Koç University

MECH 427: Rocket Propulsion - Project

Lunar Descent Vehicle Design

Hakancan Öztürk

20 January 2023

Instructor: Arif Karabeyoğlu

Table of Contents

Introduction

In this project, a lunar descent vehicle will be designed to transport a payload from the lunar orbit to the surface of the Moon. The design of a lunar descent vehicle is a complex and challenging task, but for sake of simplicity, the scope of the project is trimmed. This vehicle will be responsible for transporting a payload but should be designed suitable for a man-rating mission, and the fuel must be able to withstand the harsh conditions of the lunar surface.

There are some constraints regarding the trajectory, fuel and oxidizer selection, and fuel handling, therefore the design was done in accordance with the requirements and constraints. In this report, the main emphasis will be on the trajectory, propellant selection and optimization, design of the vehicle, survivability, and operational issues.

Mission Objective and Requirements & Constraints

Mission Objective:

Design a vehicle that will take the payload from the lunar orbit to the lunar surface. The following are the priorities for the design:

• Safety is essential. An eventual man rating might be required.

• Minimize the hardware mass that needs to be brought from Earth.

Requirements:

• Land at a site at 0 degrees latitude (mean surface level)

• Lunar Orbit: 500 km, circular, 0 degrees inclination

• Payload that will be landed: 200 kg

• Landing velocity needs to be less than 1 m/sec

• Must have 10% reserve fuel

• Maximum acceleration: 3 g's

Target(Moon) Properties:

• $M_{moon} = 7.347 \cdot 10^{22}\ kg$

- $R_{moon} = 1737.4\ km$
- $\mu_{moon} = 4.90 \cdot 10^{12} \frac{m^3}{s^2}$
- $g_{moon}^{surface} = 1.62 \frac{m}{s^2}$
- $-120°C < T_{moon}^{surface} < 170°C$
- Atmosphere density is negligible, yielding no lift or drag forces
- Rotation of the moon was disregarded for simplicity

Rocket and Propellant Selection & Optimization

The propellant selection plays a very important role in this project since all of the requirements and calculations strictly depend on the chemical, physical, and performance traits of the propellant. For the given mission, literature research was done on previous lunar modules that succeeded in the past. For many of the Apollo missions the lunar module, namely the Descent Propulsion System uses a liquid hypergolic propellant engine SE-10, which will be an inspiration for the fuel and component selections done in this project.

In this project, I've also used the same flight- and mission-proven hypergolic propellant which is Aerozine-50 and nitrogen tetroxide(N_2O_4/NTO). This fuel-oxidizer combination with the suitable component performs very well in the given conditions and is throttleable which is very advantageous in a descent and landing scenario. Another advantage of this selection is that due to the Hypergolic nature of the propellant, the rocket complexity is reduced because the ignition control is easier and does not require any extra parts, and they can be restarted and throttled with ease. One disadvantage of this engine selection is that the engine runs on a pressure-fed cycle which would require a Helium tank causing the structural mass increase.

The mission will be performed using a single stage with the specified propellant meaning that the rocket will be a hypergolic liquid rocket. The number of burns and the details of the motion will be elaborated on in the trajectory part.

Fuel: Aerozine 50

Aerozine 50 is a type of hydrazine-based fuel that was developed for usage in the upper stages of rockets. It is a 50:50 mixture of hydrazine and unsymmetrical dimethylhydrazine (UDMH), which are both highly toxic and hypergolic (self-igniting) chemicals. Despite their toxic and self-igniting properties, they are flight-proven and can be utilized when treated with the required care. Aerozine is a storable, high-performance fuel that can be stored on the lunar surface due to its low melting point.

Storability, Toxicity, and Explosiveness

In terms of storability, Aerozine 50 must be kept in specially designed-containers to prevent leaks or spills. It should be stored in a cool, dry place away from heat sources, sparks, or open flames. It is important to follow proper handling procedures when working with Aerozine 50 to avoid accidents. Therefore, the lunar environment is favorable in terms of storability due to low temperature and the absence of an atmosphere.

The ingredients of Aerozine 50, hydrazine, and UDMH are toxic and can cause harm to humans if ingested, inhaled, or absorbed through the skin. They can cause irritation to the eyes, skin, and respiratory system, and are potential carcinogens. They can also cause damage to the liver, kidneys, and central nervous system. Meaning that a potential spillage would pose a lethal risk on a human-rating mission.

Aerozine 50 is also explosive when ignited which is the main reason behind the performance of the substance. It can ignite easily and burn vigorously, releasing toxic gasses and vapors in the process. There have been some explosion incidents in the past so it is important to handle Aerozine 50 with caution and to follow proper safety procedures to avoid accidents.

Oxidizer: NTO

Nitrogen tetroxide (NTO) is a highly reactive chemical compound that is used as an oxidizer in rocket propulsion systems. It is a powerful oxidizing agent, which means that it can supply oxygen to support the combustion of fuels. NTO is a storable liquid that is usually stored in pressurized containers and handled using specialized equipment to avoid accidents or spills. Again, it must be handled with caution but is flight- and mission-proven, making it a great candidate for our oxidizer.

The compatibility of the fuel and the oxidizer with the tank materials will be elaborated in the design part, which also directly influences the storability of our propellant.

Moreover, as mentioned above, the rocket I designed was inspired by the SE-10 engine, which has a 110 psi (760 kPa) chamber pressure at full throttle. We will use this number for the CEA calculations and develop our own performance metrics, by optimizing for the O/F ratio and expansion rate.

Storability, Toxicity, and Explosiveness

Liquid nitrogen tetroxide (NTO) is a highly reactive chemical that is a very strong oxidizer. It is not stable and can decompose violently, especially when heated or contaminated.

NTO is also classified as toxic and can cause respiratory irritation or injury if inhaled. For these reasons, it must be handled with caution and stored in specially designed containers to prevent accidents. Therefore, in case of a spillage on a human-rating mission, it may pose a lethal or non-lethal risk to the crew.

NTO is generally not considered to be explosive on its own, but it can contribute to the explosion of other materials if it is involved in a fire. Also, the heating of the container may cause an outbreak of fire. In lunar conditions due to the gradient of temperature during the day this may pose a problem, therefore, caution should be taken when storing the liquid NTO.

O/F Optimization

Using the CEA program, I've performed a study on the O/F ratio with the given propellant properties, along with the chamber pressure of 110 psi and pressure ratio of 10,000 since the ambient environment is a vacuum. The simulation was performed with the Equilibrium option, and the post-processing of the results was done via Python and MATLAB.

The performance of our propellant at different O/F ratios can be seen in Figure 1, which peaks around O/F= 1.6 which is in accordance with LMDE's ratio.

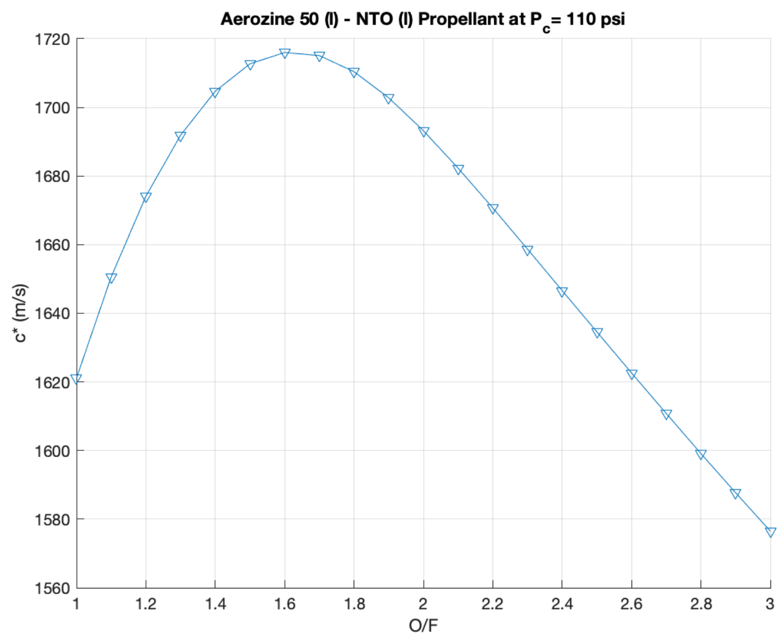

Figure 1. c* performance of the Aerozine 50 - NTO propellant curve with respect to the oxidizer-fuel ratio

Although, c* is the metric that we should ideally maximize, we should also consider the effect of the O/F ratio on the Isp values so that we find a sweet spot such that our propellant is powerful and the nozzle ratio is not extremely high such that it would yield a bulky nozzle.

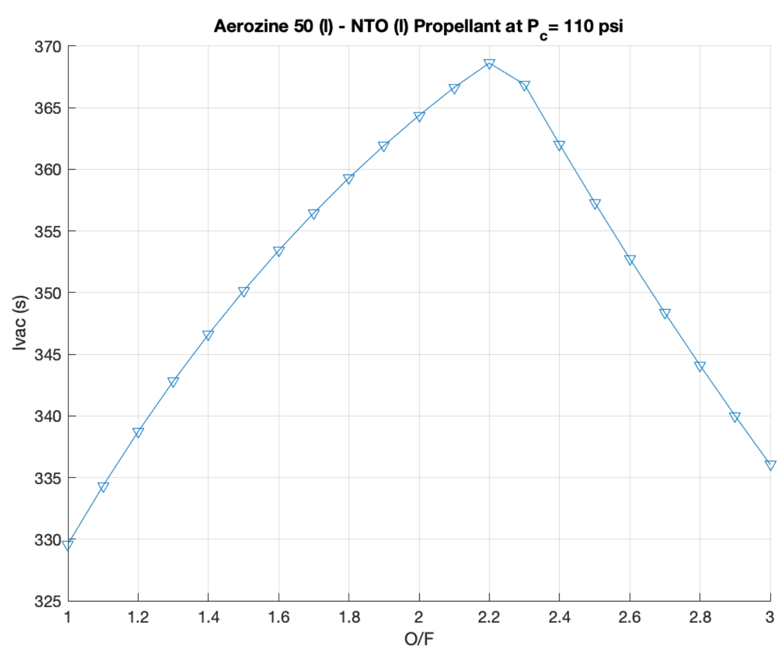

Figure 2. Vacuum Isp of the Aerozine 50 - NTO propellant curve with respect to the oxidizer-fuel ratio

Keeping in mind that the vacuum Isp peaks around O/F= 2.2 as can be seen in Figure 2, we can select our operational O/F ratio to be 1.8 so that we are not sacrificing so much propellant performance while keeping the Isp value at a reasonable range.

After selecting the operational O/F ratio as 1.8, we can conduct another study on the expansion rate so that we can design our nozzle and set the expansion rate as our liking to perform the best given the constraints. For this, we again use the NASA CEA program by fixing the O/F= 1.8 and sweeping for different nozzle ratios.

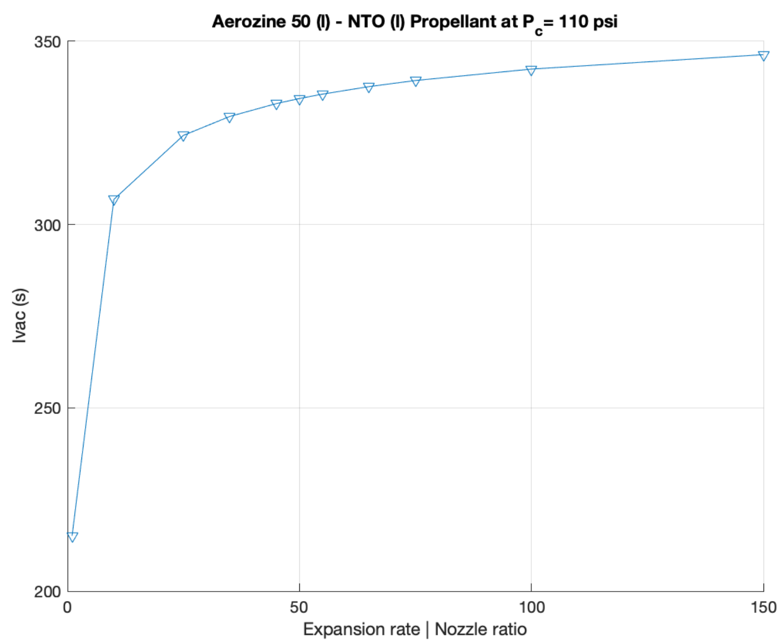

Figure 3. Vacuum Isp of the Aerozine 50 - NTO propellant curve with respect to the nozzle ratio

As can be seen in Figure 3, the vacuum Isp monotonically increases with increasing nozzle ratio yet the effect disappears with increasing nozzle ratio meaning that increasing the nozzle ratio after a spot would only increase the nozzle mass which would impede our mission. To optimize this trade-off, I selected the nozzle ratio as 50 which yields Ivac= 334 s, which is 5% less than the nozzle ratio of 150. This selection is again in accordance with the LMDE's nozzle ratio specifications, although not the same.

To conclude, the Aerozine 50 and NTO were selected as the propellant due to their ignition capabilities, storability, high performance, and historical success. The O/F ratio was optimized and selected as 1.8 for this mission yielding a high c* while not compromising the Isp. The expansion rate was optimized to be 50 that best utilizes the ejected gas thrust while not compromising the mass requirements and causing bulkiness. When another CEA inquiry is done on the final specifications we get the performance metrics as,

$$Ivac = 335\,s \qquad Isp = 320\,s \qquad c^* = 1710.4 \qquad C_F \Rightarrow (0.65 - 1.84)$$

Design of the Rocket

The design of the liquid engine for this project involves several key components, including the nozzle, fuel, and oxidizer tanks, and combustion chamber. The nozzle is responsible for shaping and directing the exhaust gases and must be carefully designed to ensure optimal performance. The fuel and oxidizer tanks must be able to withstand the pressures and temperatures of the engine, while also being able to accurately meter the flow of these fluids into the combustion chamber. The combustion chamber is where the fuel and oxidizer are mixed and ignited and must be able to withstand the high temperatures and pressures generated during operation. At this stage, the delta-V requirement for the engine has not yet been computed, so the design process is an iterative one. As the delta-V requirement becomes clearer, the design of these components may be refined to better meet the needs of the engine.

For this stage of the design, we can take the delta-V required for this mission to be 2300 m/s as a result of the preliminary design. The O/F ratio and the expansion rates were already determined in the previous section as O/F= 1.8 and ε= 50. By using the delta-V requirement, reserve mass constraint, and the dry mass; the required wet mass is found to be around 600 kg for the final trajectory. Using the O/F ratio of 1.8 and the maximum thrust information of the SE-10 engine,

$$\dot{m}_{oxidizer} = 2.15\,kg/s \text{ and } \dot{m}_{fuel} = 1.20\,kg/s$$

Nozzle and Combustion Chamber

For the design of the nozzle, de Laval type nozzle is adapted and the geometric properties of the throat and the exit is required for the design of the nozzle. Using the c* value, propellant flow rate, chamber pressure, and the expansion ratio we get,

$$A_{throat} = 75.6\ cm^2\ and\ d_{throat} = 50\ mm;\ A_{exit} = 3850\ cm^2\ and\ d_{exit} = 350\ mm,$$

$$L_{nozzle} = 1.12\ m$$

Another parameter for the nozzle design is the effective divergence angle, which for sake of performance and mass budget was chosen as 15 degrees, which yields with 98% nozzle efficiency. Graphite epoxy is a material that is made of graphite fibers and epoxy which is a great fit for the nozzle material because it is strong, lightweight, and can handle high temperatures. It is also less expensive than other materials and easy to manufacture. The average density for Graphite epoxy resin is 2500 kg/m^3. The total nozzle mass comes up to 7.5 kg.

The combustion chamber design is a hard process involving many analyses and simulations. For simplicity, a characteristic chamber length was found for a Hydrazine-NTO system and I designed the chamber radius to be, chamber length to be 135 mm with a 2-1 elliptical inlet curvature. The design geometry can be seen in Figure 5.

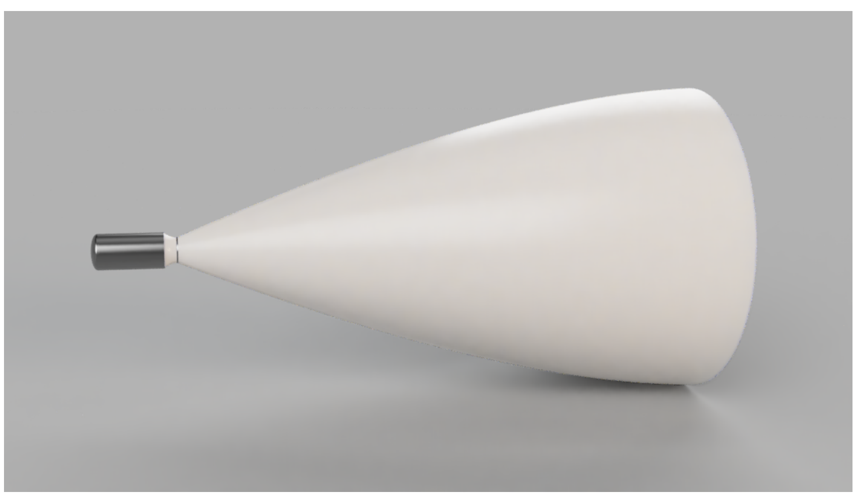

Figure 4. CAD design of the combustion chamber and the nozzle, where the expansion rate is 50

Haynes 188 is a suitable material for the combustion chamber of a rocket engine using Aerozine-50 as a propellant. This nickel-chromium-tungsten alloy has a high melting point of approximately 1800 K, which is suitable for the high temperatures generated during combustion. Additionally, Haynes 188 has excellent corrosion resistance properties, which makes it resistant to the corrosive nature of the propellant. This material also has a relatively low density of approximately 8300 kg/m³, this helps to keep the weight of the combustion chamber down. At high temperatures, around 1360 K, the ultimate strength, and the yield strength of the Hayness 188 drop down to $\sigma_{strength} = 120\, MPa$, $\sigma_{yield} = 65\, MPa$. Using a safety factor of 5, the chamber thickness is found to be 2 mm. So the mass of the chamber is approximately 0.6 kg.

Oxidizer, Fuel, and Pressurizer Tank Design

For the design of the tanks, the tank pressures must be selected, according to considering various parameters which all affect the design and the performance of the system. Again, for simplicity, I adapted the fuel and oxidizer tank pressures from the SE-10 engine, as 225 psi (1.55 MPa). And since the system operates on a Pressure-Fed cycle, there must also be a pressurant tank. The pressurizer gas is selected as cryogenic Helium due to its low molecular weight and reluctance to react, combust, and explode; and the pressure of the He tank is adapted as 4500 psi (31 MPa). Using the final trajectory results, the oxidizer and the fuel mass are found to be

$$m_{ox} = 385\, kg,\ \ m_{fuel} = 215\, kg.$$

NTO oxidizer and Aerozine-50 fuel can both be stored safely in steel and aluminum tanks, therefore the oxidizer tank material is selected to be Steel 316L due to its good corrosion resistance and mechanical properties. The 316L alloy has a density of approximately 8300 kg/m³, and has good ultimate strength and yield strength, with a minimum of 485 MPa and 170 MPa respectively.

The oxidizer and the fuel tanks are cylindrical with 2-1 ellipse top and hemispherical bottoms, and using a safety factor of 5 the oxidizer and fuel tank geometry takes the form,

$$V_{ox} = 265L,\ V_{fuel} = 240L;$$

$$t_{ox,tank} = 1\, mm,\ t_{fuel,tank} = 1\, mm;$$

$$L_{ox,tank} = 125 + 1100 + 250\, mm = 1375mm = 1.475m$$

$$L_{fuel,tank} = 125 + 1000 + 250\,mm = 1375mm = 1.375m$$

$$m_{ox,tank} = 20.0\,kg, m_{fuel,tank} = 18.2\,kg;$$

The pressurant is cryogenic supercritical Helium and to find the mass I referred to the Apollo Lunar module and scaled their propellant use to find, $m_{He} = 0.75\,kg$. For this high pressure tank application, the 7075-T6 aluminum alloy was chosen. The alloy has a density of 2810 kg/m³, which is lower than most metals and makes it an efficient option for high-pressure applications where weight is a concern. The strength of the 7075-T6 aluminum alloy is also very high, with a yield strength of 503 MPa and a tensile strength of 572 MPa. Using a safety factor of 1.5 the radius, wall thickness, and mass are found to be,

$$r_{He,\,tank} = 115\,mm,\ t = 10\,mm,\ m_{He,\,tank} = 2.5\,kg$$

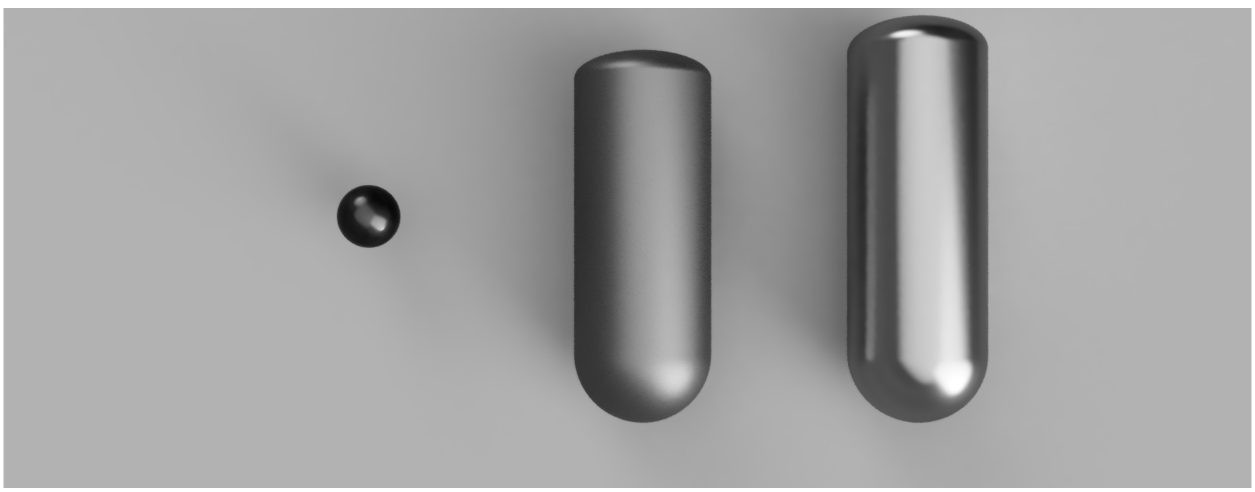

Figure 5. CAD design of the propellant and pressurant tanks, a- He tank, b- Fuel tank, c- Oxidizer tank

Besides the tanks, nozzle, and pressurant, there are many components such as the supports, slosh baffles, electronic and mechanical components such as the injectors, flight control and etc. Since this is a very detailed job, I approximated the extra masses as below. For supports, fiberglass is used due to its high strength-to-weight ratio, and for the body of the vehicle. Furthermore, for the body of the vehicle, Aluminum 7075 is used because of its high strength-to-weight ratio, good thermal stability, and ability to withstand extreme temperatures.

Table 1. Mass budget of the Lunar landing mission

Component	**Material**	**Mass (kg)**
Nozzle	Graphite epoxy	7.5
Combustion Chamber	Haynes 188 Alloy	0.6
Fuel Tank	Steel 316L Alloy	18.2
Oxidizer Tank	Steel 316L Alloy	20
Pressurant & Pressurant Tank	He & 7075-T6 Alloy	3.25
Supports & Vehicle Body	Fiberglass & Al 7075	95
Valves & Injectors	Steel	5
Electronics & Batteries & Solar Panels	Electronics	50

The total structural mass sums up to 200 kg, which will be used in the simulations and further calculations.

Survivability on Lunar Surface

In order to ensure the survivability of the descent vehicle on the lunar surface, various measures must be taken into consideration. Firstly, a battery system must be implemented to ensure continued communication with the rocket throughout its months of operation on the lunar surface. Solar panels are the most suitable option for providing electricity to the battery system, as they utilize the sun's resources and do not consume any of the rocket's fuel. An array of lithium-ion batteries is a suitable choice due to their efficiency and lower risk of explosion. Also, the materials used in the construction of the descent vehicle must be able to withstand the harsh conditions of the lunar surface. They must be able to withstand high temperatures, radiation, and micrometeoroids. The vehicle must also be equipped with a thermal control system to ensure that the internal temperature remains within the operational range of its systems.

Mission Operation, Staging, and Trajectory Analysis

Mission and the Governing Equations

Trajectory analysis play a crucial role in the design and execution of a lunar landing mission. The goal of this analysis is to predict the rocket's position, velocity, and acceleration as the rocket approaches the Moon, and to use this information to control the rocket's trajectory and ensure a safe landing on the lunar surface taking the mission constraints into account. For simplicity, the governing equations are in 2 degrees of freedom, and the expectation is to control the thrusting and thrust vectoring. To achieve this, there are various methods such as optimum control, solving analytically via approximations, and iteratively and strategically changing the control parameters.

The motion equations were derived via numerous simplifications and assumptions such as no eccentricity and no rotation, and they take the general form,

$$\frac{dV}{dt} = \frac{T \cdot cos(\alpha)}{M} - \frac{D}{M} - \frac{\mu \cdot sin(\gamma)}{r^2}$$

$$\frac{d\gamma}{dt} = \frac{V \cdot cos(\gamma)}{r} + \frac{T \cdot sin(\alpha)}{V \cdot M} - \frac{L}{V \cdot M} - \frac{\mu \cdot cos(\gamma)}{V \cdot r^2}$$

The atmospheric density of the Moon is practically zero, we can neglect the terms including the drag and lift forces. So the final form of the governing equations is,

$$\frac{dV}{dt} = \frac{T \cdot cos(\alpha)}{M} - \frac{\mu \cdot sin(\gamma)}{r^2}$$

$$\frac{d\gamma}{dt} = \frac{V \cdot cos(\gamma)}{r} + \frac{T \cdot sin(\alpha)}{V \cdot M} - \frac{\mu \cdot cos(\gamma)}{V \cdot r^2}$$

$$\frac{dM}{dt} = - \frac{T}{I_{sp} \cdot g_0}$$

Table 2. The parameters of the governing equations

Symbol	Description	Units
V	Velocity	m/s
M	Mass of the vehicle	kg

r	Radial position from the center of the Moon	m
γ	Trajectory angle with respect to the horizon	rad
T	Thrust force	N
α	Thrust velocity/vectoring angle	rad
D, L	Drag and Lift forces	N
t	Time	s
μ, Isp, g0	Gravitational parameter, Specific Impulse, Earth gravity	$\frac{m^3}{s^2}$, s, m/s^2

Therefore, the governing equations have two degrees of freedom, V and γ; and there are two control parameters, T, α. Where the change of T and α in time dictates the motion of the rocket, therefore a successful landing requires robust control of the thrust and vectoring.

To perform this analysis, I used MATLAB, which allows us to simulate the rocket's motion and control its trajectory in a virtual environment. We start by specifying the initial conditions of the rocket, such as its position, velocity, and orientation, and then use the 2 DOF equations to predict its motion over time. We can then use time marching to integrate these equations and find the rocket's trajectory and the effect of the control parameters. For time marching a first-order model (Euler method) was used, and the control strategy was adapted from other landing missions.

The flight strategy was composed of four stages: entry, cruise, braking, and landing. All stages serve different purposes with distinctive properties. In this project, I came up with two different trajectories with different staging parameters. Both of the trajectories will be elaborated on but the general purpose and the control of the stages can be found on the next page.

Staging Outline of the Lunar landing Mission

Stage	Purpose	Ending Trigger	Control Parameters*
Entry stage	To transfer the rocket into a closer trajectory from the initial circular trajectory.	When the velocity is reduced to 50% of the initial circular orbit velocity.	$\dot{m} = 0.6 \cdot \dot{m}_{max}$** TVC= 180°, (No gimballing)
Cruise stage	To gain gravity turn and approach vertical orientation while getting closer to the lunar surface, without any propellent expenditure.	When the altitude of the rocket is less than 50 km above the Lunar surface.	$\dot{m} = 0$, (No burn) TVC= 180°, (No gimballing)
Braking stage	To prepare for the landing stage and reduce the velocity so that the landing stage can be controlled easily.	When the altitude of the rocket is less than 20-25 km above the Lunar surface.	$\dot{m} = 0.1 \cdot \dot{m}_{max}$ TVC= 180-186°, (gimbal range 0-6°)
Landing stage	To land with a very low velocity with a vertical alignment, in accordance with the mission requisites.	When the rocket touches the ground, meaning the altitude is less than 0 km.	$\dot{m} = (\frac{V}{V_{ref}})^{p} \cdot c \cdot \dot{m}_{max}$*** X: $\dot{m}_{strong} = (\frac{V}{V_{ref}})^{1.91} \cdot 0.6 \cdot \dot{m}_{max}$ Z: $\dot{m}_{weak} = (\frac{V}{V_{ref}})^{2.10} \cdot 0.4 \cdot \dot{m}_{max}$ TVC= 180°
Touchdown stage	To check if the mission requirements are satisfied successfully and the vehicle is on the ground.	• more than 10% of the initial fuel is remaining. • landing velocity is less than 1 m/s. • alignment is 90° with the ground.	No control

* The $\dot{m}_{max}$ term is found using propulsion system capabilities, and the TVC is equal to the gimbal angle since the rocket is aligned with the velocity vector.

** The throttle was set to either 100% or between 10-60% due to corrosion risk, so the flow rate is bounded between 10-60% of the $\dot{m}_{max}$.

*** V_{ref} is the velocity starting the landing stage, p is the velocity ratio power which measures the aggression, and c is the throttle ratio.

The trigger conditions, throttling, and thrust vector controlling were mainly found using trial and error along with boundary value solutions. In the solution, there were some approximations regarding the structural mass, propellant mass, and efficiency, which will be explained in the next part.

Simulation and Results

As explained in the previous parts, the simulation was solved with a time-marching method where I used Euler's method over the governing equations. Initially, there were some parameters that were unknown, such as the delivered Isp, structural mass, propellant mass, delta-V, and the maximum thrust. I assumed that the structural mass was 800 kg which is actually an overdesign considering the payload being 200 kg. If we need to reduce the dry load, we can just scale the propellant mass and keep the stages the same. Another assumption was that the efficiency of the propulsion system was 90% so the delivered Isp was 305s, which again is an overdesign since the efficiency rates for liquid rockets tend to be higher. And finally, for the maximum thrust, I used 45 kN which is the full-throttle thrust of the inspired engine SE-10. The time domain was discretized with 1-microsecond steps which allowed convergence in the numerical integrals.

Trajectory X

This trajectory was solved for Isp= 335, meaning that the overall efficiency of the propulsion system is 100%. The dry mass and the maximum thrust variables are as mentioned above, and the wet mass was chosen such that the final propellant mass is higher than the reserve capacity requirement. This trajectory serves as a preliminary study so that some parameters can be found for use in the design part. The motion variables can be found in Figure 6.a-e, the control parameters can be found in Figure 6.f-h, the mission panel can be found in Figure 7, and the delta-V budgeting can be found in Figure 8.

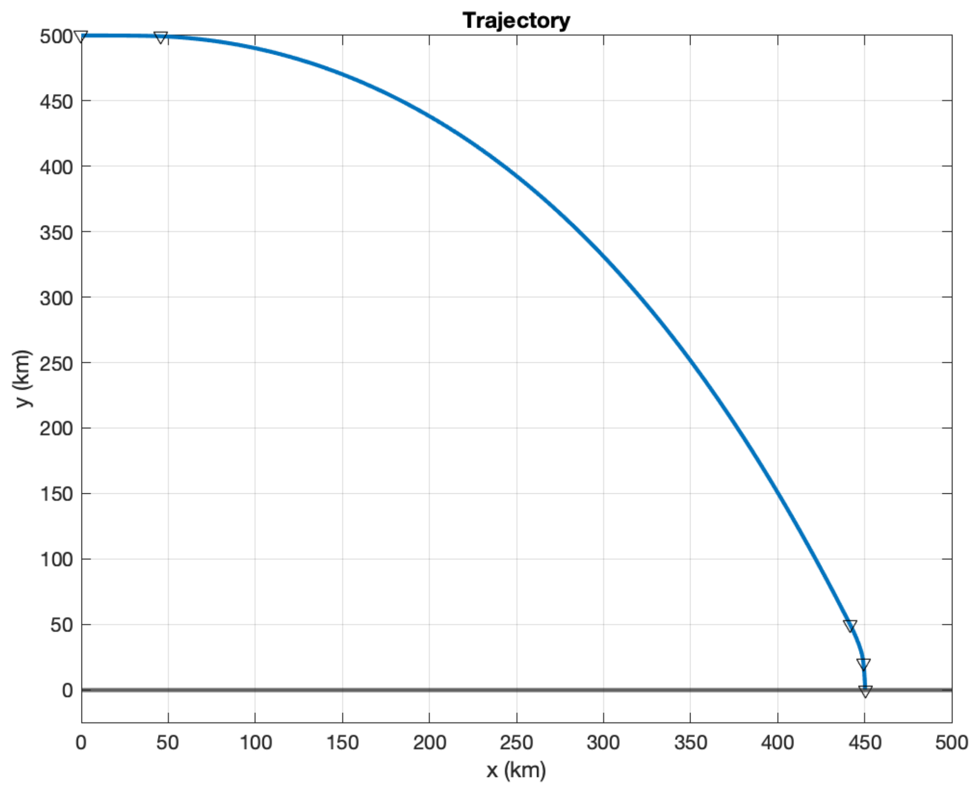

Figure 6-a.: Trajectory of the rocket through the lunar horizon, where the triangles indicate the stages

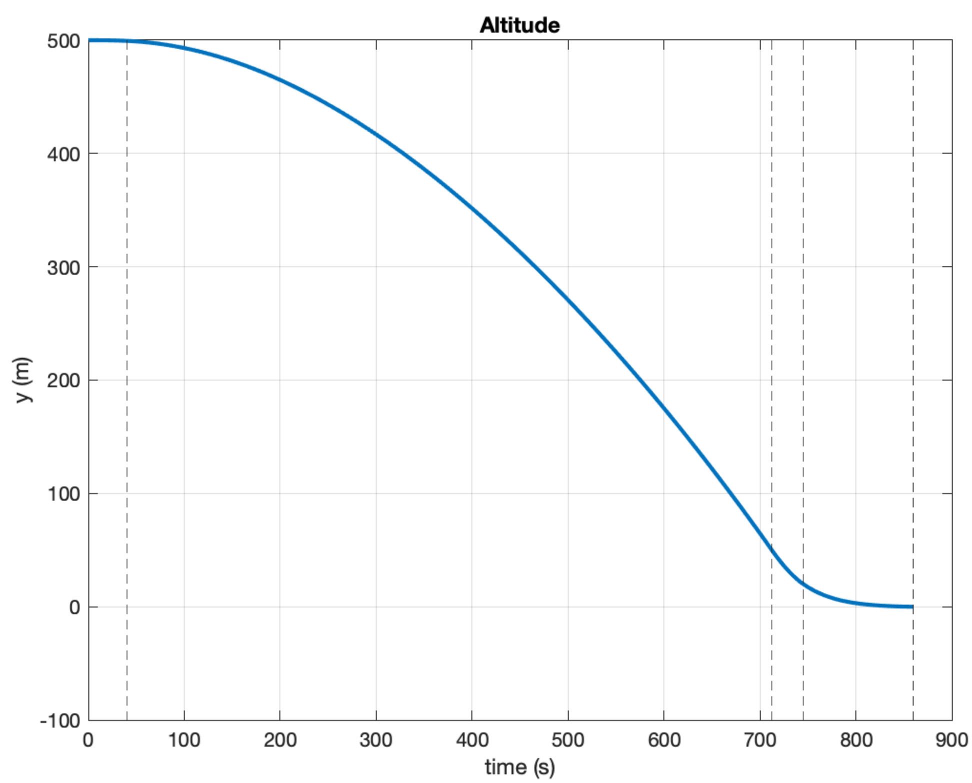

Figure 6-b. Altitude profile of the rocket throughout the mission, where the vertical lines indicate the stage boundaries

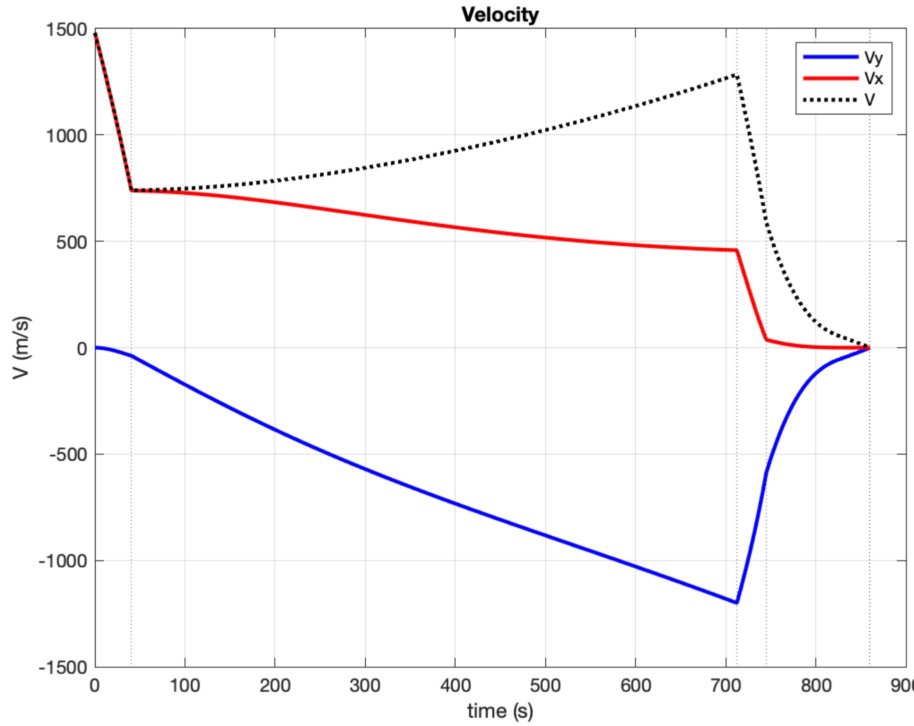

Figure 6-c. Velocity profile of the rocket through the mission, where the red line is for the velocity parallel to the horizon, blue for radial velocity and dotted line for the total velocity

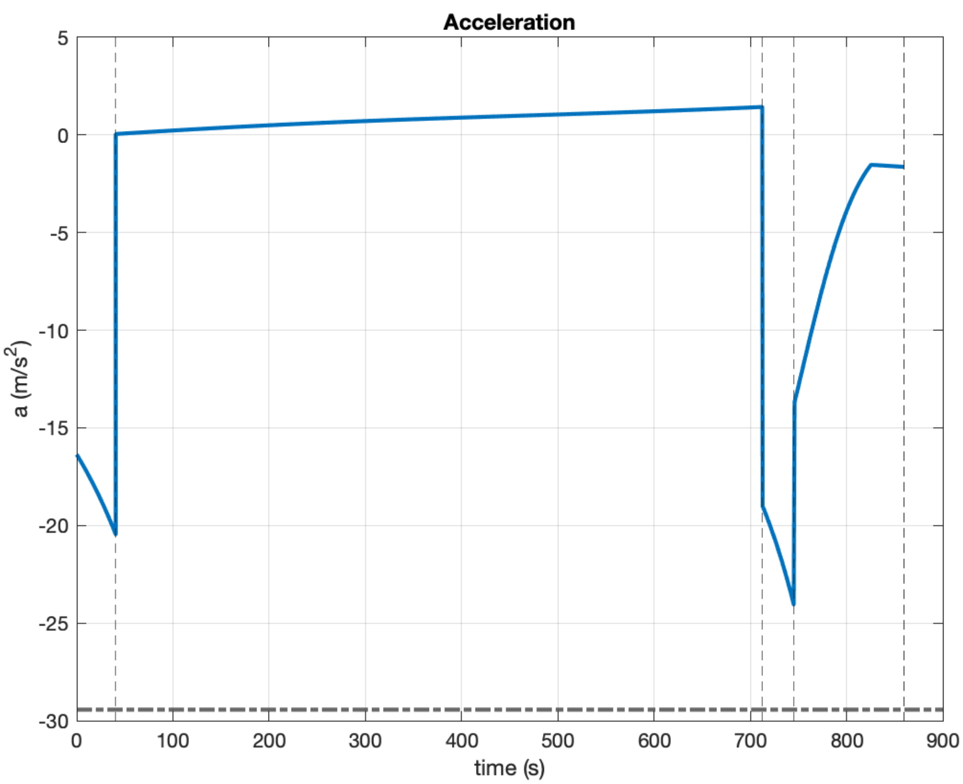

Figure 6-d. Acceleration profile of the rocket through the mission, where the dotted horizontal line indicates the 3g constraint

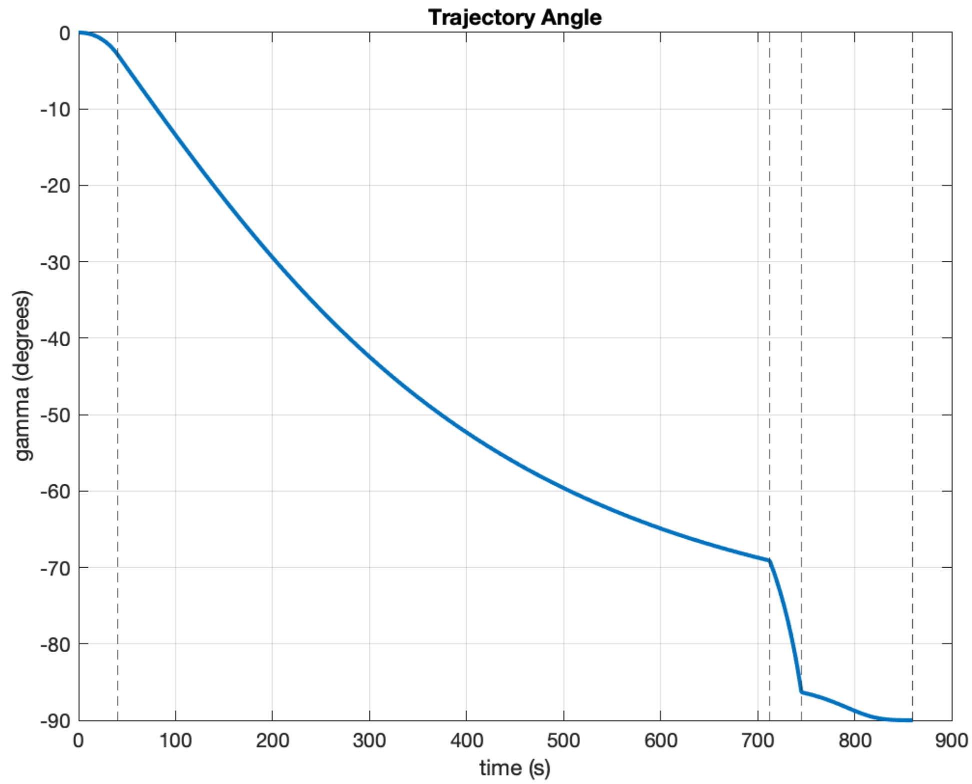

Figure 6-e. Trajectory angle profile of the rocket throughout the mission

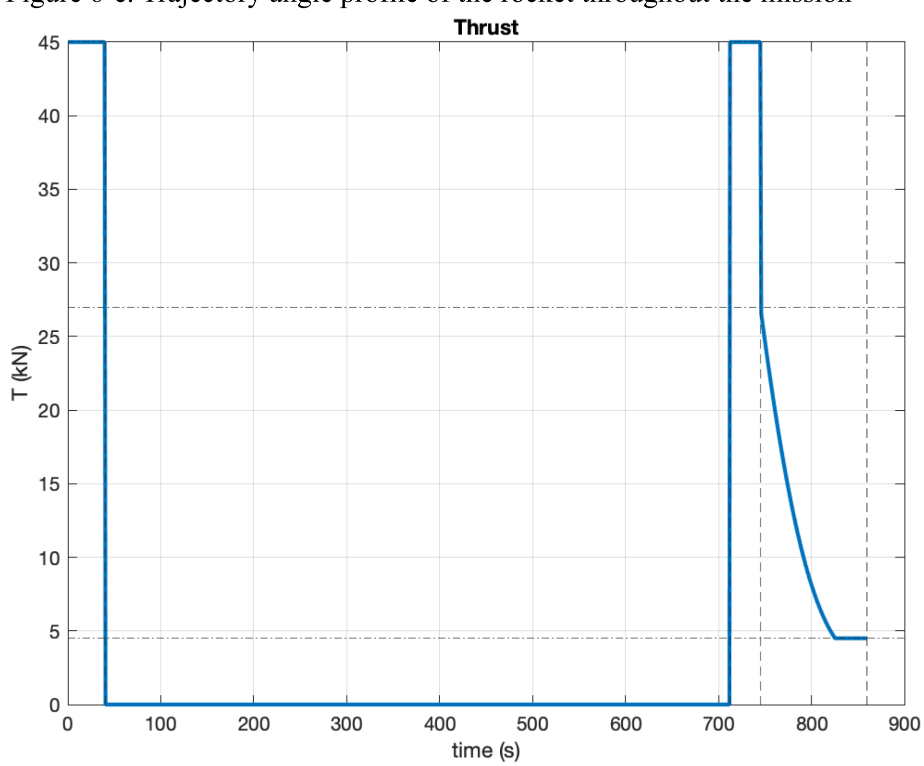

Figure 6-f. Thrust profile of the rocket through the mission, where the dotted horizontal lines indicate the throttle range of 10-60%

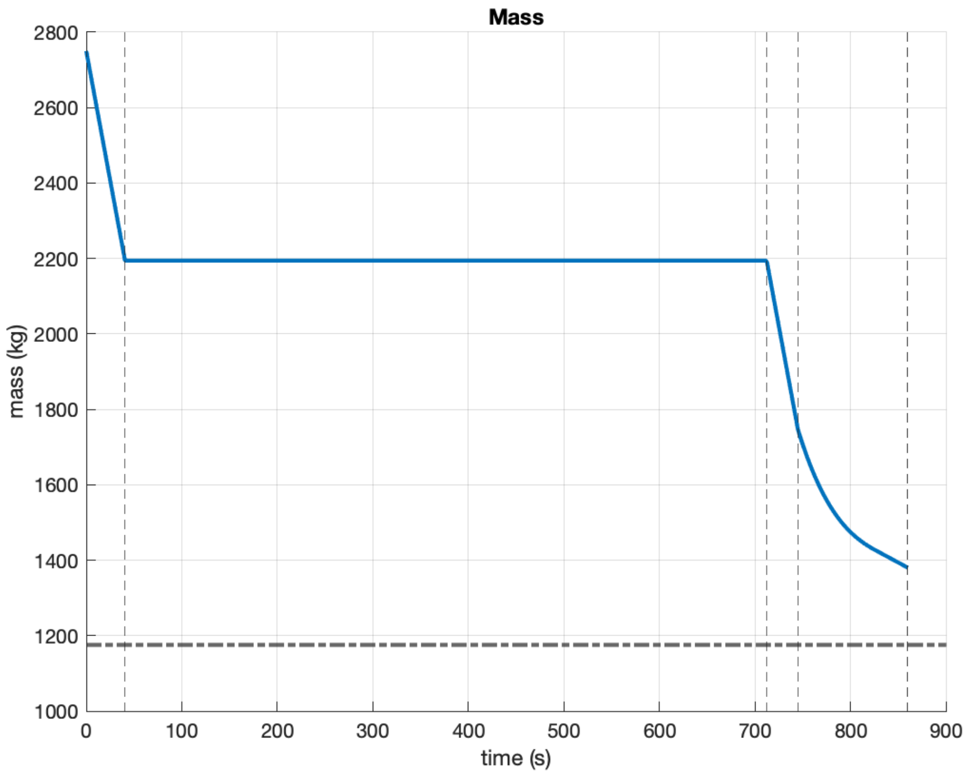

Figure 6-g. The mass of the rocket throughout the mission, where the horizontal dotted line indicates the reserve mass

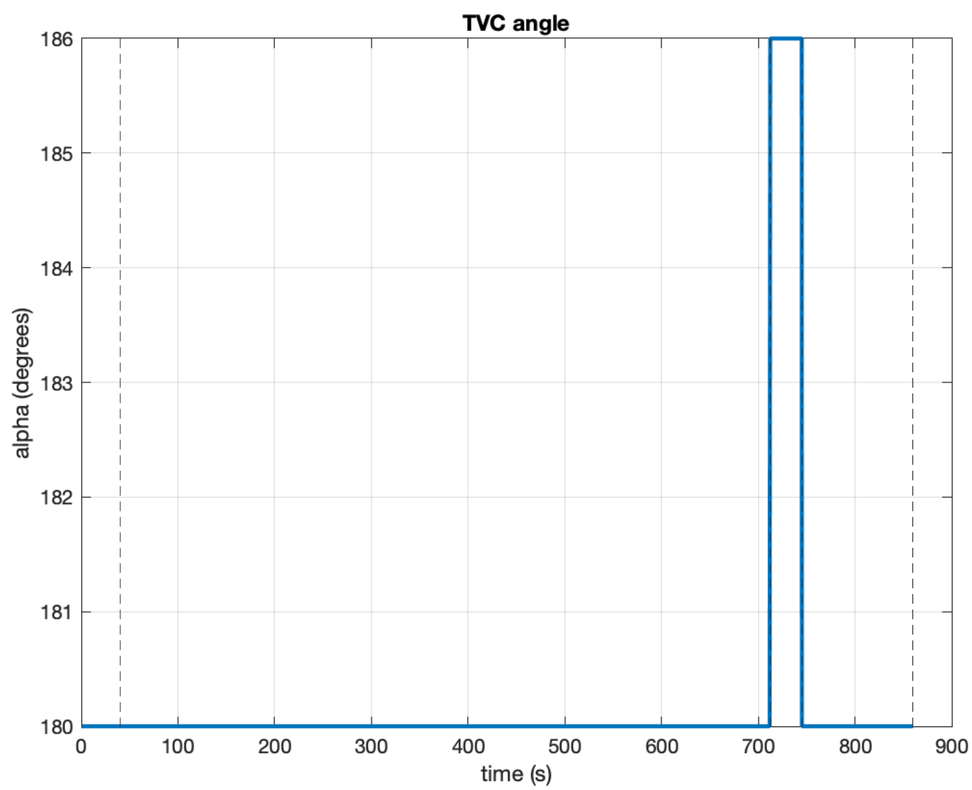

Figure 6-h. TVC angle profile where the gimballing at the brake stage is 6 degrees

```
Entry Phase Sucessful at t= 40.56 s
Cruise Phase Sucessful at t= 712.36 s
Brake Phase Sucessful at t= 745.10 s
Landing Sucessful at t= 860.13 s
TOUCH DOWN at t= 860.14 s

 ********************* Touchdown State *********************
Velocity: V= 0.75 m/s, Trajectory angle: gamma= -90.0°
Landing mass: m= 1380.0 kg, Reserve fuel rate: 0.22
Mission delta-V: 2265.21 m/s
```

Figure 7. Mission panel of the Trajectory X, consisting of the staging information and touchdown state variables

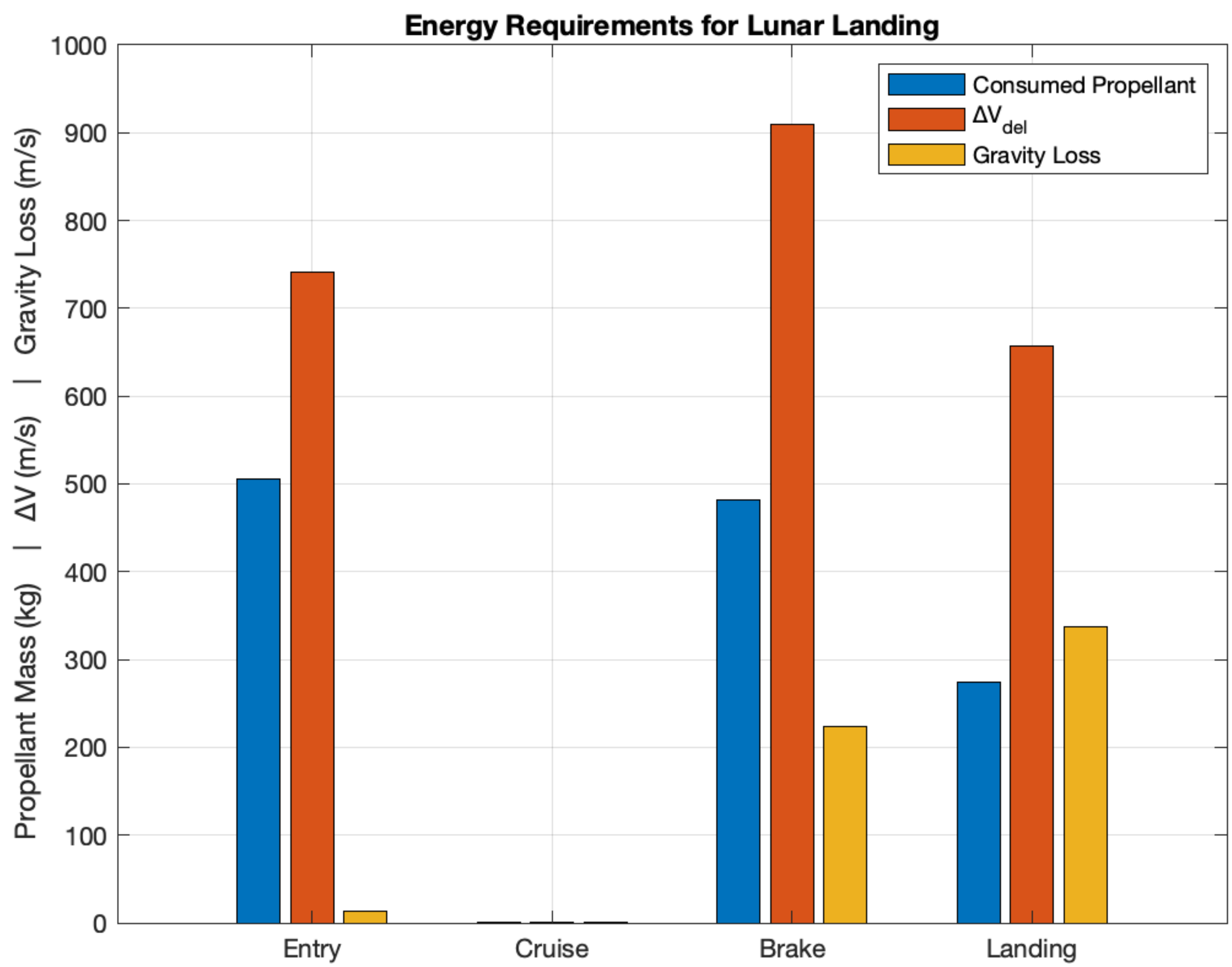

Figure 8. The delta-V budget of Trajectory X

Trajectory Z

This trajectory was solved for Isp= 305, unlike the Trajectory X, meaning that the overall efficiency of the propulsion system is 90%. The dry mass was reduced to 400 kg and wet mass to 600 kg after the initial design process. In this trajectory, the entry burn is not optimized therefore the gravity loss is greater, however, the landing is more optimized reducing the overall delta-V expenditure. The motion variables can be found in Figure 9.a-e, the control parameters can be found in Figure 6.f-h, the mission panel can be found in Figure 10, and the delta-V budgeting can be found in Figure 11.

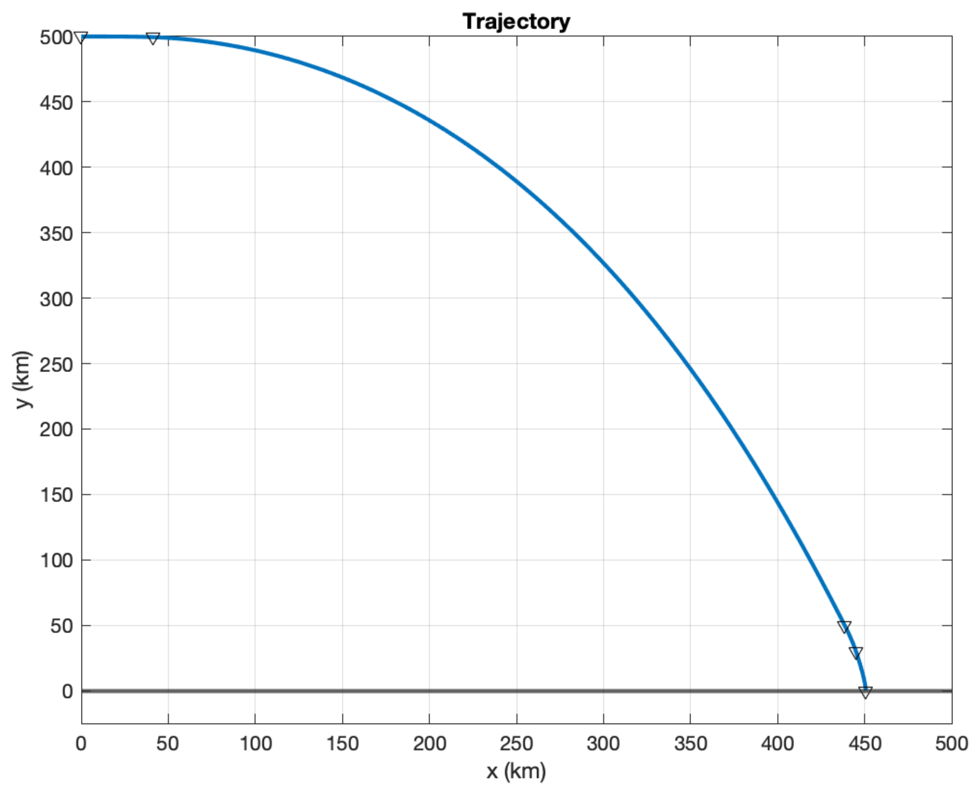

Figure 9-a.: Trajectory of the rocket through the lunar horizon, where the triangles indicate the stages

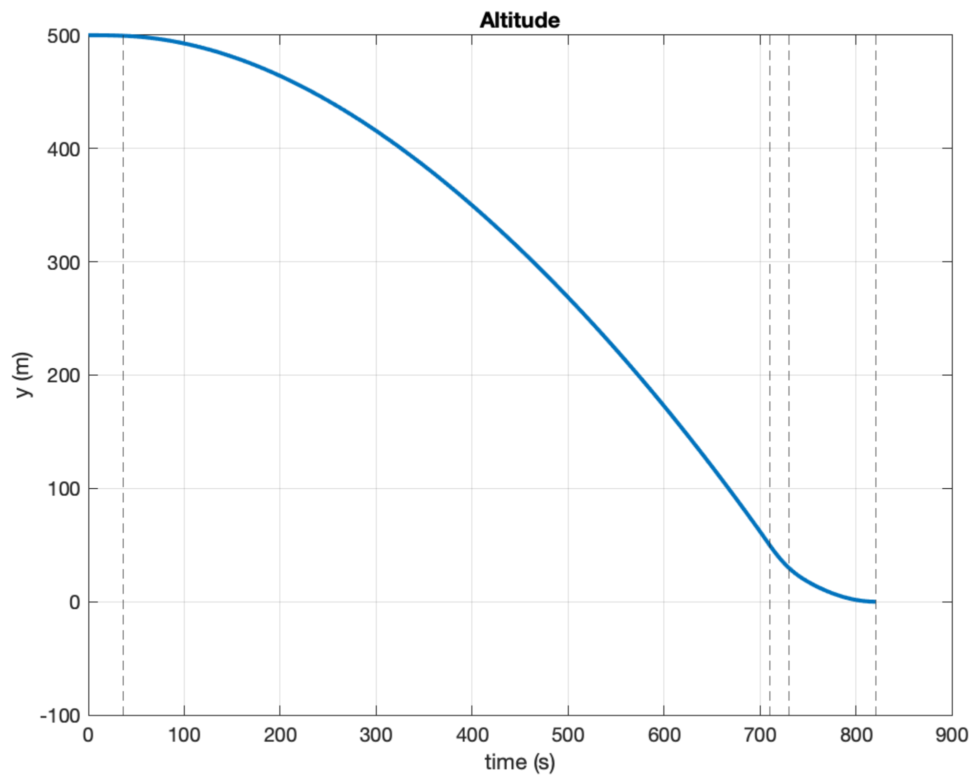

Figure 9-b. Altitude profile of the rocket throughout the mission, where the vertical lines indicate the stage boundaries

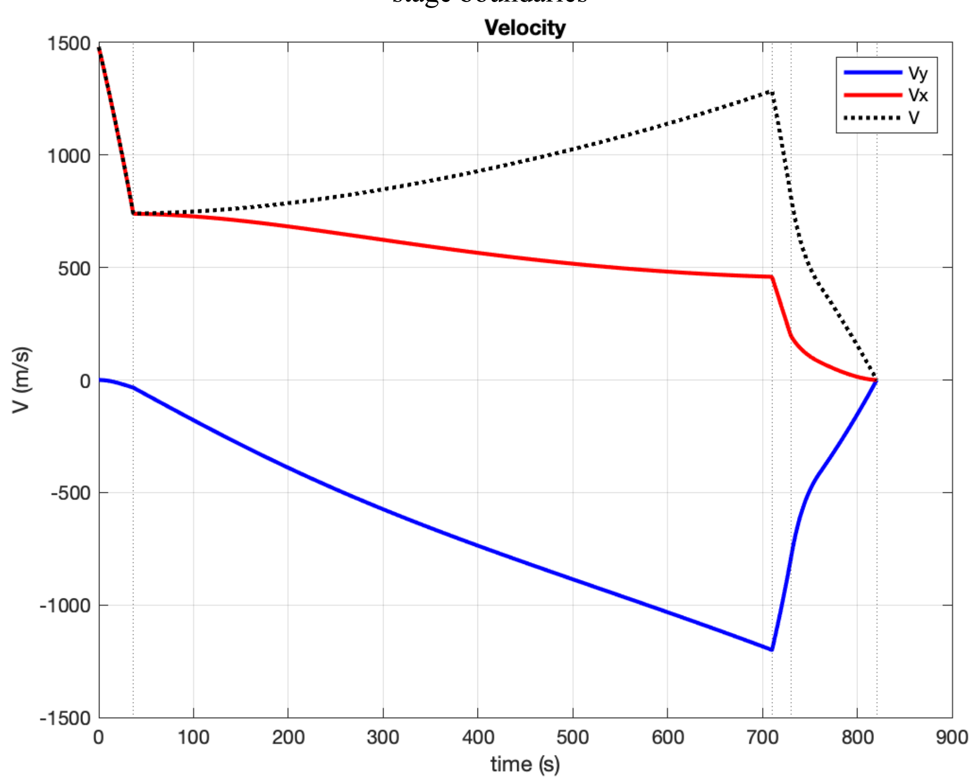

Figure 9-c. Velocity profile of the rocket through the mission, where the red line is for the velocity parallel to the horizon, blue for radial velocity and dotted line for the total velocity

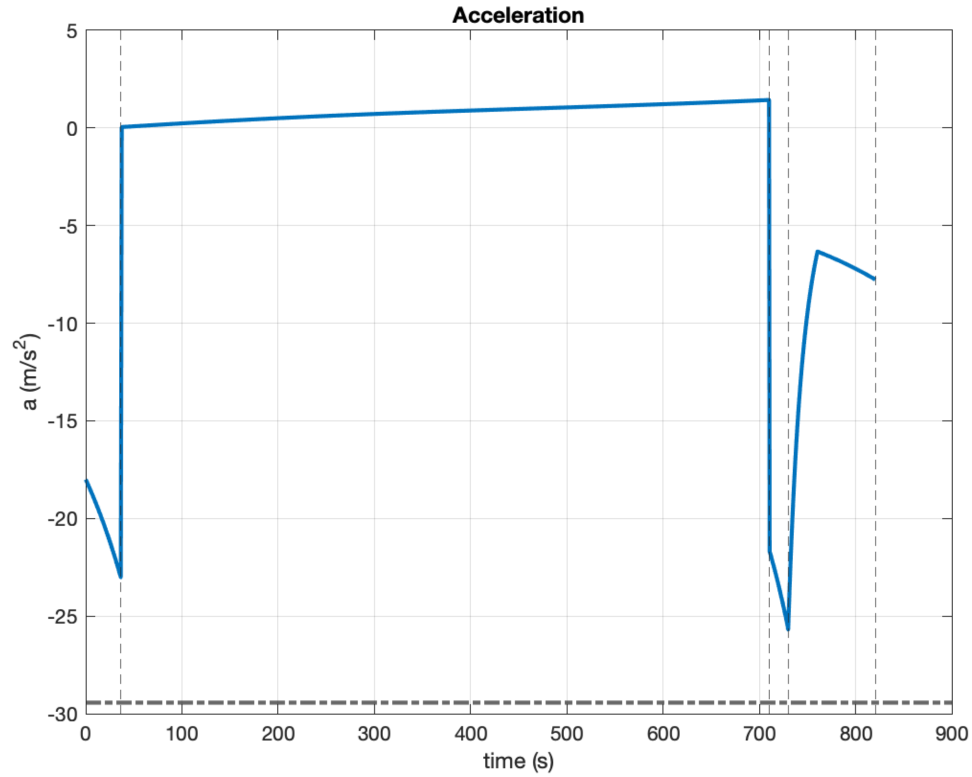

Figure 9-d. Acceleration profile of the rocket through the mission, where the dotted horizontal line indicates the 3g constraint

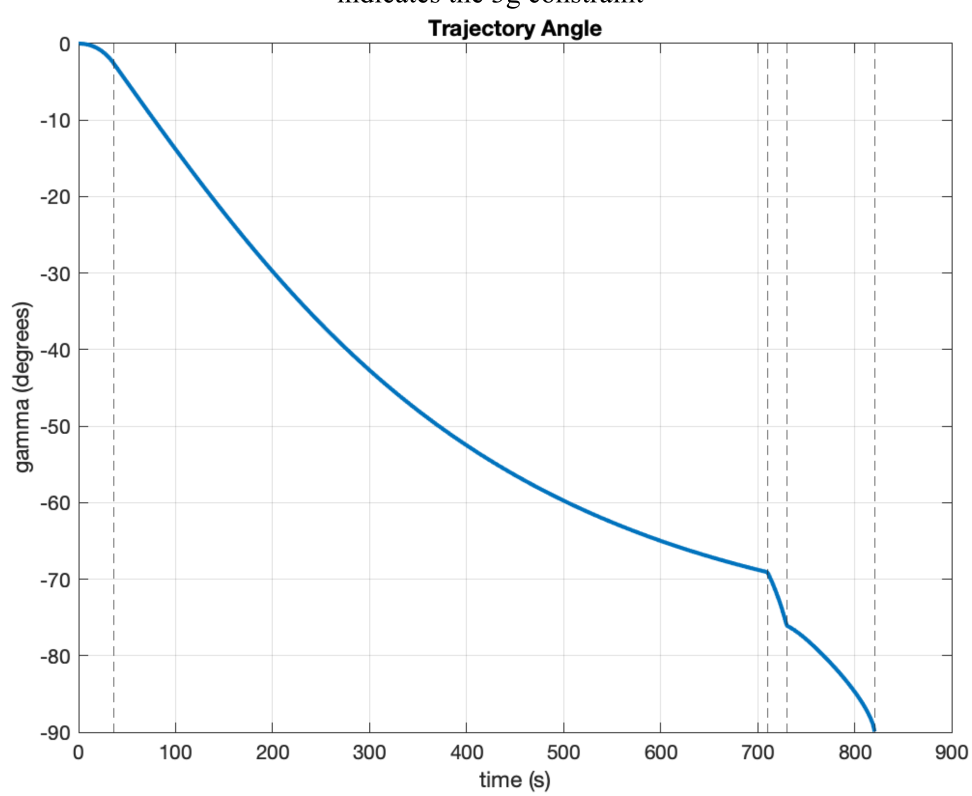

Figure 9-e. Trajectory angle profile of the rocket throughout the mission

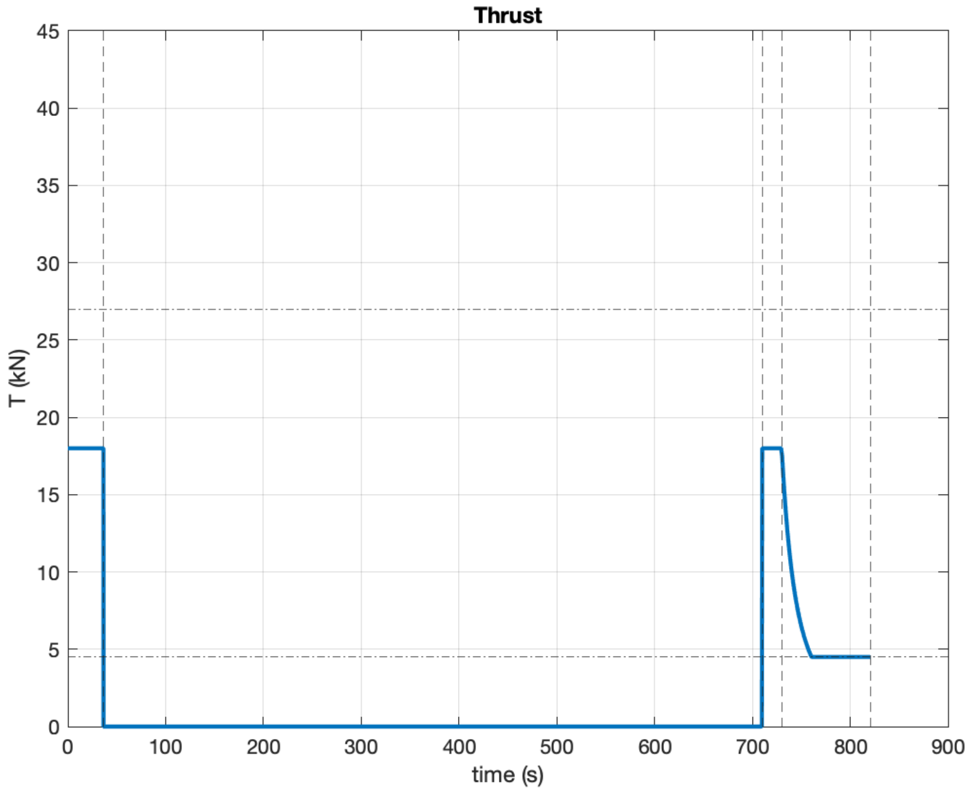

Figure 9-f. Thrust profile of the rocket through the mission, where the dotted horizontal lines indicate the throttle range of 10-60%

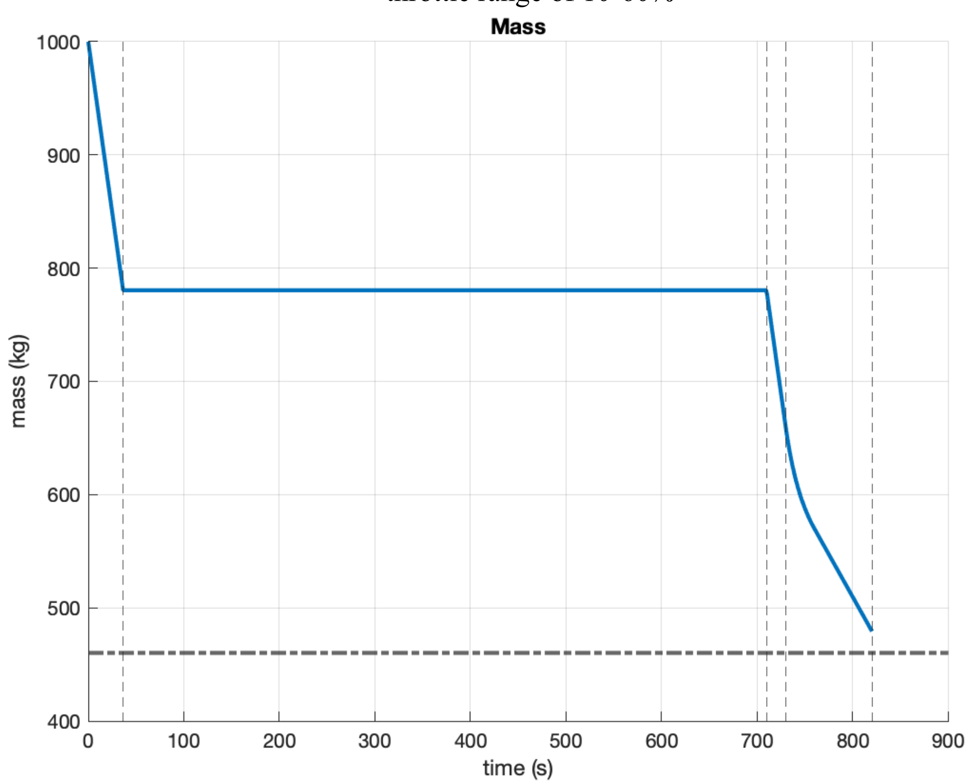

Figure 9-g. The mass of the rocket throughout the mission, where the horizontal dotted line indicates the reserve mass

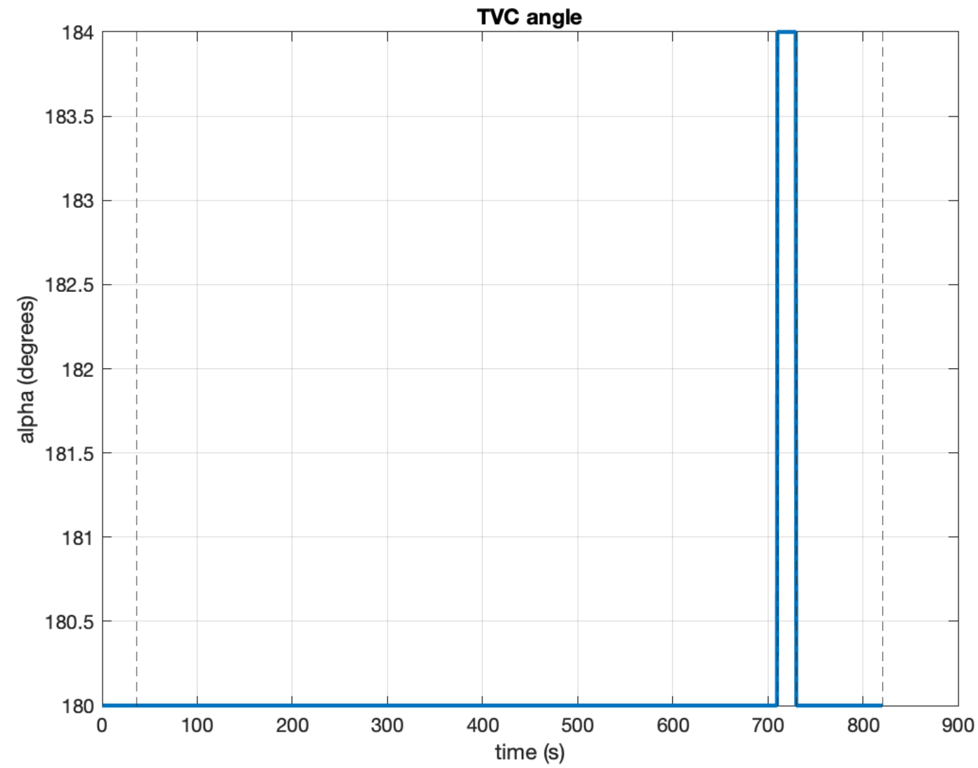

Figure 9-h. TVC angle profile where the gimballing at the braking stage is 4 degrees

```
Entry Phase Sucessful at t= 36.47 s
Cruise Phase Sucessful at t= 710.08 s
Brake Phase Sucessful at t= 729.96 s
Landing Sucessful at t= 820.52 s
TOUCH DOWN at t= 820.53 s

 ******************** Touchdown State ********************
Velocity: V= 0.16 m/s, Trajectory angle: gamma= -90°
Landing mass: m= 479.2 kg, Reserve fuel rate: 0.13
Mission delta-V: 2200.51 m/s
```

Figure 10. Mission panel of the Trajectory Z, consisting of the staging information and touchdown state variables

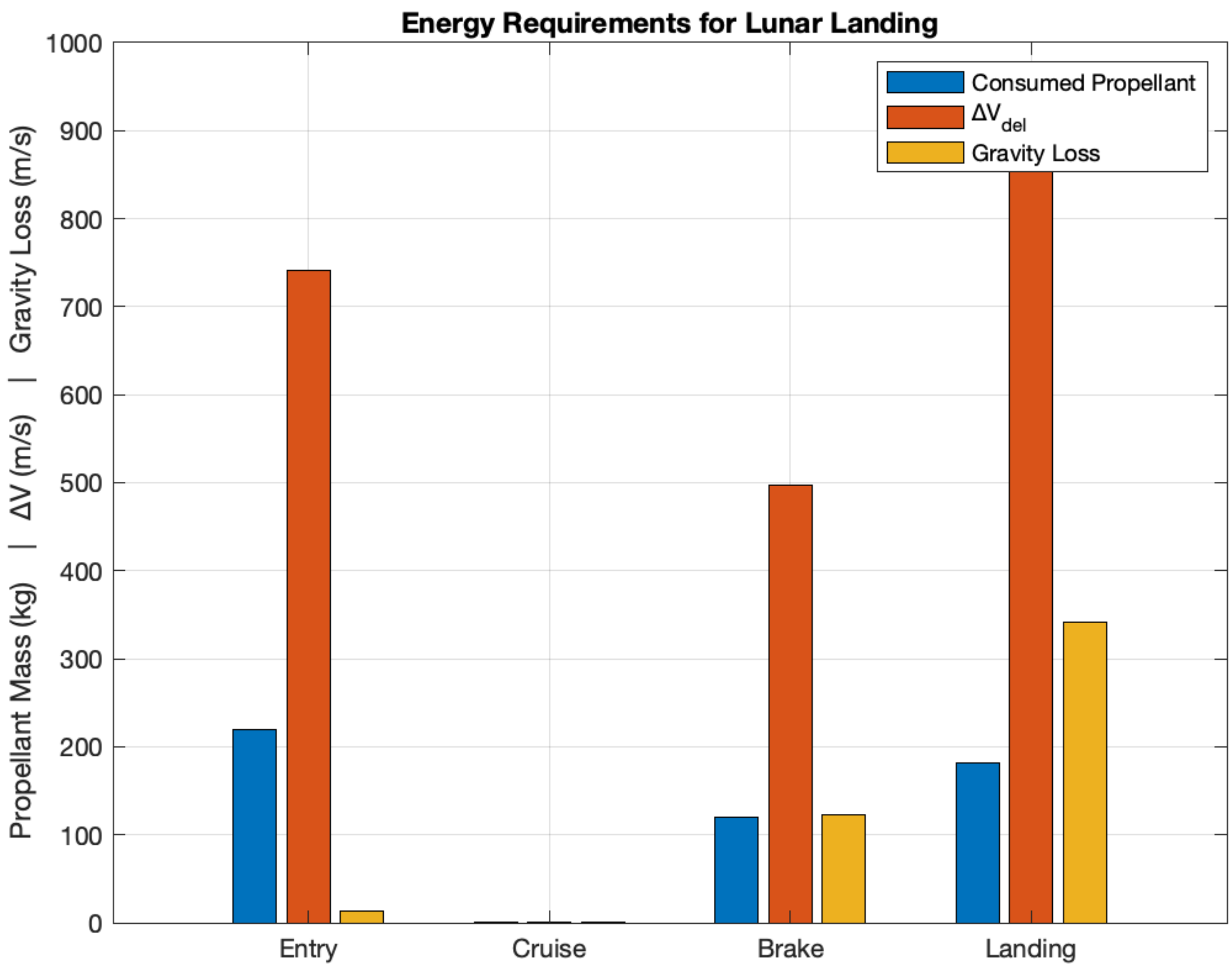

Figure 11. The delta-V budget of Trajectory Z

Table 4. delta-V budgeting of the Lunar descent mission

Stage	Delivered delta-V	Consumed Propellant
Entry stage	741 m/s	219.5 kg
Cruise stage	0.2 m/s	0 kg
Braking stage	497 m/s	119.5 kg
Landing stage	961.6 m/s	181.7 kg

The total delta-V requirement sums up to 2.2 km/s for the total mission, and the consumed propellant is 520 kg.

Discussion of the Results

The required delta-V from the Low Lunar Orbit (100 km) to the Lunar surface is 1.87 km/s, although my trajectories require around 2.2-2.3 m/s. The first reason is that our initial position is higher than the LLO where it takes more delta-V to compensate for the potential energy in between, and the second reason is that my control system is not very robust therefore there is some loss in the performance. However, after the design process, the wet and the dry masses dropped, overall reducing the delta-V requirement and yielding better performance. The Oberth effect can be utilized to better control the rocket, or some sort of optimal control algorithm can be utilized to minimize the delta-V requirement or the propellant consumption.

Executive Summary

In the project, I focused on meeting the design priorities of safety and minimizing hardware mass brought from Earth. To accomplish this, I considered the requirements for the Lunar Descent Vehicle, starting in a specific orbit, and ensuring a safe landing velocity and reserve fuel. I also took into account the constraints of the project, such as the maximum acceleration that the vehicle could endure. To design the vehicle, I had the freedom to choose the system architecture, number of stages, propulsion type and propellants, and number of firings. I also optimized the propulsion system for each stage in detail, including propellant selection, nozzle area ratio, chamber pressure, oxidizer to fuel ratio, and geometrical properties such as length, diameter, nozzle throat area, nozzle exit area, nozzle shape, and combustion chamber.

The design of the liquid engine for this project involves several key components, including the nozzle, fuel, and oxidizer tanks, and combustion chamber. The fuel and oxidizer tanks must be able to withstand the pressures and temperatures of the engine, while also being able to accurately meter the flow of these fluids into the combustion chamber. For the preliminary stage of the design, the delta-V required for this mission is 2300 m/s.

The nozzle is designed with a de Laval-type nozzle and the combustion chamber is designed with a length of 135 mm with a 2-1 elliptical inlet curvature. The effective divergence angle is selected as 15 degrees, and the nozzle geometry is designed, where the nozzle material is selected as graphite epoxy. The combustion chamber was designed and the material is selected as Haynes 188 for strength and high-temperature operation. The tank pressures are selected with

225 psi for the fuel and oxidizer tanks, and 4500 psi for the pressurizer gas tanks, as inspired by SE-10. And the shapes of the tanks were designed and optimized for stress handling and safe operation.

In the trajectory analysis part, I conducted two simulations, to control the rocket's trajectory and ensure a safe landing on the lunar surface taking the mission constraints into account. The governing equations were derived via simplifications and assumptions and neglecting drag and lift forces. The flight strategy was composed of four stages: entry, cruise, braking, and landing. The analysis is done using MATLAB via time marching and boundary value solution, which allows simulation of the rocket's motion and control of its trajectory in a virtual environment.

References & Appendix

1. NASA CEA Web Application
2. William R. Hammock, Jr., Eldon C. Currie, Arlie E. Fisher, NASA. "Lunar Descent Vehicle Design." NASA, 1973
3. For Material Properties: https://www.rsc.org/, https://pubchem.ncbi.nlm.nih.gov/
4. Arif Karabeyoglu - MECH 427 Lecture Notes

Final CEA Simulation Results

NASA-GLENN CHEMICAL EQUILIBRIUM PROGRAM CEA2, FEBRUARY 5, 2004
BY BONNIE MCBRIDE AND SANFORD GORDON
REFS: NASA RP-1311, PART I, 1994 AND NASA RP-1311, PART II, 1996

```
*******************************************************************************

### CEA analysis performed on Wed 04-Jan-2023 13:12:52

# Problem Type: "Rocket" (Infinite Area Combustor)

prob case=hakancanp____5142 ro equilibrium

# Pressure (1 value):
p,psia= 110
# Supersonic Area Ratio (1 value):
supar= 50

# Oxidizer/Fuel Wt. ratio (1 value):
o/f= 1.8

# You selected the following fuels and oxidizers:
reac
fuel C2H8N2(L),UDMH    wt%= 50.0000
fuel N2H4(L)           wt%= 50.0000
oxid N2O4(L)           wt%=100.0000

# You selected these options for output:
# short version of output
output short
# Proportions of any products will be expressed as Mass Fractions.
output massf
# Heat will be expressed as siunits
output siunits
# Plot parameters:
output plot ae ivac
```

Input prepared by this script:/var/www/sites/cearun.grc.nasa.gov/cgi-bin/CEARU
N/prepareInputFile.cgi

IMPORTANT: The following line is the end of your CEA input file!
end

THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM

COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR

Pin =   110.0 PSIA
CASE = hakancanp____

REACTANT		WT FRACTION (SEE NOTE)	ENERGY KJ/KG-MOL	TEMP K
FUEL	C2H8N2(L), UDMH	0.5000000	48900.000	298.150
FUEL	N2H4(L)	0.5000000	0.000	0.000
OXIDANT	N2O4(L)	1.0000000	-17549.000	298.150

O/F=   1.80000 %FUEL= 35.714286 R,EQ.RATIO= 1.249350 PHI,EQ.RATIO= 1.249350

	CHAMBER	THROAT	EXIT
Pinf/P	1.0000	1.7339	675.07
P, BAR	7.5842	4.3741	0.01123
T, K	3068.05	2904.51	1079.26
RHO, KG/CU M	6.4091-1	3.9496-1	2.8132-3
H, KJ/KG	22.687	-607.53	-4945.85
U, KJ/KG	-1160.67	-1715.00	-5345.21
G, KJ/KG	-37928.4	-36535.7	-18296.1
S, KJ/(KG)(K)	12.3698	12.3698	12.3698

M, (1/n)	21.557	21.806	22.470
(dLV/dLP)t	-1.02159	-1.01621	-1.00000
(dLV/dLT)p	1.4447	1.3536	1.0000
Cp, KJ/(KG)(K)	5.6909	5.0786	1.8222
GAMMAs	1.1362	1.1381	1.2548
SON VEL,M/SEC	1159.5	1122.7	707.9
MACH NUMBER	0.000	1.000	4.453

PERFORMANCE PARAMETERS

Ae/At	1.0000	50.000
CSTAR, M/SEC	1710.4	1710.4
CF	0.6564	1.8430
Ivac, M/SEC	2109.1	3279.0
Isp, M/SEC	1122.7	3152.3

MASS FRACTIONS

*CO	0.10913	0.10390	0.03868
*CO2	0.09006	0.09829	0.20077
*H	0.00101	0.00078	0.00000
HO2	0.00002	0.00001	0.00000
*H2	0.00885	0.00835	0.01126
H2O	0.31318	0.32332	0.31423
*NO	0.00821	0.00553	0.00000
*N2	0.43122	0.43248	0.43506
*O	0.00308	0.00188	0.00000
*OH	0.02559	0.01897	0.00000
*O2	0.00962	0.00650	0.00000

* THERMODYNAMIC PROPERTIES FITTED TO 20000.K

NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS

MATLAB Simulation of Trajectory Z

Main Code

```
close all; clear, clc
% Moon constants
g0= 9.807; % Earth gravity (m/s^2)
mu= 4.905e12; % mu value of the Moon (m^3/s^2)
rho= 0; % atmospheric density of the Moon (kg/m^3)
r0= 1737e3; % radius of moon (m)
mpl= 200+200; % mass of the payload + structure (kg)
mpr= 600; % mass of the propellant (kg)
mr= mpr*.1+mpl; % mass of the reserve (kg)
Isp= 305; % specific impulse of the propulsion system (s)
mdot_max= 45e3/(Isp*g0); % maximum mass flow rate (kg/s)
% Setting up the simulation time domain
t_end= 60*30; % end time (s)
dt= 1e-2; % time step (s)
t= 0:dt:t_end; % time domain
% Initialize the state variables
x= zeros(size(t)); % horizontal position (m)
y= zeros(size(t)); % vertical position (m)
V= zeros(size(t)); %  velocity (m/s)
gamma= zeros(size(t)); %  trajectory angle (rad)
alpha= 180*ones(size(t)); %  TVC angle (rad)
r= zeros(size(t)); % radial position (m)
mdot= zeros(size(t)); % mass flow rate (kg/s)
a= zeros(size(t)); % acceleration (m/s^2)
m= zeros(size(t)); % mass of the vehicle (kg)
T= zeros(size(t)); % thrust of the propulsion system (N)
dGdt= zeros(size(t)); % derivative of trajectory angle (rad/s)
g= zeros(size(t));
% Set the initial conditions
y(1)= 500e3; % initial altitude (m)
r(1)= (y(1)+r0); % initial radius (m)
V(1)= sqrt(mu/r(1)); % circular orbit trajectory velocity (m/s)
m(1)= mpl+mpr; % initial mass (kg)
stage= "E"; % Start with entry stage
i_stage(1)= 1; % stage index
fuelFlag= 1; % boolean for fuel depletion
% Run the simulation
for i= 1:length(t)-1
   r(i)= (r0+y(i)); % instantanerous radius
   g(i)= mu/r(i)^2; % instantanerous gravity
   switch stage
       case "E"
           mdot(i)= 0.4*mdot_max;
           alpha(i)= 180;
```

```
        if V(i) < V(1)*.5
            a(1)= a(2);
            i_stage(end+1)= i;
            stage= "C";
            fprintf("Entry Phase Sucessful at t= %.2f s \n", t(i))
        end
    case "C"
        if y(i) < 50e3
            i_stage(end+1)= i;
            V_ref= V(i);
            stage= "B";
            fprintf("Cruise Phase Sucessful at t= %.2f s \n", t(i))
        end
    case "B"
        mdot(i)= .4*mdot_max;
        alpha(i)= 180+4;
        if y(i) < 30e3
            i_stage(end+1)= i;
            y_ref= y(i);
            V_ref= V(i);
            gamma_ref= gamma(i);
            stage= "L";
            fprintf("Brake Phase Sucessful at t= %.2f s \n", t(i))
        end
    case "L"
        Vpower= 2.0834; % Found by
        mdot(i)= (V(i)/V_ref)^Vpower*(0.4*mdot_max);
        mdot(i)= min(0.6*mdot_max, mdot(i)); % Min throttle
        mdot(i)= max(0.1*mdot_max, mdot(i)); % Max throttle
        alpha(i)= 180;
        if y(i) < 0
            stage= "TD";
            fprintf("Landing Sucessful at t= %.2f s \n", t(i))
        end
    case "TD"
        fprintf("TOUCH DOWN at t= %.2f s \n", t(i))
        i_stage(end+1)= i-1;
        t= t(1:i);
        mdot= mdot(1:i);
        x= x(1:i);
        y= y(1:i);
        r= r(1:i);
        V= V(1:i);
        a= a(1:i);
        m= m(1:i);
        T= T(1:i);
        gamma= gamma(1:i);
        dGdt= dGdt(1:i);
        alpha= alpha(1:i);
```

```
            break;
        case "NF" % No fuel stage
            mdot(i)= 0;
            alpha(i)= 180;
            if y(i) < 0
                stage= "TD";
            end
    end
    T(i)= mdot(i)*g0*Isp;
    % Time Marching
    a(i+1)= T(i)*cosd(alpha(i))/m(i) - g(i)*sind(gamma(i));
    V(i+1)= V(i) + a(i)*dt;
    dGdt(i+1)= V(i)*cosd(gamma(i))/r(i) + T(i)*sind(alpha(i))/(V(i)*m(i)) -
g(i)*cosd(gamma(i))/V(i);
    gamma(i+1)= gamma(i) + 180*dGdt(i)*dt; % radians to deegree
    x(i+1)= x(i) + V(i)*cosd(gamma(i))*dt; % x position of the rocket
    y(i+1)= y(i) + V(i)*sind(gamma(i))*dt; % y position of the rocket
    m(i+1)= m(i) - mdot(i)*dt; % Update the mass of the vehicle
    % Mass control
    if m(i+1)-mr < 0 && fuelFlag== 1 % Check if the propellant mass is valid
        stage= "NF"; % Set the stage to No Fuel
        fprintf("No fuel at t= %.2f s \n", t(i));
        fuelFlag= 0;
    end
end
deltaV= Isp*g0*log(m(1)/m(end));
fprintf("\n ********************* Touchdown State ********************* \n")
fprintf("Velocity: V= %.2f m/s, Trajectory angle: gamma= %.0f° \n", V(end),
gamma(end))
fprintf("Landing mass: m= %.1f kg, Reserve fuel rate: %.2f \n", m(end),
(m(end)-mpl)/mpr)
fprintf("Mission delta-V: %.2f m/s \n", deltaV)
plotMission(t, i_stage, x, y, V, a, T, m, mr, alpha, gamma, dGdt, 1)
plotEnergy(i_stage, m, mdot, V, Isp, g, gamma, dt)
save("TrajectoryZ.mat", "t", "x", "y", "i_stage")
```

plotMission

```
function plotMission(t, i_stage, x, y, V, a, T, m, mr, alpha, gamma, dGdt, iS)
a_max= -9.807*3; % 3*g constaint
Tmax= 45e3; % Maximum thrust
Ei= i_stage(iS):i_stage(end); % index range of the desired stages
tE= t(Ei); % time domain of the desired stages
figure, plot(x(Ei)*1e-3, y(Ei)*1e-3, LineWidth= 2), title("Trajectory")
hold on, plot(x(i_stage)*1e-3, y(i_stage)*1e-3, "kv")
xlabel("x (km)"), ylabel("y (km)"), grid, axis([0 500 -25 500])
yline(0, "LineWidth", 2)
figure, plot(tE, y(Ei)*1e-3, LineWidth= 2), title("Altitude")
```

```
xlabel("time (s)"), ylabel("y (m)"), grid
xline(t(i_stage(iS:end)), "LineStyle", "--")
figure, plot(tE, V(Ei).*sind(gamma(Ei)), "b-", tE, V(Ei).*cosd(gamma(Ei)),
"r-", tE, V(Ei), "k:", LineWidth= 2)
xlabel("time (s)"), ylabel("V (m/s)"), title("Velocity"), grid
legend("Vy", "Vx", "V"), xline(t(i_stage(iS:end)), "LineStyle", ":",
"HandleVisibility", "off")
figure, plot(tE, a(Ei), LineWidth= 2), title("Acceleration")
xlabel("time (s)"), ylabel("a (m/s^2)"), grid
xline(t(i_stage(iS:end)), "LineStyle", "--")
yline(a_max, "LineStyle", "-.", "LineWidth", 2)
figure, plot(tE, T(Ei)*1e-3, LineWidth= 2), title("Thrust")
xlabel("time (s)"), ylabel("T (kN)"), grid
yline(Tmax*[1 .6 .1]*1e-3, "LineStyle", "-.")
xline(t(i_stage(iS:end)), "LineStyle", "--")
figure, hold on, plot(tE, m(Ei), LineWidth= 2), title("Mass")
xlabel("time (s)"), ylabel("mass (kg)"), grid
xline(t(i_stage(iS:end)), "LineStyle", "--")
yline(mr, "LineStyle", "-.", "LineWidth", 2)
figure, plot(tE, alpha(Ei), LineWidth= 2), title("TVC angle")
xlabel("time (s)"), ylabel("alpha (degrees)"), grid
xline(t(i_stage(iS:end)), "LineStyle", "--")
figure, plot(tE, gamma(Ei), LineWidth= 2), title("Trajectory Angle")
xlabel("time (s)"), ylabel("gamma (degrees)"), grid
xline(t(i_stage(iS:end)), "LineStyle", "--")
figure, plot(tE, dGdt(Ei), LineWidth= 2), title("dGdt")
xlabel("time (s)"), ylabel("dGdt (degree/s)"), grid
xline(t(i_stage(iS:end)), "LineStyle", "--")
```

plotEnergy

```
function plotEnergy(ix, m, mdot, V, Isp, g, gamma, dt)
g0= 9.807;
Mc= [m(ix(1))-m(ix(2));
   m(ix(2))-m(ix(3));
   m(ix(3))-m(ix(4));
   m(ix(4))-m(ix(5))];
dVc= Isp*g0*[log(m(ix(1))/m(ix(2)));
           log(m(ix(2))/m(ix(3)));
           log(m(ix(3))/m(ix(4)));
           log(m(ix(4))/m(ix(5)))];
gLc= dt*Isp*g0*
[sum(log(m(ix(1):ix(2))/m(ix(2))).*(g(ix(1):ix(2)).*sind(gamma(ix(1):ix(2)))));

sum(log(m(ix(2):ix(3))/m(ix(3))).*(g(ix(2):ix(3)).*sind(gamma(ix(2):ix(3)))));

sum(log(m(ix(3):ix(4))/m(ix(4))).*(g(ix(3):ix(4)).*sind(gamma(ix(3):ix(4)))));

sum(log(m(ix(4):ix(5))/m(ix(5))).*(g(ix(4):ix(5)).*sind(gamma(ix(4):ix(5)))))];
```

```
gLc= sqrt(-2*gLc); % G= .5*V^2
X= categorical(["Entry", "Cruise", "Brake", "Landing"]);
X = reordercats(X,string(X));
Y= [Mc, dVc, gLc];
figure
bar(X, Y), title("Energy Requirements for Lunar Landing")
grid, legend(["Consumed Propellant", "ΔV_{del}", "Gravity Loss"])
ylabel("Propellant Mass (kg)     |    ΔV (m/s)     |    Gravity Loss (m/s)")
```

Trajectory Animation

```
close all; clear, clc
r0= 1737; % radius of moon (km)
load TrajectoryY.mat % load the trajectory
aRate= 750; % animation plot rate (frame/index)
aPause= 1e-3; % animation pause (s)
% Convert positions to polar
x= x*1e-3;
z= y*1e-3+r0; % radial position
y= zeros(1, length(t));
% Plot the trajectory and the Moon
hold on, title("Trajectory of Lunar Landing")
[Xm, Ym, Zm]= sphere(100); % moon surface
Xm= Xm*r0; Ym= Ym*r0; Zm= Zm*r0;
surf(Xm, Ym, Zm, lineStyle= "none", FaceColor= "flat")
zlim([0, 2250])
colormap gray
axis equal, view([10 5])
for i= 1:length(t)
   if mod(i, aRate)== 1
       plot3(x(i), y(i), z(i), "k.", MarkerSize= 1)
       pause(aPause)
   end
end
```