target satellite at the same inclination as the Sisyphus Station. However, this assumption is only made to simplify our analysis, we are accounting for various orbital patterns. Our assumption here is even offset by the decision to deorbit a geostationary satellite which is unreasonable and will add a layer of extra complexity that ensures the system has operational flexibility. A satellite like the GOES-15 would be moved to a graveyard orbit.

STK allows us to add a few elements of the spacecraft that will improve the accuracy of this model. We can accurately model the starting fuel mass and the dry mass of the Tug and we can update that fuel mass continually while it is used. Additionally, STK will allow us to factor in J2 orbital perturbations for our simulation. The STK program uses a generic ion propulsion engine, which is one element where we would have to be aware that it is a potential error source. The other assumption that is being made in this simulation is that the rendezvous with the target satellite is made at an optimal time in the two objects' orbits so that the rendezvous is simple with negligible ΔV. This may seem like a large assumption, but when we consider that our system operates with very long burn times, we are operating with the understanding that waiting for a point where all costs are minimized is the preferred commodity over speed. The transfers to and from the geostationary satellite are modeled with two Hohmann transfer burns which account for the total ΔV budget. In total, the mission to the target satellite requires a ΔV of 7.592 km/s. When we compare this to our mathematical calculations we see that the results are very similar with a result of 7.9 km/s. With our knowledge of the assumptions we made in each case, these results are satisfying as confirmation that our methods are adequate predictors of the system's behavior.

The second simulation that we modelled was the theoretical mission that would provide a resupply capsule with transport to the LOP-G. The starting criteria for this simulation was the Tug at its parking orbit stationed on the Station at an orbital altitude of 800 km. That starting point is chosen because the resupply capsule would have rendezvoused with the Sisyphus Station where it would be transferred to the Tug. From there we engage in a lunar transfer orbit with B-plane targeting with the intention of trying to achieve an orbit at an altitude of 70,000 kilometers, which is the current estimate for the location of the LOP-G. B-Plane targeting is just a reference to a trajectory plotting which we used to insert into a specific Lunar orbit. Just like the previous scenario, there are important metrics that we had to input to the STK model to increase the simulations fidelity. These inputs are the same as the previous scenario and give us an extra level of precision in our model. Below is a top-down view of the Lunar insertion and capture of the Tug.

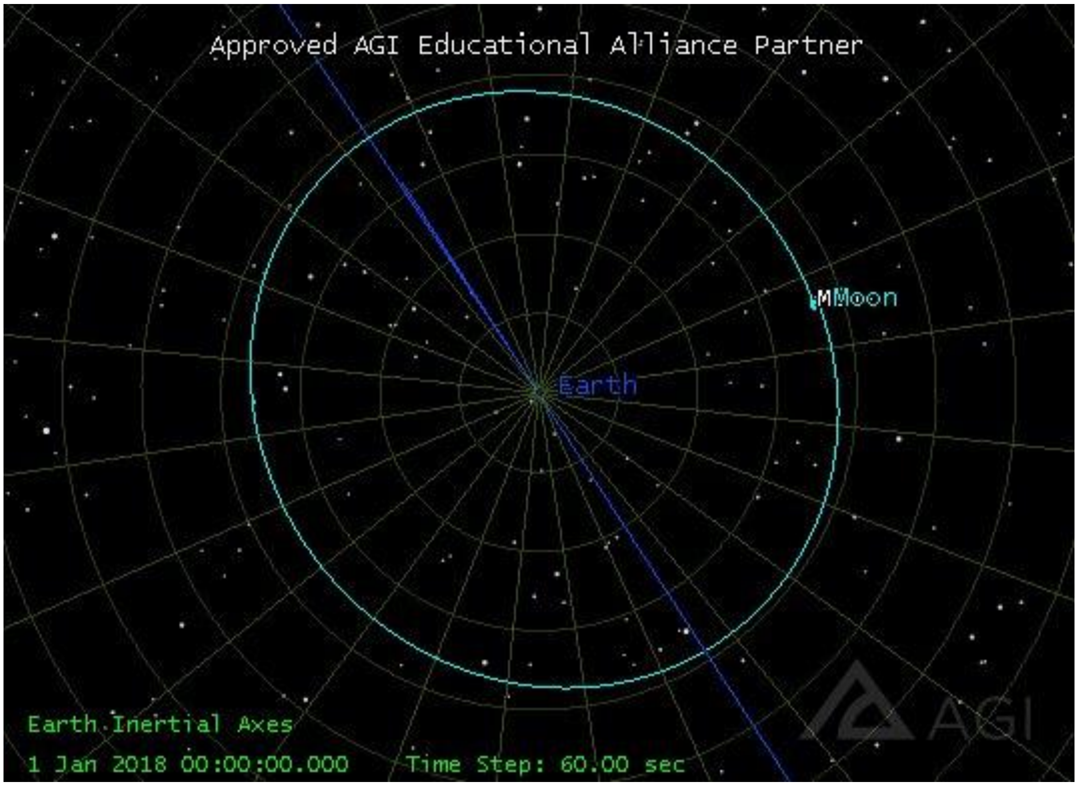

Figure 23: Geocentric Lunar Orbit

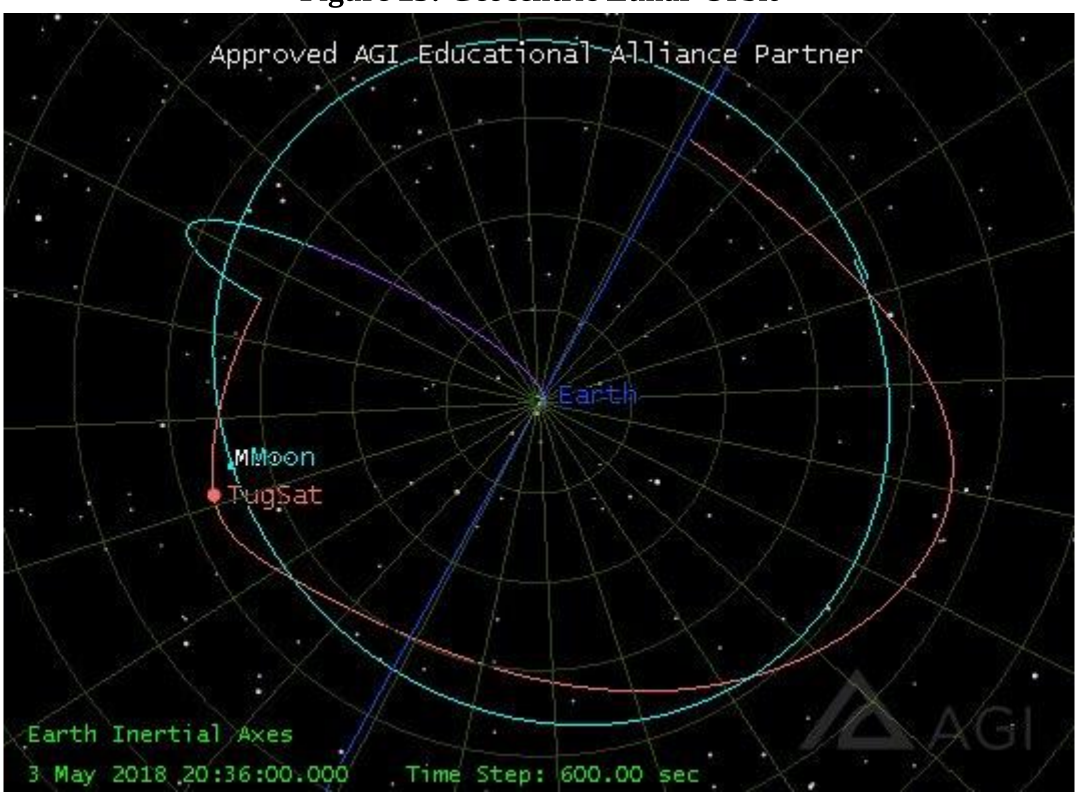

Figure 24: Geocentric Tug Orbital Insertion

The simulation shows the successful transfer of the Tug to the moon, which, according to STK, used a total of 4.626 km/s ΔV. This simulation does not include the return to LEO because it is unclear whether there would be a strategy to let the Tug remain in Lunar orbit for future missions. However, we must again compare this result to our mathematical calculations. In this case our analytical calculations showed that the ΔV cost would only be 3.522 km/s. This difference is much larger than the previous test, but this was expected due to the type of mission we were performing and our understanding of the assumptions that we made. In this case, our analytical solution was simplified because of the assumption that it would be the exact optimal moment to perform a Lunar insertion. This meant that the transfer would be coplanar and that the LOP-G orbit was also coplanar. In the STK model, we used the B-plane target to give the final Lunar orbit some inclination, and we did not initiate the mission at a point where the transfer would be coplanar. With this knowledge, cargo mass can be changed to allow for non-ideal LOP-G rendezvous cases. For this case, the cargo mass will have to decrease from 3500 kg to 1500 kg. This decrease in mass size can be mitigated in future iterations of this design with a larger fuel tank.

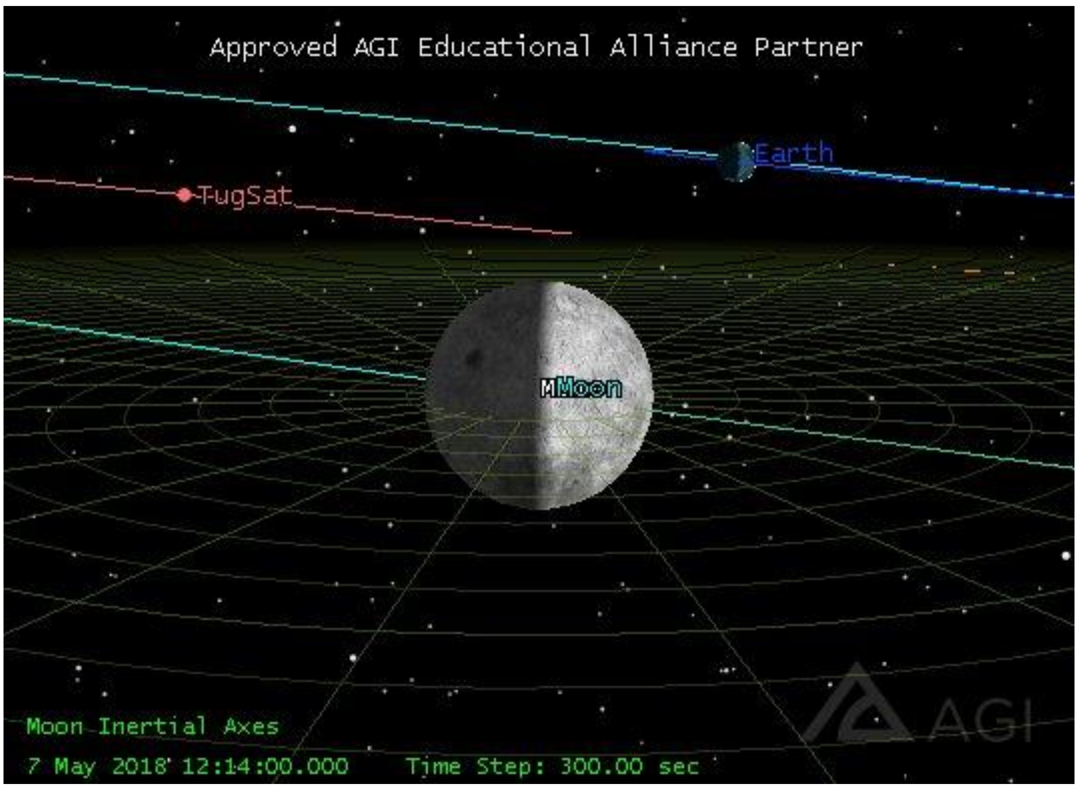

Figure 25: Closeup showing Inclination of the Tug and the Moon

Cost Analysis

The cost breakdown shown below in Table 10 is intended to be the minimum cost for the initial launch with minimum capability. In other words, Table 10 shows the cost to launch the Sisyphus

Tug and Station system for the first time with only one full tank of propellant. The cost to launch additional fuel tanks as well as refueling missions is approximated below. This is a rough estimate for the cost to demonstrate our capabilities real time and fixed investment costs, excluding labor.

The initial startup costs are summarized in the table below. The launch cost was obtained by using the Falcon Heavy pricing page. The current estimated assumed a flat rate per launch of $90M, however since we will be using more frequent launches for refueling, the price may go down for bulk or long-term contracts with SpaceX [23]. The VASIMR Engine was the most speculative as the engine had not been produced yet. However, looking at NASA’s contracts for developing such an engine, a rough approximation can be made, totaling around $10M [24]. The cost of the robotic arms was based on the cost of the Canadarm system, roughly $100M each [25]. Solar Panel pricing can vary from mission to mission based on how the solar cells are laid out. A general estimate was made from the total cost of the International Space Station’s solar arrays, divided by the mass of the solar arrays. This produces the cost of the arrays per kilogram, multiplying this by the mass of our solar arrays should produce a rough estimate for the total cost, approximately $360,000 [26]. The material cost of the system was approximated of it all being aluminum 2024, commonly used for aerospace systems, and multiplying by the cost per kilogram. However, this would be the raw cost and not the manufactured cost, a safety factor of 2 was used to cover manufacturing costs, resulting in a price of $56,000 [27]. However, this very dependent on the machining process and certain parts may cost significantly more for the same mass due to shape constraints. The price of argon was found and converted to kilograms using the ideal gas law, resulting in a total cost of $260,000 [28]. The cost of batteries is estimated by taking the cost per kilogram of an individual battery and multiplying by the total mass of the batteries, resulting in a total cost of $562,500 [29]. Summing all the costs together and the total initial cost of the system arises to be roughly $300.7M.

Table 10: Breakdown of approximate fabrication and launch costs for initial launch

Item	Amount	Units	Price/Unit	$/Unit	Total Price
Falcon Heavy Launch	1	Launch	90,000,000	$/launch	$ 90,000,000.00
VASIMR Engine	1	Engine	10,000,000	$/engine	$ 10,000,000.00
Robotic Arm	2	Arms	100,000,000	$/arm	$ 200,000,000.00
Solar Panels	300	kg	1,200	$/kg	$ 360,000.00
Aluminum 2024	3500	kg	16	$/kg	$ 56,000.00
Argon Gas	26000	kg	10	$/kg	$ 260,000.00
Batteries	3000	kg	188	$/kg	$ 562,500.00
Total Cost	--	--	--	--	$ 300,676,000.00

Once the system is in orbit, the only recurring costs would be the cost of argon and the launch vehicle required for the fuel. Since the fuel mass would be less than the total mass of the system, it is possible to launch with a lighter launch system, driving the operating price even lower. We expect intermittent refueling of our Station, so the mass of argon is not a full Station amount. The operating costs are expected to be around $62.3M as seen in the Table 11. Labor, design and other smaller costs were not included in this calculation as they did not seem to have a large enough effect on the total cost.

Table 11: Running Costs for Refueling Tug

Item	Amount	Units	Price/Unit	$/Unit	Total Price
Falcon 9 Launch	1	Launch	62,000,000	$/launch	$ 62,000,000.00
Argon Gas	15000	kg	10	$/kg	$ 150,000.00
Total Cost					$ 62,150,000.00

Market for Services

Observing the capabilities of our Tug, there are three potential markets that we can break into: Satellite repositioning, servicing the LOP-G, and CubeSat deployment. In 2016, the satellite industry revenue was $260.5 billion, meaning niche companies can focus on small, underdeveloped parts of the market and still make a large profit due to the large industry [30]. Although our product doesn't fit into the predefined satellite markets, like launch services, ground equipment, etc., our product aids and improves these satellite markets, and does not pertain to a specific one. This is especially useful in case one market experiences revenue loss, but the entire market continues to grow. Being able to work in multiple markets allows us to be free of dependence on one aspect of space and focus on the entire picture. Each part of the space market we can work with are discussed below.

Satellite Repositioning

Mission lifetime is a critical part of satellite design, as it determines how useful and cost efficient a satellite can be. Some satellites do not need dedicated engines and only use them to achieve their desired orbit while using smaller thrusters or attitude control systems for station keeping. There are currently 1,738 active satellites in orbit, with 2,897 acting as space debris or waste [31]. Most of these satellites are either communication or Earth observation satellites. One of the proposed ways to mitigate space junk production is the use of a graveyard orbit at the end of the mission life. Additionally, occasional launches do not go as expected and the final orbits of some satellites are different than originally planned. The ability to move the spacecraft back to its intended orbit would be a very useful function. Currently, Orbital ATK has the main proposal for moving satellites with their Mission Extension Vehicle (MEV). The MEV attaches its claw to the engine nozzle or launch adaptors of the targeted spacecraft and moves it to the desired orbit.

Along with their MEV, they will also supply the Mission Extension Pod (MEP), which will aid in orbit control of geostationary satellites, and a Mission Robotic Vehicle (MRV) which can attach MEP's to satellites as well as perform basic repairs and inspections [32]. This system would be our top competitor in this market for satellite repositioning, but they currently lack the ability to maneuver any orbital object that does not have a nozzle or launch adaptor.

Large satellite design by itself is complicated and requires many tradeoffs and analyses. The inclusion of propulsion systems on board makes the satellite more complex, expensive, and prone to increased risk of failure. Propulsion systems also require a considerable fuel and dead mass for the engine and tanks. Therefore, the addition of a propulsion system greatly complicates the design and is only implemented if necessary. Current space etiquette requires dead satellites to either deorbit themselves or move to a graveyard orbit. This kind of maneuver would require a propulsion system on board. On a similar note, extending the design life of a spacecraft also requires engines. For low earth orbiting satellites, like Earth imaging satellites, the design life is highly dependent on the atmospheric drag and altitude. Without onboard propulsion, the satellite could maintain its operation but would have to be decommissioned by reentering the Earth's atmosphere. We do not have an accurate model for the Earth's upper atmosphere and determining the effects of drag on the orbit can become complicated. Allowing for mission lifetime extension in case of rapid orbital decay from our system is a great benefit to all future and current LEO missions.

With our Tug system we could reposition satellites and have our efficient thrusters move customer satellites, so they would not need to include a propulsion system. The Tug can grab onto a wide variety of satellites in orbit and maneuver them using efficient ion engines. Our customers would have all the benefits of a propulsion system such as mission life extension, orbit assurance, and the ability to change missions and move to graveyard orbits without the added mass, costs, and complexities of using their own engine. Using our Tug would lighten the load of designing large satellites and save our customers money in the long run.

Larger satellite companies that deploy heavy satellites into higher orbits would likely buy into this program to ensure that their spacecraft reach the correct orbit regardless of propulsion or launch failures. Government organizations with the same plans may also be interested in our product. Especially with communication networks, it's possible that one node could fail and hinder the entire network. In 2016, satellite services, such as satellite TV and broadband, accounted for 127.7 billion in revenue [30]. Having adaptability and reliability in the network is extremely beneficial to these services considering the effects of a satellite being in the wrong orbit. With the Tug, it would be possible to move satellites without large engines to accommodate for delays and to repurpose the satellite's mission, thus allowing for better reliability and adaptability. These are just a few cases but having the ability to move satellites without worrying about fuel or rocket engines is extremely useful and can be applied to most situations. At its core, any large organization with heavy satellites would benefit from and be interested in our product.

LOP-G Servicing

The need for a heavy lift space tug system has significantly grown with the announcement of NASA's plan for the Lunar Orbital Platform-Gateway (LOP-G). Having a space station orbiting the moon requires continual supply missions and transportation to the station. This project was proposed by the President to Congress with a dedicated commercial launch of the Power and Propulsion Element (PPE) in 2022 [33]. Although there are no formal competitors, they will likely be the usual suspects, such as Lockheed Martin, Airbus, the European Space Agency, Orbital ATK, and others. These companies have previously proposed project ideas like the Jupiter Exoliner, but no further plans have currently been announced.

Servicing, transport, and resupply of the LOP-G is out of reach of most rockets and space modules due to the lack of a need to reach the Moon. Because most space operations currently deal with orbits in a range from Low Earth Orbit to Geostationary Orbit, many spacecrafts were not designed to transfer cargo to the Moon. Similar to satellite repositioning, if the satellite needed to be moved to the Moon, it would need its own propulsion system and may not be able to carry a significant amount of cargo.

However, since our vehicle is capable of both heavy lifting and transport to the Moon, the Sisyphus Tug is an ideal option for NASA to resupply and transport cargo to the LOP-G. After a customer launches their supplies to orbit, our Tug would be able to take the cargo and transport it out to the Moon using slow but efficient ion engines. This may take more time than a conventional engine, but the cost savings would be significant.

Our customer for this service would be NASA, as they would need to resupply food, fuel, and supplies to the LOP-G. Ideally, NASA would purchase this service for our cheap transfer costs and ability to reach lunar orbit from Earth. Although NASA would be the only customer for this mission, they will have a consistent need for this service, similar to the resupply missions that Cygnus and Dragon perform to the International Space Station. Initial missions of repositioning satellites would demonstrate our system's reliability, and hopefully secure the contract to transport cargo to the Moon.

CubeSat Deployment

There are currently three primary means to launch a CubeSat to orbit: buy your own rocket, rideshare to a specific orbit, or rideshare to the International Space Station. This could be a market that we can enter given little modification. In 2019, there are expected to be 431 CubeSat deployments with 369 currently planned, with that number nearly doubling to an expected 703 CubeSats launched in 2023. Currently ISS launches account for 61% of the launches, with 34% going to Sun Synchronous and Polar. In the next three years this trend is expected to shift to SSO/Polar when more satellite deployment opportunities arise [30]. There are many companies willing to sell the use of their rockets, namely SpaceX, Orbital ATK, and Rocket Lab, among others. SpaceFlight's pricing for a 3U CubeSat is about $295,000 to LEO and $915,000 to GTO [34]. NanoRacks launch service from the ISS is about $85,000 per unit [35]. Spaceflight Inc's SHERPA has been proposed as a Space Tug system as a rideshare attachment on SpaceX's Falcon 9 to deploy CubeSats to their correct orbit, but this system has not been deployed or tested. For ridesharing and deployment from the International Space Station, NanoRacks is the primary solution. .

Although this method is much cheaper than the alternative, it lacks the infrastructure to deploy CubeSats to any other orbit besides that of the ISS. If a customer wanted a CubeSat to use a less popular orbit, or move to a higher altitude, there are currently no capabilities to do so. This may hinder the CubeSat's usefulness if it needs to observe certain parts of the Earth, operate at a higher altitude to avoid drag, or have a larger Earth observation angle.

With the ability of our Tug to efficiently move satellites to different orbits, we will be able to service this unexplored part of the market. Companies that desire specific orbits would have the opportunity to launch their CubeSats there, without having to buy their own rocket. Of course, the more customers that desire similar orbits would lower the cost, but the difference in our system is that they do not all have to be deployed in the same orbit. We could deploy CubeSats at any desired orbit before performing a burn to the next desired orbit. Opening this area of the market would allow companies to tailor their orbits to their specific mission, instead of having the orbit dictate their mission. Simply, CubeSats will be more useful and beneficial with the capability to choose their orbit in a cost-effective manner.

Lastly, these capabilities would be extremely useful for companies that make use of Earth observation, as a larger range of orbits equates to better mapping of the Earth. Some data-sensing companies may need orbits around the globe at different inclinations. Quite simply, as these satellites gain the ability to enter different orbits, many companies can offer more advanced and effective services. For example, Spire is a company that tracks shipments around the world. If they wanted their satellites to use highly elliptical orbits to stay above certain locations longer, or to extend mission life at higher altitudes, they could use our Tug system to achieve that.

Conclusions

The Sisyphus Tug-Station System concept provides an efficient means for governments and private space entities to access the evolving space economy. This system focuses on being a viable product in the initial development of the space economy, with the flexibility to evolve as the industry changes. The mission objectives that we chose highlight some of the most important missions in the early stages of the space economy. The functionality of the system can be drastically expanded in the future to include more STSS systems and specialized robotic arm modifications. The use of ion engines, the Sisyphus Station concept, and beneficial business partnerships will provide us with the market advantage needed to build a stable product and highlights our focus on economic efficiency and flexibility. By accessing the space economy, a new market can be created that benefits scientific research and provides the opportunity to gather new resources and sustain economic development. The STSS design provides an efficient and profitable option to expand satellite ranges of utility, remove potentially catastrophic space debris, and provide a workhorse for material transport for Earth and Lunar operations.

Appendix of Standard Values

Variable	Description	Value	Units
μ_{Earth}	Standard gravitational parameter for the Earth	3.986E5	km^3/s^2
ϕ_{KSC}	Latitude of Kennedy Space Center	28.5729	deg
r_{Moon}	Average distance of the Moon from the Earth	384,400	km
μ_{Moon}	Standard gravitational parameter for the Moon	4.903E3	km^3/s^2

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