Lunar Mass Driver Implementation

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by

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ABSTRACT

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After nearly five decades, mankind is planning to return to the Moon with the intent of setting up a colony. This colony could accomplish several important scientific studies and breakthroughs, but a lunar colony also presents the potential to harvest the Moon's natural resources. This paper expands on the work performed by a team of NASA scientists led by Dr. Gerard O'Neill who designed a lunar mass driver capable of transporting harvested lunar resources back to Earth. Specifically, this paper focuses on the design of the communications, thermal, and power subsystems which would make a mass driver functional. This paper also examines potential trajectories for the mass driver utilizing the restricted circular planar three-body problem approach. The design and trajectory analysis were performed utilizing MATLAB and utilizing techniques outlined in Space Mission Analysis and Design. The result are subsystems which enable a functioning lunar mass driver capable of transferring 100,000 metric tons of lunar material from the Moon to low Earth orbit for a mission life of 20 years.

Acknowledgements

I would like to acknowledge and thank the countless people who helped support me through this project and my entire master's endeavor. First, I want to thank Dr. Periklis Papadopoulos for helping guide me through this process and for giving me the tools necessary to complete this project. Without his assistance, I wouldn't even have been able to attempt this project.

I would also like to thank Dr. Christian Ayers and Professor Jeanine Hunter for their assistance with the trajectory analysis. Without them I would still be staring at MATLAB trying to understand why my code predicting the payload trajectory was sending the payload away from Earth rather than toward it. They were willing to take time out of their very busy lives to enable me to complete this project.

I also want to thank all my friends who were willing to listen to me ramble on about this project, and who were happy to let me bounce ideas off them, even when it was inconvenient. When things got tough, they were always willing to either lend a helping hand or give me the emotional support I needed.

Finally, I want to thank my parents and grandparents for everything they have done and are doing. Without their financial support I would not have even been able to start this project, and when tuition suddenly increased, they had no hesitation about helping cover the cost. They also have always been there for me, ready to listen or help in their own ways, even if the math went over their heads.

This project could not have been completed if it hadn't been for all the wonderful and incredible people who have supported me day in and day out, and I will be eternally grateful for their unconditional love and support.

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 [( )] TJETQq0.00000912 0 612 792 reW* nBT/F2 12 Tf1 0 0 1 231.05 191.54 Tm0 g0 G612

Symbol	Definition	Units (SI)

Symbol	**Definition**	**Units (SI)**
$()_b$	Battery	-----------------
$()_{ea}$	Earth	-----------------
$()_m$	Moon	-----------------
$()_{Payload}$	Payload	-----------------
Acronyms		
NASA	National Aeronautics and Space Administration	-----------------
SCR's	Silicon-Controlled Rectifiers	-----------------
SLS	Space Launch System	-----------------
SSME	Space Shuttle Main Engine	-----------------
LEO	Low Earth Orbit	-----------------
SMAD	Space Mission Analysis and Design	-----------------
BER	Bit Error Rate	-----------------
TCS	Thermal Control Subsystem	-----------------
IR	Infrared	-----------------
EPS	Electrical Power Subsystem	-----------------
DET	Direct-Energy Transfer	-----------------
GEO	Geostationary Earth Orbit	-----------------
ISS	International Space Station	-----------------

1. Introduction

1.1

rating would mean the system temperature could be increased to 440 K, which is the optimal system temperature

1.2.3 Thermal Control Technology

Thermal control is an integral part of a lunar mass driver. The temperature of the electrical components must be kept low enough to prevent hardware failure. When O'Neill's team was designing the lunar mass driver they identified radiators as being the method that the temperature of the SCR's and superconductors would be regulated.

Radiators are one of the few ways to disperse heat in space, but over the years there are several additional methods which have been experimented with and refined. Specifically, heat pipes enable heat to be transferred through conduction, which is a more volume-efficient method of heat transfer than radiation [6]. Heat pipes provide an interesting alternative or complement to radiators which could reduce the overall volume, potentially reducing the launch cost of a lunar mass driver.

1.2.4 Lunar Topography

Over the years, humanity has had many theories regarding the make-up and topography of the Moon. When the first telescopes gave astronomers a much closer glimpse of the Moon, it was believed that the Moon had seas

With the outputs, design the support systems for mass driver and perform trajectory analysis.

2. System Requirements

2.1 Mass Driver Requirements

3. Mass Driver Parameters

3.1 Mass Throughput Rate

The first two design variables for a lunar mass driver, according to O'Neill's team [4], are the mass and launch rate of the payload. Together, these variables have increased power and thermal requirements. This trade-off between mass driver length and power and thermal requirements should be investigated and analyzed in a future project to determine the optimal acceleration for a lunar mass driver.

For this design, an acceleration of 1,000 Earth gravities, or 9,800 m/s^2 shall be used. This is the number used by O'Neill's team in their calculations and provides a solid benchmark for future optimization [4].

3.4 Mass Driver Derived Parameters

Utilizing the parameters determined earlier in this section, the remaining design parameters can be calculated using O'Neill's team's equations. The goal of these equations is to eventually determine the overall system mass, power required, and thermal load, as these values will define the design of the subsystems supporting the mass driver.

A table summarizing the design parameters can be found at the end of this section, while the remaining design parameters will be calculated now. Several of the parameters in the equations below are derived in Appendix A.

3.4.1 Mass Driver Length Derivation

Length of accelerating section:

$$L_a = \frac{v_e^2}{2a} = \frac{(2400\ \text{m/s})^2}{2(9800\ \text{m/s}^2)} = 293\ \text{m} \tag{3.3}$$

Length of decelerating section:

$$L_d = \frac{a}{a_d}\frac{v_e^2}{2a} = 0.69 \cdot 293\ \text{m}$$

Waste power:

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Mass driver efficiency:

– ῦﺧﺢ   ῦب / ῦﺧﺢ   ψϑχ ᶻ ρπᶠ ώ   ρϑσω ᶻ ρπᶠ ώ / ψϑχ ᶻ ρπᶠ ώ   πϑψτ   ψτ Þ (3.15)

Table 3.1- Mass driver dependent system parameters

Dependent Parameter	**Calculated Value**
SCR Mass	$6.60 * 10^{4}$ kg
Winding Mass	$2.66 * 10^{4}$ kg
Feeder Mass	$2.06 * 10^{4}$ kg
Kinetic Power Mass	$1.02 * 10^{5}$ kg
Total Electric Mass	$2.83 * 10^{5}$ kg
Total Mass	$4.25 * 10^{5}$ kg
Total Length	$4.95 * 10^{2}$ m
Waste Power	$1.39 * 10^{6}$ W
Total Power	$8.7 * 10^{6}$ W
Efficiency	84%

4. Communications Subsystem

4.1 Requirements

The requirements for the communications subsystem outlined in chapter 2 are:

- The communications subsystem shall enable at least 25 kbps data transfer from the mass driver to an Earth ground station.
- The communications subsystem shall enable at least 1 kbps data transfer from an Earth ground station to the mass driver.
- The communications subsystem shall enable at least 25 kbps data transfer between the mass driver and a lunar colony.
- The communications subsystem shall have a link margin of at least 10 dB to account for poor weather conditions during data transfer.

The data that will be transmitted from the mass driver to the lunar colony and Earth ground station includes:

- Temperature measurements from superconducting segments.
- Exit velocity at each payload launch.
- Launch trajectory of each payload.
- System warnings or errors.

The data that the mass driver communications subsystem will receive from the Earth ground station and lunar colony includes:

- Launch frequency.
- System commands.

The requirements for the communications subsystem will allow the desired data to be transmitted and received by the lunar mass driver. The necessary power will be determined in the next section by analyzing the link budget for the subsystem, as well as outlining the data flow.

4.2 Historical Communications Architecture

Traditionally, spacecraft communications systems have been comprised of a series of specially manufactured analog circuits designed to survive the harsh environment of space. Due to analog circuits not being particularly adaptable, a series of analog circuits designed for specific tasks was required, resulting in a “communications stack” [14]. This communications stack, while fully space tested and ready, is technologically outdated when compared to modern terrestrial communication standards.

More recently, spacecraft have begun to use field programmable gate arrays (FPGA) as a replacement for the analog communications stack. The advantage of FPGAs is that FPGAs are highly adaptable and flexible and can be programmed to accomplish each of the operations previously accomplished by each component of the communications stack. An FPGA can be

This equation provides a mathematical relationship between the transmitter power, antenna gain and size, data rate, and system losses with the signal-to-noise ratio of the subsystem. The system losses are broken down into several categories and can all be determined by analyzing historical trends, or be directly calculated. The transmitter power, antenna gain and size, and the data rate are all design variables that can either be defined or solved for. For this project, the data rate is specified in the requirements and the transmitter power is the variable that will be solved for, so the antenna size will be defined as the main input parameter.

Since the mass driver will be communicating with both an Earth ground station and a lunar colony, two separate calculations will be required. Only the calculations performed to find the required transmitter power in the Earth ground station case will be shown below, but the calculations for the lunar colony case can be found in Appendix B. Table 4.1 displays the values of the input parameters and system r n[(of thyq0.00000912 0 612 792 reW* nBT/F262T7F2 12 Tf1 0 0 1 72.024

With the desired E_b/N_o calculated, the required transmitter power can be determined utilizing the transmitter gain, space loss, receiver gain, and system noise temperature calculated below.

Transmitter antenna gain:

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Another potential variable to examine is the type of modulation and coding utilized during communication. The calculations earlier in this section assumed either a binary or quadriphased phase shift keying modulation and scheme, but SMAD [15] presents several other potential options. The main effect changing the modulation and coding scheme will have on the calculations is it will change the required E_b/N_o value. Figure 4.3 shows the effects of each modulation and coding scheme on the required transmitter power.

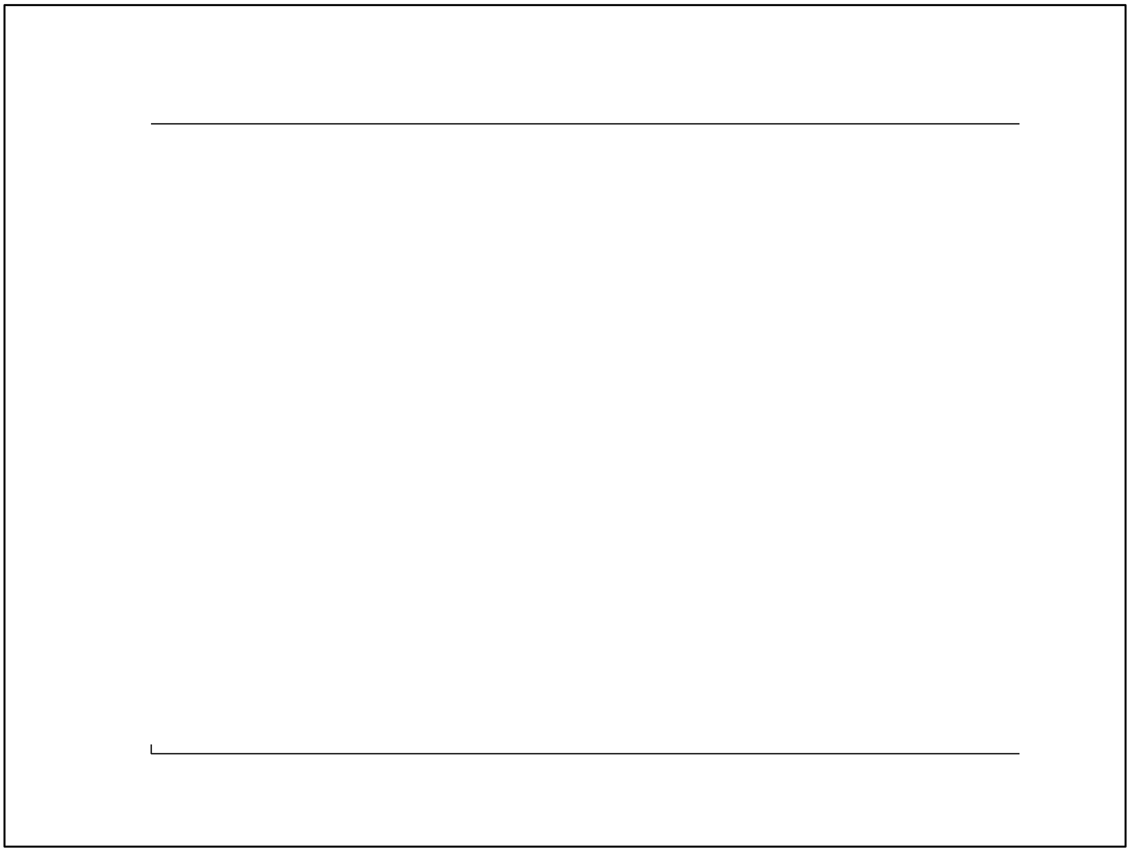

Figure 4.3- Effect of antenna diameter and modulation schemes on required transmitter power

Figure 4.3 shows that the type of modulation and coding scheme appears to have no effect on the overall trend between the antenna diameter and transmitter power. The modulation and coding scheme appears to only shift the magnitude of the trend. The most power intensive modulation and coding scheme is the frequency shift keying which makes sense as SMAD [15] indicates that this scheme has the highest required E_b/N_o. The advantage of this scheme is that it, along with other frequency shift keying schemes, is generally not susceptible to phase disturbances. However, the disadvantage is the higher required transmitter power necessary to perform this scheme.

The least power intensive scheme is the binary phase shift keying scheme with RS Viterbi decoding. This modulation and coding scheme also provides the highest bit error rate performance of all the schemes described in SMAD [15]. The one downside is this scheme is computationally complex. Based on Figure 4.3, if a system requires a small diameter antenna, a scheme that uses Viterbi decoding is most beneficial. However, if size of the antenna is not an issue, a computationally simpler scheme that does not use Viterbi decoding is likely more beneficial.

4.6 Communications Subsystem Summary

For the mass driver communications subsystem, a single antenna will be used alongside a diplexer for both transmitting and receiving data from an Earth ground station and a lunar colony. The antenna diameter will be 1.5 m, which when combined with a binary phase shift keying scheme and RS Viterbi decoding will require 4.27 W during transmission to an Earth ground station and require $8.99e^{-9}$ W during transmission to a lunar colony.

To accommodate for unexpected rain attenuation or other adverse atmospheric conditions, the electrical power subsystem will be designed to provide up to 5 W to the communications subsystem.

5. Thermal Control Subsystem

5.3 Thermal Control Options

A number of thermal control options are available for the mass driver. Ideally, the TCS wi be composed of entirely passive thermal control elements to minimize the power requirement the TCS. The main passive thermal control elements are radiators, multilayer insulation, and surface finishes. One other method of passive thermal control is the use of heat pipes, but due the Moon’s gravity, simple capillary heat pipes would be ineffective. To make heat pipes wor while under the influence of the Moon’s gravity, more complex and expensive Loop heat pipe would likely be required [15].

While passive thermal control elements may be sufficient for keeping the mass driver below t required maximum operating temperature during the hot case, active heating elements may be required to keep the mass driver above the minimum survival temperature during the cold cas If active heating elements are required, the simplest options are patch or cartridge heaters. Between these options, patch heaters are preferrable as they are reusable and more easily controllable via thermostat.

5.4 Radiator Sizing

The governing equations for the design of a passive thermal control system are outlined in SMAD [15] and shown below:

...or area required to keep the ma... maximum operating temp... ...ated. During the radiator sizing pro... ...certainty margin will be ... differences between theory and

With these values, the radiator size for each module for the hot case can be determined using rearranged equations from SMAD [15]:

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For a 2 m long module, 11.4 m^2 is a large, required radiator area. This area could be reduced by up to 9% if the radiators could be placed in a location which received no sunlight, but that still results in a large area of 9.14 m^2. If Loop heat pipes are utilized to evenly distribute the heat across the radiator surface, the required radiator area could be further reduced.

With the required radiator area calculated, the amount of heater power required during the cold case can be determined using equations from SMAD [15]:

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The resulting negative number indicates that heater power is not required to keep the mass driver and electronics above the non-operational survival temperature of 233 K, which also means the thermal system can be entirely passive.

5.5 Thermal Control Analysis

While the calculations in the previous section assume a 5 mil aluminized Teflon radiator surface finish, there are additional surface finishes listed in SMAD [15] which have different emissivity. MATLAB code used to examine the effects of other surface finishes 1 0 0 1 1477a56tions in tS3mi[0

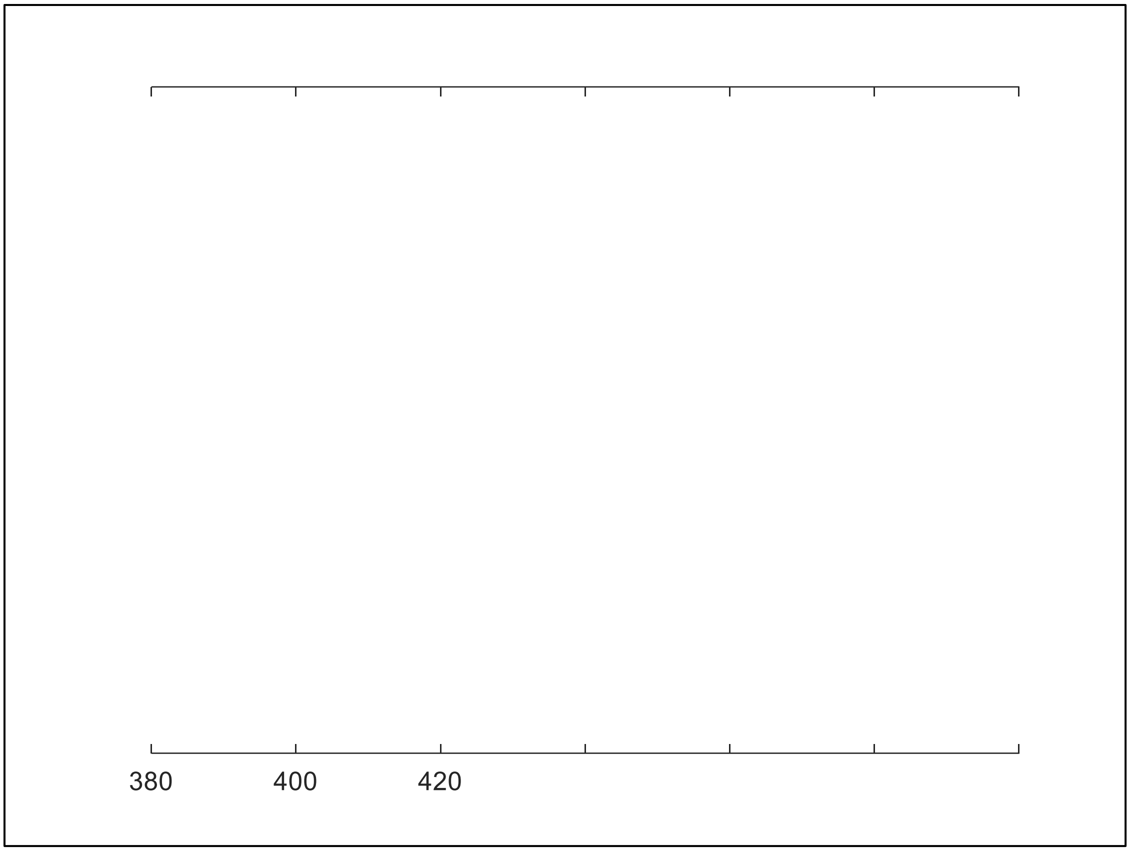

Figure 5.2- Effect of maximum temperature and surface finish on heater power

5.6 Thermal Control Summary

The primary element of the thermal control subsystem will be radiators with a Z93 white paint finish. Loop heat pipes will also be used to optimize the effectiveness of the radiators. While not explicitly necessary, additional electrical resistance patch heaters will be added to the mass driver to raise the system temperature in the event of emergency. The entire TCS will be controlled by a solid-state temperature controller that is linked to mechanical temperature sensors in each mass driver module. Each 2 m long module will have approximately 6.2 m^2 of radiators to dissipate the waste heat generated by the mass driver superconducting coils.

While the calculations were performed assuming a 5°sunlight incidence angle on the radiators, it is recommended that the radiators should be placed such that they are almost never in sunlight as this can further reduce the required radiator area by upwards of 9%. Additionally, in the future, further TCS analysis and optimization specifically regarding the placement and utilization of Loop heat pipes should be able to further reduce the required radiator area. However, until that advanced analysis is performed, it is impossible to know how effective the Loop heat pipes will be.

The primary

6. Electrical Power Subsystem

6.1 Requirements

The requirements for the electrical power subsystem (EPS) outlined in chapter 2 are:

- The power subsystem shall provide at least 8.7 MW to the mass driver while the mass driver is operational.
- The power subsystem shall provide enough power to operate the active component of the thermal control subsystem.
- The power subsystem shall provide enough power to operate the communications subsystem.

Based on the calculations performed in chapters 4 and 5, the required power for both the TCS and communications subsystems is knowactive component of the S1 197.0000091 2 0ti

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To successfully power the mass driver, based on the solar array area calculated above, a 260 m square solar array will be required. The size of the solar array may seem staggering compared to the size of solar arrays powering traditional satellites. However, research performed in solar collection satellites which propose solar array sizes of nearly 5 km^2 [16] indicate that the size of solar array required to power the mass driver is actually feasible. Additionally, it is possible that solar cells made of a material other than indium-phosphide may result in a smaller and more efficient solar array. This possibility will be examined in a later section.

6.4 Energy Storage

While the solar array required is large, since the mass driver will not be operating during eclipse, the energy storage system will be much smaller. Due to the long mission life, two batteries will be used, but the batteries will be redundant and not used in series or parallel, so each battery must individually provide enough power during eclipse.

The required battery capacity is calculated using equations from SMAD [15]:

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Table 6.3 - Efficiency and rate of degradation for various solar panel types

Solar Panel Type	Efficiency	Rate of Degradation (per year)

Figure 6.2 - Effect of solar panel efficiency on solar array area at incidence angle of 15°

Based on Figure 6.2, even bringing the rate of degradation for GaInP/GaInAs multijunction panels from the current rate of 1.35% per year to the GaAs 0.49% per year would result in a substantial reduction in solar array area. Specifically, it results in an area reduction of approximately 15% which is significant.

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A solar array of approximately 37,000 m^2 composed of multijunction GaInP/GaInAs panels with a solar incidence angle of 15° or less will be used to power the mass driver. An energy storage system holding 2,240 W*hr in two redundant lithium-ion batteries with a mass of approximately 16 kg will be used to power the communications subsystem and an emergency heating system during eclipse. The EPS is controlled using a direct-energy-transfer control subsystem and an unregulated bus subsystem will be utilized to increase battery life and overall power efficiency.

In the future, it is suggested that a closer look be taken at a nuclear reactor as a potentially viable energy source for the mass driver. A nuclear reactor would ideally not take up as much space as the required solar array and would not be contingent on sunlight, increasing the amount of operational time for the mass driver. At this time, there is not enough research into the implementation of a nuclear reactor on the lunar surface for the idea to be fully analyzed or considered.

7. Orbital Trajectory Analysis

7.1 Initialization and Assumptions

In addition to the design of the mass driver subsystems, the orbital trajectory of the payload must be analyzed. One of the requirements for the mass driver is that it delivers the payload to LEO, so the mass driver must be designed to accomplish this task. Since the magnitude of the exit velocity of the payload was determined in Chapter 3, the magnitude of the exit velocity is not a variable that can be changed to manipulate the orbital trajectory of the payload. What can be manipulated, during the design process, is the exit velocity vector.

The mass driver will be built at the lunar northern pole for the reasons outlined in Chapter 1. Since the Moon orbits the Earth at approximately the same speed the Moon rotates about its axis, a mass driver built at the lunar northern pole will always point toward Earth, with minimal variance in angle. This means that the mass driver can be built at whatever angle results in a payload trajectory that reaches LEO in the shortest possible amount of travel time.

To determine the optimal payload trajectory, a circular planar restricted three-body problem approach will be utilized. A three-body problem approach will be utilized as both the Moon and the Earth have noticeable gravitational effects on the payload during the payload orbit, and neglecting either celestial body would result in erroneous results. The mass of the payload is negligible when compared to the mass of the Moon and of the Earth, which means the three-body problem is restricted. The circular and planar assumptions simplify the problem further by ignoring

The.2 reW* nBT/F2 14.04 Tf1 0 0 1 107.42 675.1 Tm0 g0 G[(In)-4(iIn)-4(iIn)-4(42 792 re84/F211ke)5(.2 riIn5 0 1 514/F23ke)

Figure 7.1- Planar Barycenter Frame for Restricted Three-Body Problem

Next, the angular velocity of the Barycenter frame with respect to the Newtonian reference frame is defined:

$$^{S}\ \ {}^{B} = \omega\ \mathbf{b}_{z} \tag{7.1}$$

The magnitude of the angular velocity is constant and is directly related to the orbital period, τ, of the Earth-Moon system around the Barycenter.

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[illegible] (7.14)

Substituting the resulting vectors into equation 7.12 and separating the equation into scalar components results in the following equations of motion:

($\mathbf{b}_x$)

[illegible] (7.15)

($\mathbf{b}_y$) [illegible] (7.16)

($\mathbf{b}_z$) [illegible] (7.17)

Equations 7.15 and 7.16 are coupled, indicating that a numerical approach will be required to solve the equations of motion in the x and y directions. The z direction, however, is independent of both the x and y directions, meaning that if the payload begins in planar motion, as assumed, the payload will remain in planar motion. This indicates that equation 7.17 should not be needed for this orbital analysis.

7.3 Methodology

Since equations 7.15 and 7.16 are coupled, MATLAB will be utilized to provide a numerical solution. Specifically, the function ode45 will be used. The initial conditions used to solve the equations of motion will be the x and y positions of the payload at the instant it is launched from the mass driver, and the magnitude of the mass driver exit velocity in the x and y directions.

The magnitude of the mass driver exit velocity was determined in Chapter 3 and will not be changed. Rather, the angle of the mass driver will be changed, ranging from -60° to 60° from the Barycentric x-axis and rotating about the Barycentric z-axis, to provide a series of potential orbital trajectories. From the resulting trajectories, the trajectory that reaches LEO in the shortest time, and that does not result in an impact with the Earth, will be selected. The MATLAB code utilized can be found in Appendix B.

7.4 Results and Analysis

7.4.1 MATLAB Data

Across the -60°

Table 7.1- Time required to reach GEO, LEO, and Earth for a given launch angle

Angle (°)	**Time to GEO (days)**	[redacted]	
20	1.7475	2.1786	[redacted]

[redacted]

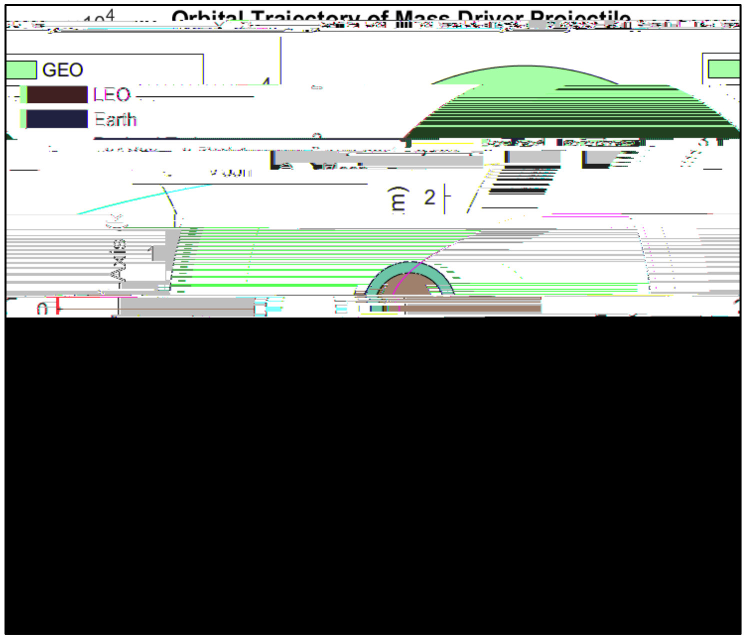

Figure 7.2- Close-up of orbital trajectory of payload launched at 21°

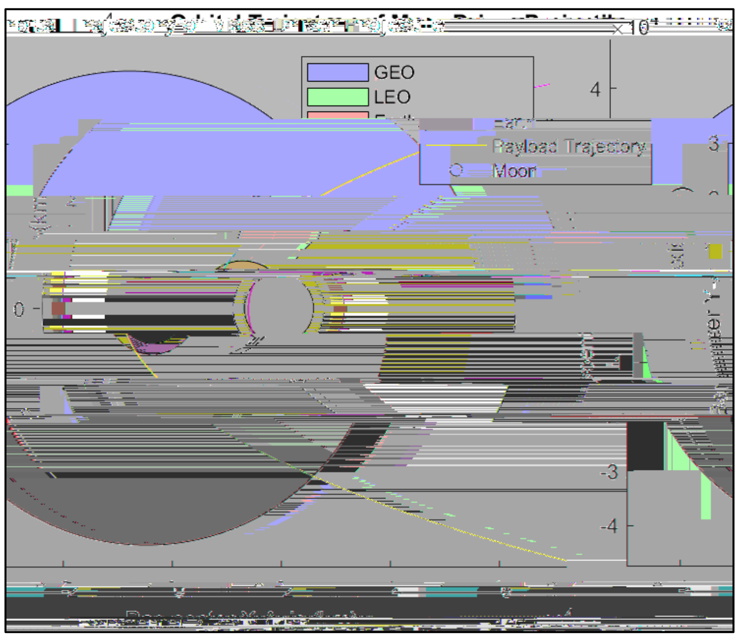

Figure 7.3- Close-up of orbital trajectory of payload launched at 22°

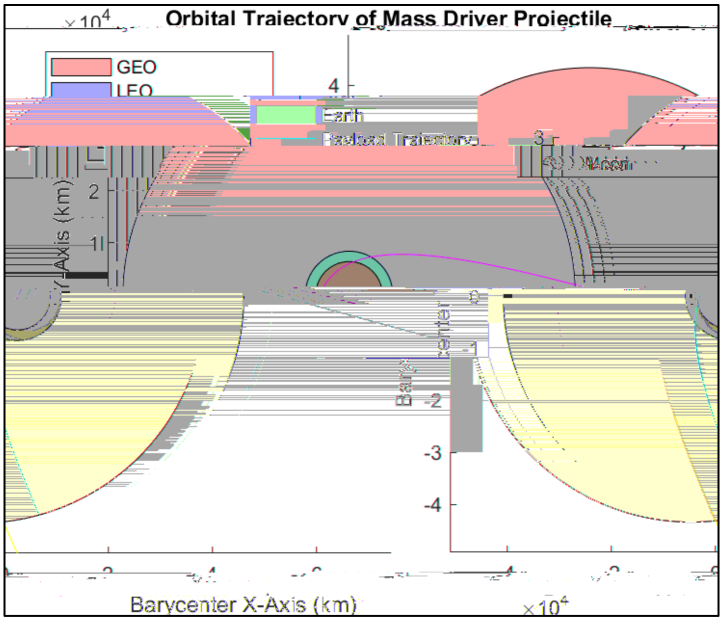

Figure 7.4- Close-up of orbital trajectory of payload launched at 30°

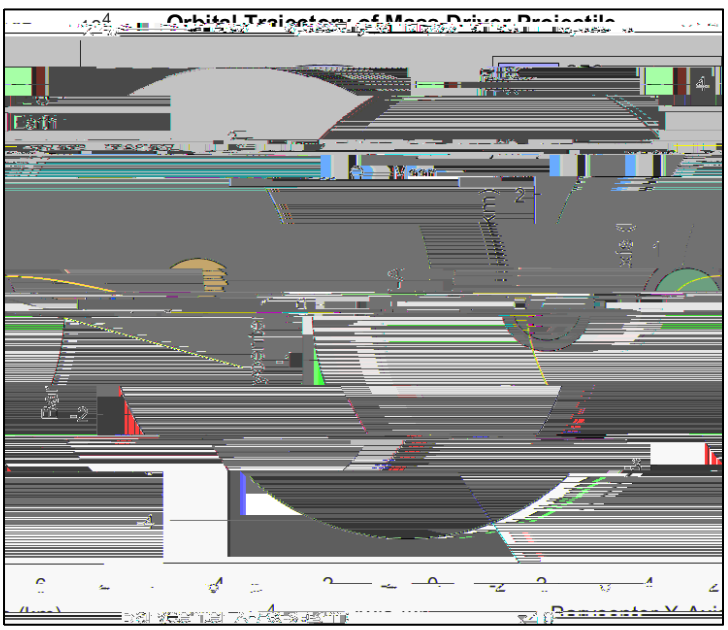

Figure 7.5- Close-up of orbital trajectory of payload launched at 29°

7.4.3 Complete Solution

The single complete solution for a mass driver launch angle that results in a trajectory passing through LEO while not impacting the Earth is an angle of 31°. Figure 7.6 displays the resulting payload trajectory as it leaves the lunar surface and eventually passes through LEO. As can be seen, if the payload is not intercepted while in LEO, the payload will eventually begin a return trajectory back toward the Moon. Figure 7.7 shows a close-up of the payload passing through LEO.

Figure 7.6- Orbital trajectory of payload at 31°

Figure 7.7- Close

7.5 Summary

Utilizing a circular planar restricted three-body approach, an orbital trajectory was identified which would result in a payload being launched from the lunar mass driver and eventually reaching LEO in approximately 2.14 days. The specific mass drive launch angle required to achieve this trajectory is 31°, so the lunar mass driver will be constructed on a line that is 31° from the Barycentric x-axis. An orbital platform or station in LEO will be required to collect the transported payload, but the design of such a station is beyond the scope of this project.

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8. External Considerations

8.1 Economic Considerations

The building of a lunar mass driver has some interesting economic ramifications. As mentioned in Section 1.1, the main resources transported by the mass driver are helium-3, ice, methane, and platinum. Of those, helium-3 and platinum are the two that would most likely be used on Earth.

Currently, helium-3 is primarily used for medical imaging and for nuclear fusion experiments. Due to the Earth's atmosphere protecting the Earth from solar wind, almost all helium-3 on Earth has been artificially created using an expensive process involving the radioactive decay of tritium, a hydrogen isotope primarily used in nuclear warheads. Introducing naturally occurring helium-3 from the Moon would likely dramatically reduce the price of helium-3. This would in turn likely reduce the cost of both medical imaging and nuclear fusion experiments.

Platinum is one of the rarer metals on Earth, consequently making it quite valuable. Unlike gold or silver, platinum has a limited number of potentially commercial or electronic applications. This has resulted in platinum being a symbol of wealth and status. The introduction of a new source of platinum might reduce the overall price of platinum, but since platinum has so few commercial applications, it is unlikely to have a major economic impact.

8.2 Environmental Considerations

A lunar mass driver itself has no major environmental considerations. However, a mass driver is predicated on the existence of a lunar mining operation that requires the existence of a reliable and cheap transportation system. A lunar mining operation does present some potential environmental considerations.

As discussed previously, the main materials transported back to Earth would be platinum and helium-3. However, both of those materials are relatively scarce on the Moon and would require something akin to strip mining to result in meaningful quantities of each material [1]. A mining operation on this scale would certainly do irrevocable harm to the lunar surface and environment. While there is no known ecosystem on the Moon to destroy, the geological state of the Moon would be forever altered, and the potential effects of strip mining the Moon are completely unknown.

8.3 Political Considerations

Problems with the construction of a lunar mass driver largely begin with political considerations. The Moon does not belong to any one country, and any attempt to build a structure on the Moon or to gather resources from the Moon's surface could result in major political backlash from foreign countries. Additionally, the building of a large mass driver could be seen as placing a potential weapon on the Moon, which would likely violate the 1967 Outer Space Treaty banning the militarization of celestial bodies [21].

Any successful effort to colonize, mine and transport resources from the Moon would require international cooperation through an organization such as the United Nations. Additional safeguards would have to be put in place to ensure the mass driver could not be easily weaponized by any one country or member state. It is also unlikely that the United Nations would allow private companies to mine the Moon, meaning that it would be up to international governments to build the lunar infrastructure necessary to operate a mass driver.

8.4 Ethical Considerations

The ethical considerations regarding the construction and operation of a lunar mass driver are largely related to considerations already outlined. There is a distinct ethical argument against mining the Moon and potentially destroying geological data pertaining to the early life of the Earth. Additionally, there is concern that a lunar mass driver could be weaponized, adding another potential weapon of mass destruction to the already sizeable international stockpile.

This particular design could not easily be weaponized as the acceleration and trajectory of the payload is determined by the size and angle of the mass driver. However, it is not impossible to transform the mass driver into a weapon capable of taking down satellites. Theoretically, by either reducing the charge within the capacitors or by changing the mass, the trajectory could be altered enough to turn the mass driver in a pseudo shotgun capable of damaging satellites in LEO and GEO. Safeguards during the actual implementation of the mass driver must be put in place to prevent the mass driver’s weaponization.

8.5 Health Considerations

The mass driver is designed to be fully automated, making human technicians unnecessary. However, it is likely humans will be necessary to build this colossal structure. There are absolutely health risks to working in space, and the long-

The best method currently available would involve designing the mass driver into a series of identical modules which could be connected via a locking mechanism similar to the ones used during docking with the International Space Station (ISS). Breaking down the mass driver into a series of identical modules would also make manufacturing easier and faster and make testing each module easier. However, it also presents an issue that if there is a flaw in the module, there would be a flaw in all the modules.

9. Final Mass Driver System Design

9.1 Mass Driver Design Results

The mass driver shall transport at least 100,000 metric tons of lunar material each year back to LEO. This design accomplishes this goal by launching a 25.4 kg payload every 10 seconds. The mass driver consists of an accelerating section that is approximately 293 m in length which accelerates a payload bucket to a speed of 2.4 km/s, which is greater than the necessary 2.38 km/s required to escape the Moon's gravitational pull.

The decelerating section is approximately 202 m in length and will return the empty bucket back to the beginning of the accelerating section after slowing it down. The entire mass driver has an approximate mass of 424.8 metric tons and will be broken down into nearly identical modules which can be connected and assembled at the lunar north pole at an angle of 31° which will result in the proper trajectory. The mass driver will require 8.7 MW of power which will be generated using a field of solar panels. The overall system has a power efficiency of 84%.

9.2 Communications Subsystem Design Results

The communications subsystem for the mass driver provides a 25 kbps uplink and downlink between the mass driver and either an Earth ground station or a lunar colony ground station. This is accomplished using a dual-purpose 1.5 m diameter antenna with a diplexer enabling both uplink and downlink. The transmitter antenna gain is 37.2 dB while the receiver antenna gain is 37.7 dB. A binary phase shift keying system with RS Viterbi decoding will be used to minimize the BER and ensure effective and accurate data transfer.

9.3 Thermal Control Subsystem Design Results

The thermal control subsystem for the mass driver keeps the system below 400 K during operation and above 233 K during maintenance. This is accomplished by having a radiator area of approximately 6.2 m^2 for each 2 m long module, with the radiators using a Z93 white paint finish. Due to the mass driver being at the pole with almost constant sunlight, no active thermal system is necessary, however patch heaters will be added to each module in the event of a temperature emergency. The system will be monitored by a solid-state temperature controller linked to temperature sensors within each module. In the future, heat pipes can be added to the radiators to increase the efficiency of the radiators.

An energy storage system consisting of stable lithium-ion batteries will be used to store power during eclipse and contains 2,240 W*hr. Specifically, it is a system with two batteries that are redundant where each battery has an energy capacity of 2,240 W*hr, which is enough to power

In the future, further work can be done narrowing down specific components and optimizing the internal workings of the mass driver, but that would require substantial physical construction and testing and is beyond the scope of the original definition of this project. This project focuses on the design of the subsystems supporting the mass driver and the determination of the orbital trajectory required to transfer the payload from the lunar surface to LEO, and the design outlined in this report accomplishes that task, meeting all of the requirements presented in the requirements section.

[19] Dalton, P. J., Schwanbeck, E., North, T., and Balcer, S., “International Space Station Lithium-Ion Battery,” NASA Aerospace Battery Workshop, November 2016.

[20] Hunter, J. M., “Chapter 10: The Three-Body Problem,” *AE 242 Orbital Mechanics and Mission Design: Astrodynamics Course Reader*, SJSU, San Jose, CA, 2022, pp. 126-136.

[21] “Treaties on Principles Governing the Activities of States in the Exploration and Use of Outer Space, including the Moon and other Celestial Bodies,” United Nation Treaty 2222 (XXI), Dec. 1966.

Appendix A: Mass Driver Component Sizing Derivations

Bucket coil width:

ὡ͇   π϶ρ ᶻ Ὀ   π϶ρ π϶φσχ ά   π϶πφσχ ά (A.1)

Effective radius of the bucket coil:

ὶ π϶ςφ ᶻ Ὀ   π϶ςφ π϶φσχ ά   π϶ρφφ ά (A.2)

Length of single-turn drive winding:

℔̧   "Ὀ " π϶φσχ ά   ς϶ππ ά (A.3)

IndG -0.040/F2 12 Tf1 0 0 1 72.024 666.1 Tm0 g0 G[(B)-2(uc)4(ke)4(t coil widt)-3(h:)] TJETQ60 g04 Tle2 546

Empty bucket mass:

ά ̥ ς ᶻ ά خةحخخ ς ᶻ ςψ8ψ ῼΏ υφ8φ ῼΏ (A.9)

Loaded bucket mass:

ά ̠ۂ ά ̠ ά ̥ υφ8φ ῼΏ ςυ8τ ῼΏ ψς ῼΏ (A.10)

Ratio of unloaded to loaded bucket mass:

ά ̥ j ά ̠ۇ υφ8φ ῼΏ / ψς ῼΏ π8φω

Appendix B: MATLAB Code

```
%% Master's Project
%Ethan Miller
clc
clear
close all

%% Communications Analysis
%Constants
Link_m = 10; %Link Margin [dB]
Loss_i = -2; %Implementation Loss [dB]
BER = 10^-5; %Bit Error Rate
%Data_R = 25; %Data Rate [kbps]
Er = 0.2; %Receive Antenna Pointing Error [degrees]
Dr = 0.5:0.1:5; %Receive Antenna Diameter [m]
Se = 3.85*10^8; %Propogation Path Length to Earth [m]
Sm = 10000; %Propogation path Length to Lunar Base [m]
Et = 0.5; %Transmit Antenna Pointing Offset [degrees]
Dt = 0.5:0.1:5; %Transmit Antenna Diameter [m]
Ll = -1; %Transmitter Line Loss [dB]
f = 2; %Frequency [GHz]
L_ab = -3.4; %Absorption Loss [dB]
Ts = 135; %System Noise Temperature [K]
%EbNo_Req = 9.6;
EbNo_Req = [9.6 10.3 13.3 9.2 4.4 2.7 4.0]'; %EbNo Required for
QPSK and BER = 10^-5 [dB]
Eta = 0.55; %Antenna Smoothness Factor (1 is perfect)

%Calculating Data Rate
BitS = 5; %Number of bits per sample for 1.5% Accuracy
Sections = 248; %Total Number of 2m Sections/Modules
Data_time = 5; %Time between Data Bursts [s]
Samples = 1*Sections; %Number of Samples per data burst
Data_R = Samples/Data_time*BitS; %Data Rate [bps]

%Calculations
EbNo = EbNo_Req-Loss_i+Link_m; %Desired EbNo [dB]

%Calculating Carrier to Noise Density Ratio
CNo = EbNo + 10*log10(Data_R);

%Calculating Receive Antenna Gain
thetaR = 21./(f.*Dr); %Receiving Half-power Beamwidth [degrees]
Lpr = -12*(Er./thetaR).^2; %Receive Antenna Pointing Loss [dB]
Grp = 20*log10(pi)+20*log10(Dr)+20*log10(f*10^9)+10*log10(Eta)-
20*log10(3*10^8); %Peak Receive Antenna Gain [dB]
Gr = Grp+Lpr; %Receive Antenna Gain [dB]
```

```
%Calculating Space Loss
Lse = 20*log10(3*10^8)-20*log10(4*pi)-20*log10(Se)-
20*log10(f*10^9); %Space Loss to Earth [dB]
Lsm = 20*log10(3*10^8)-20*log10(4*pi)-20*log10(Sm)-
20*log10(f*10^9); %Space Loss to Moon [dB]

%Calculating Transmit Antenna Gain
thetaT = 21./(f*Dt); %Transmitting Half-Power Beamwidth
[degrees]
Lpt = -12*(Et./thetaT).^2; %Transmit Antenna Pointing Loss [dB]
Gtp = 44.3-10*log10(thetaT.^2); %Peak Transmit Antenna Gain [dB]
Gt = Gtp+Lpt; %Transmit Antenna Gain [dB]

%Calculating EIRP
EIRPe = EbNo-Lse-L_ab-Gr-
228.6+10*log10(Ts)+10*log10(Data_R*100); %EIRP to Earth [dB]
EIRPm = EbNo-Lsm-L_ab-Gr-
228.6+10*log10(Ts)+10*log10(Data_R*100); %EIRP to Lunar Colony
[dB]
Pte1 = EIRPe-Ll-Gt; %Transmitter Power to Earth [dBW]
Ptm1 = EIRPm-Ll-Gt; %Transmitter Power to Lunar Colony [dBW]


%Calculating Transmitter Power
Pte2 = EbNo-Ll-Gt-Lse-L_ab-Gr-
228.6+10*log10(Ts)+10*log10(Data_R*100); %Transmitter Power to
Earth [dBW]
Ptm2 = EbNo-Ll-Gt-Lsm-L_ab-Gr-
228.6+10*log10(Ts)+10*log10(Data_R*100); %Transmitter Power to
Moon [dBW]
Pte = 10.^(Pte1/10); %Transmitter Power [W]
Ptm = 10.^(Ptm1/10); %Transmitter Power [W]

figure(1)
plot(Dt,Pte1(1,:),Dt,Ptm1(1,:))
xlabel('Transmitter Antenna Diameter (m)')
ylabel('Transmitter Required Power (dBW)')
title({'Effect of Transmitter Antenna Diameter on Required
Transmitter Power'})
legend('Mass Driver to Earth','Mass Driver to Lunar
Colony','location','NE')

figure(2)
plot(Dt,Pte1)
xlabel('Transmitter Antenna Diameter (m)')
ylabel('Transmitter Required Power (dBW)')
```

```
title({'Effect of Transmitter Antenna Diameter on Required
Transmitter Power for','Mass Driver to Earth Ground Station
Transmission for','Various Modulation and Coding Schemes'})
legend('BPSK/QPSK','DPSK','FSK','8FSK','BPSK + R-1/2
Viterbi','BPSK + RS Viterbi','8FSK + R-1/2
Viterbi','location','NE')

%% Thermal Control System Analysis
%Constants
qsolarH = 1420*sind(0); %Solar Constant Hot Case [W/m^2]
qsolarC = 1360; %Solar Constant Cold Case [W/m^2]
MIR = 430; %Moon IR [W/m^2]
alphaH = 0.2; %Absorptance Hot Case
alphaC = 0.1; %Absorptance Cold Case
epsilonH = [0.72 0.78 0.85 0.88 0.92]'; %IR Emittance Hot Case
epsilonC = 0.78; %IR Emittance Cold Case
QintH = 5620; %Power Dissipation Hot Case [W]
%QintC = 0; %Power Dissipation Cold Case [W]
Boltz = 5.67*10^-8; %Boltzmann Constant [W/m^2K^4]
TradH = 380:2.5:500; %Max temperature [K]
TradC = 233; %Min Temperature [K]

%Radiator Sizing Hot Case
qExt = qsolarH+MIR;
A = QintH./(epsilonH.*Boltz.*TradH.^4-qExt);
QintC = Boltz*epsilonC*A*TradC^4-qExt*A;
Qrad = Boltz.*epsilonH.*A.*TradH.^4;
Qext = qExt*A;

figure(3)
plot(TradH,A)
xlabel('Maximum Temperature (K)')
ylabel('Radiator Area (m^2)')
title('Effect of Maximum Temperature and Reflective Coating on
Radiator Area')
legend('5 mil Aluminized Teflon (e = 0.72)','5 mil Silvered
Teflon (e = 0.78)','S13G-LO Paint (e = 0.85)','Chemglaze A276 (e
= 0.88)','Z93 Paint (e = 0.92)','location','NE')

figure(4)
plot(TradH,QintC)
xlabel('Maximum Temperature (K)')
ylabel('Heater Power during Cold Case (W)')
title({'Effect of Maximum Temperature and Reflective Coating
```

```
legend('5 mil Aluminized Teflon (e = 0.72)','5 mil Silvered
Teflon (e = 0.78)','S13G-LO Paint (e = 0.85)','Chemglaze A276 (e
= 0.88)','Z93 Paint (e = 0.92)','location','SE')
```

```
plot(CellT,A_saT)
title({'Effect of Solar Panel Efficiency on Solar Array Area at
Incidence','Angle = 15 degrees'})
xlabel('Solar Panel Efficiency')
ylabel('Solar Array Area [m^2]')
legend('Degradation = 1.61%/year','Degradation =
1.35%/year','Degradation = 1.01%/year','Degradation =
0.74%/year','Degradation = 0.54%/year','Degradation =
0.41%/year','location','NE')

%Energy Storage Calculations
Cycle = 20*365+5; %Number of discharge cycles
DOD = 0.25; %Depth of Discharge
Nbatt = 1; %Number of batteries making up Storage System
ncharge = 0.9; %Charging efficiency

Cb = Pe*Te/DOD/Nbatt/ncharge/3600; %Required Battery Capacity

%% Orbital Analysis
%Restricted Planar Circular Three Body Problem [Earth-Moon-Rock]
%Constants
Me = 5.9722e24; %Mass of Earth [kg]
Mm = 7.3477e22; %Mass of Moon [kg]
G = 6.6743e-20; %Gravitional Constant [N*km^2/kg^2]
Re = 6378; %Mean Radius of Earth [km]
Rm = 1737.4; %Mean Radius of Moon [km]
Rem = 384400; %Mean Distance between Earth and Moon [km]
Tau = 2*pi*Rem^1.5/sqrt(G*(Me+Mm)); %Orbital Period [s]
tf = 300000; %Final time [s]
V0 = -2.4; %Mass Driver Exit Velocity [km/s]
Theta = 22; %Angle of Mass Driver relative to Barycenter X-Axis
[degrees]

%Defining the Barycenter
Xe = -Rem*Mm/(Mm+Me); %Distance from Earth to Barycenter [km]
Xm = Rem*Me/(Mm+Me); %Distance from Moon to Barycenter [km]
W = 2*pi/Tau; %Angular Velocity of Barycenter in Newtonian Sun
Frame [rad/s]

%Defining Initial Conditions
x0 = Xm; %Initial Distance from Barycenter in X-Direction [km]
xdot0 = V0*cosd(Theta); %Initial Velocity in X-Direction [km/s]
y0 = 0; %Initial Distance from Barycenter in Y-Direction [km]
ydot0 = V0*sind(Theta); %Initial Velocity in Y-Direction [km/s]

%Equations of Motion for Rock using Barycenter Coordinates
```

```
%xddot-2*W*ydot+(G*(Me/Re^3+Mm/Rm^3)-W^2)*x =
G*Mm*Me*Rem/(Rm^3*(Me+Mm))-G*Me*Mm*Rem/(Re^3*(Me+Mm));
%Barycenter X Direction
%yddot+2*W*xdot+(G*Me/Re^3+G*Mm/Rm^3-W^2)*y = 0; %Barycenter Y
Direction
%zddot+(G*Me/Re^3+G*Mm/Rm^3)*z = 0; %Barycentric Z Direction

%Setting up ode45
tspan = [0 tf]; %Time span used by ode45
IC = [x0 xdot0 y0 ydot0]; %Initial Conditions [km km/s]
[t, x] = ode45(@fun, tspan, IC);
X = x(:,1);
Xdot = x(:,2);
Y = x(:,3);
Ydot = x(:,4);

%Generating a Circle for Earth
Earth = nsidedpoly(1000, 'Center', [Xe 0], 'Radius', Re);

%Generating a Circle for LEO
LEO = nsidedpoly(1000, 'Center', [Xe 0], 'Radius', Re+2000);

%Generating a Circle for GEO
GEO = nsidedpoly(1000, 'Center', [Xe 0], 'Radius', Re+37000);

%Plotting the position of Mass Driver Rock
hold on
figure(7)
plot(GEO, 'FaceColor', 'b')
plot(LEO, 'FaceColor', 'g')
plot(Earth, 'FaceColor', 'r')
plot(X,Y,'-m',Xm,0,'ok')
title('Orbital Trajectory of Mass Driver Projectile')
xlabel('Barycenter X-Axis (km)')
ylabel('Barycenter Y-Axis (km)')
legend('GEO','LEO','Earth','Payload Trajectory','Moon')
axis equal

%Determine time required to reach GEO
tGEO = find(X < Re+37000+Xe & Y < Re+37000 & X > -(Re+37000)+Xe
& Y > -(Re+37000));
timeGEO = tGEO(1)/length(t)*tf/3600/24

%Determine time required to reach LEO
tLEO = find(X < Re+2000+Xe & Y < Re+2000 & X > -(Re+2000)+Xe & Y
> -(Re+2000)); %Dummy Variable finding LEO time
```