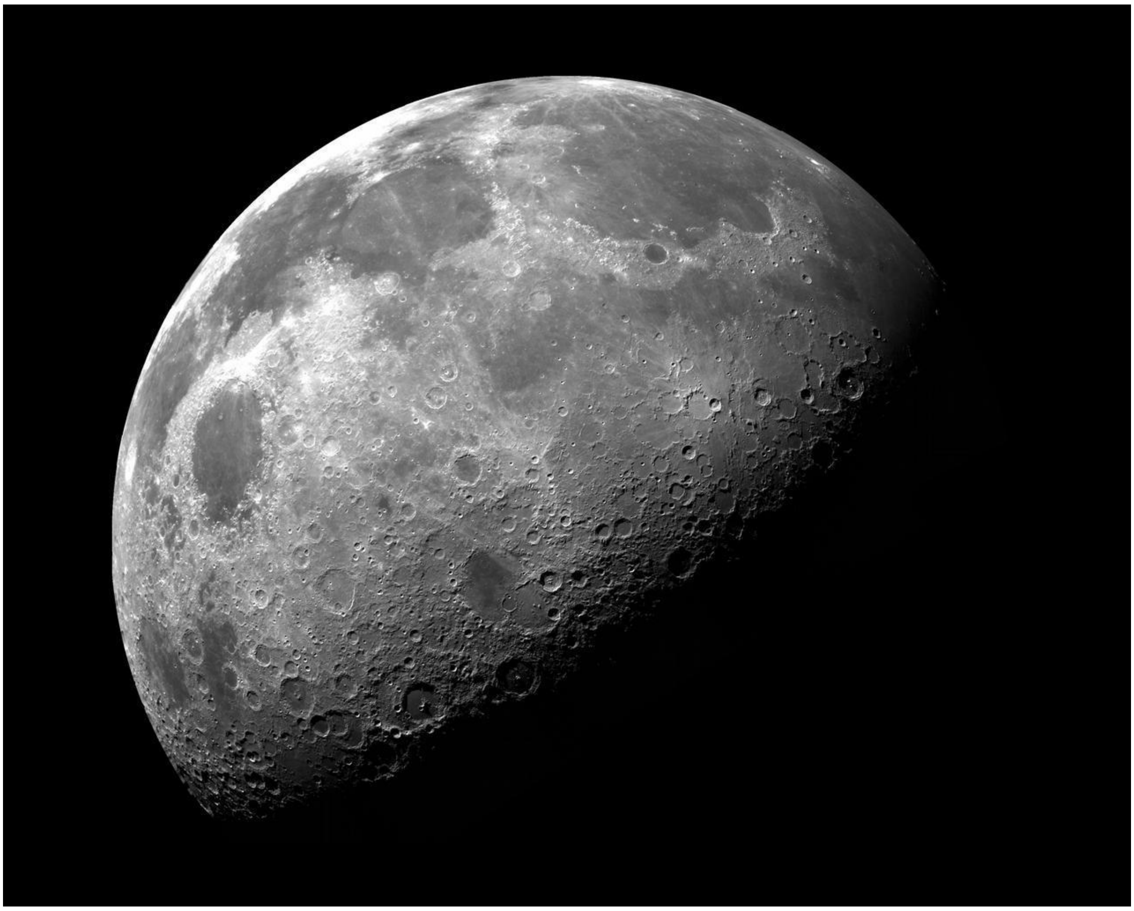

Lunar Lander Conceptual Design Report

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Abstract

This report summarizes an independent study conducted as part of a lunar lander design course. The study aims to design a lunar lander capable of delivering a 1500kg payload to the lunar surface. The study demonstrates the feasibility of constructing a lander that meets the specified payload requirement.

Contents

1 Introduction

The primary objective of this study is to design a lunar lander capable of delivering a 1500kg payload to the lunar surface. The course provided foundational knowledge of the principles and practices of lunar lander design, including the selection of a rocket, propellant and array type, engine, and tank design.

Throughout this study, we evaluated various design choices to optimize the lander's performance while adhering to the constraints and challenges inherent in lunar missions. The following sections discuss the implementation of key design components.

2 Background

2.1 Konstantin Tsiolkovsky: The Rocket Equation

Tsiolkovsky's "Rocket Equation" provides the mathematical foundation for understanding how rockets achieve the necessary velocity to reach space and travel to celestial bodies, including the Moon. It relates change in velocity of the rocket (ΔV), efficiency of the rocket engine (specific impulse, I_{sp}), initial mass of the rocket (m_0), and final mass of the rocket (m_f). We will use this equation for the delta-V budget.

$$\Delta V = g I_{sp} \ln\left(\frac{m_0}{m_f}\right) \tag{1}$$

2.2 Introduction to Orbital Mechanics: The Vis-Viva Equation

The "Vis-Viva" equation provides us with the relationship between the orbital velocity (v), the distance from the center of the celestial body (r), and the semimajor axis of the orbit (a), along with the gravitational constant (μ). This is vital for our understanding how the spacecraft will travel from Earth to the Moon and how it will enter and leave lunar orbit.

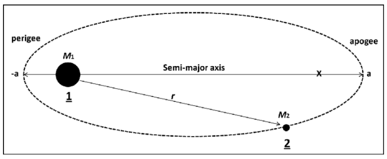

Figure 2.1: Vis-Viva Diagram

$$v = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)} \tag{2}$$

2.3 Tank Sizing Equation

The volume equation for a fuel tank helps us to understand the relationship between the total volume of the tank (V_{Tank}), the radius of the cylindrical body and the hemispherical ends (r), and the length of the cylindrical body (L). Using this equation, we will determine the storage capacity of the fuel tanks.

$$V_{\text{Tank}} = \pi r^2 L + \frac{4}{3}\pi r^3 \tag{3}$$

3 Implementation

We conducted four trades, each utilizing rockets from the Scout, Vanguard, Juno, and Nike families. Figures 3.1-3.4 show the efficiency of various rockets as a function of start mass (kg) and payload (kg). We experimented with four different types of propellants to validate our findings. Based on these experiments, we conclude that the Nike rocket family is the optimal choice to support a 1500kg payload.

In addition, in the realm of aerospace engineering, Nike rockets have consistently demonstrated exemplary reliability in numerous historical missions. Some notable examples underscore their mission success: Nike Ajax Missile System, Nike Hercules Missile System, NASA's Nike Apache Rocket, Nike Orion, and Nike Tomahawk.

3.1 Rocket Choice

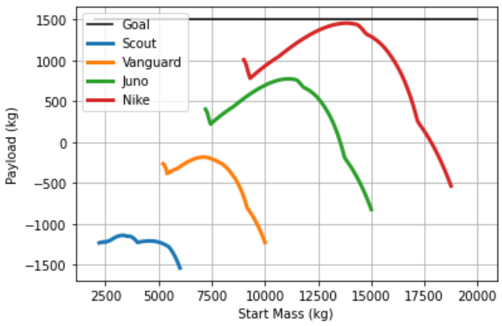

Figure 3.1: LOX/LH2 Trade

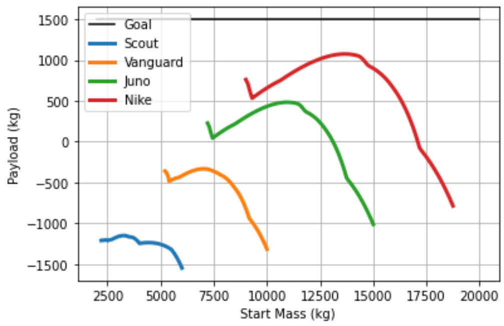

Figure 3.2: LOX/Methane Trade

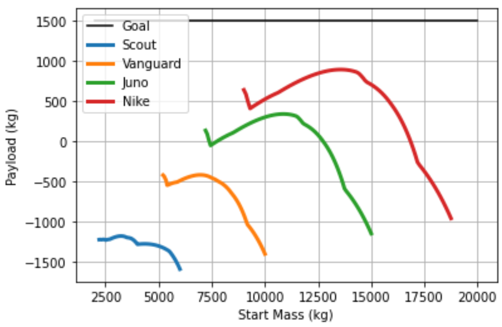

Figure 3.3: LOX/RP-1 Trade

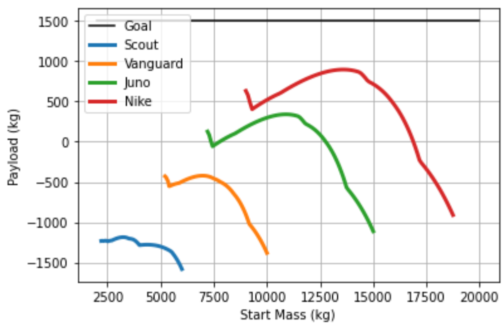

Figure 3.4: NTO/MMH Trade

3.2 Propellant Choice

Figure 3.5. shows the efficiency of various propellants as a function of start mass (kg) and payload (kg). The only propellant which supports the 1500kg payload goal is LOX/LH2.

Figure 3.6 illustrates the cost analysis of the propellants depicted in Figure 3.5 varying with start mass (kg) and cost in USD per unit mass (kg). Initially, LOX/LH2 appears more expensive at lower start masses. However, considering that it is the only fuel type that meets our payload requirements, we should proceed with it as our preferred propellant choice.

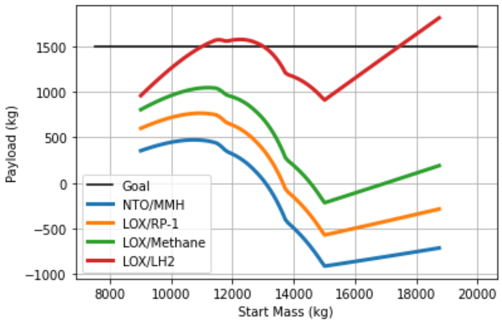

Figure 3.5: Prop. Efficiency Trade

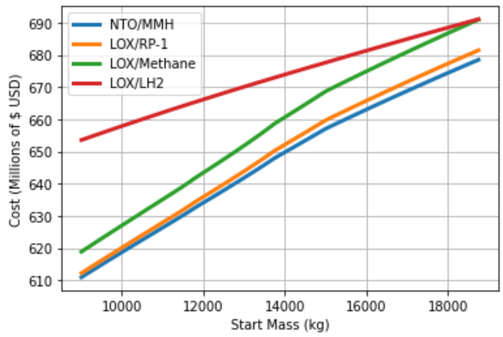

Figure 3.6: Propellant Cost Trade

3.3 Array Type

Figures 3.7 and 3.8 show the possible payloads utilizing a deployable vs. body-mounted solar array using LOX/LH2 propellant. Our analysis clearly indicates that the deployable array offers a significant advantage in payload capacity. The graph in Figure 3.7 shows that the deployment of the array results in a significantly higher payload capacity compared to a body-mounted solar array configuration. Therefore, we utilize a deployable array configuration for all further trade studies.

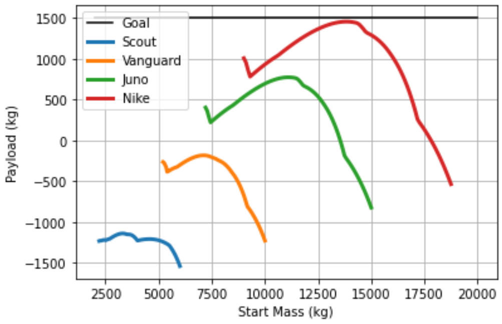

Figure 3.7: Deployable Array

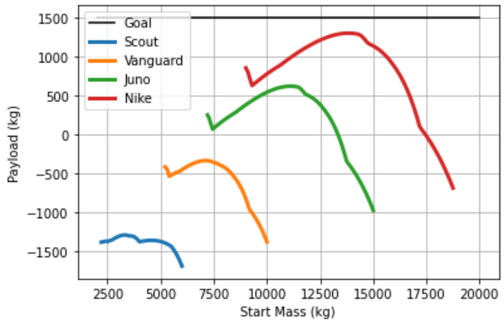

Figure 3.8: Body-Mounted Array

3.4 Engine

3.4.1 Engine Choice

Within the lunar lander course, we received four predesigned engines. Figure 3.9 shows their respective payload capacities for various start masses. As observed, none of the pre-designed engines meet our payload requirement. As a result, we developed our own engine.

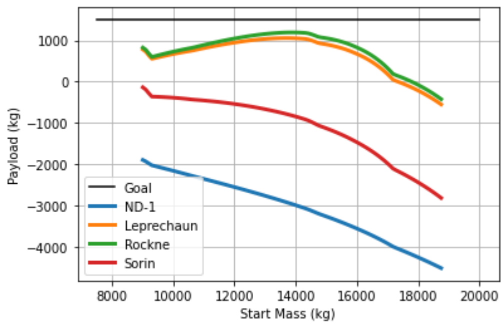

Figure 3.9: Engine Payload Trade

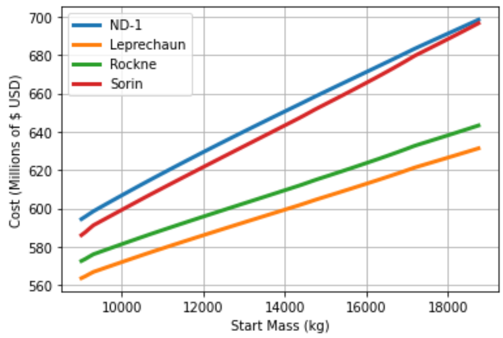

Figure 3.10: Engine Cost Trade

In Figure 3.11, we observe several thrust capacities capable of supporting a 1500 kg payload. Figure 3.11 shows that a 15 kN thrust engine successfully supports the payload target. Given this, we will proceed with the 15 kN rocket to ensure we meet our mission requirements.

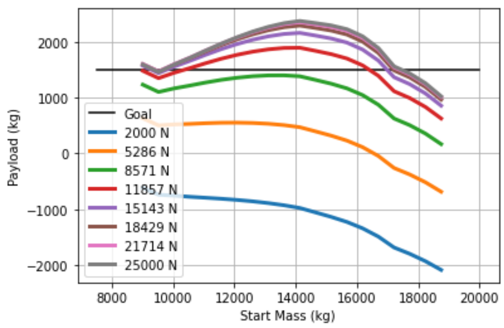

Figure 3.11: Engine Payload Trade

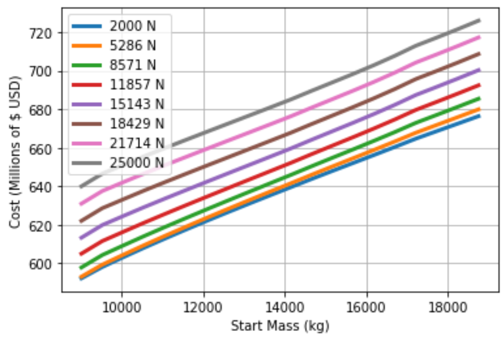

Figure 3.12: Engine Cost Trade

3.4.2 Pump vs. Pressure Fed

Figures 3.13 and 3.14 show the engine parameters previously discussed in 3.4.1, now investigating our choice of propellant feed system. Specifically, in Figure 3.13 we depict the engine as pump-fed while in Figure 3.14 we show the engine as pressure-fed. Based on these comparisons, we conclude that the pump-fed configuration will allow us to achieve the highest payload capacity. Therefore, we proceed with the pump-fed configuration for our design.

Our engine type closely mimics the LE-5A engine used on the H-IIA rocket, developed by Mitsubishi Heavy Industries and utilized by the Japan Aerospace Exploration Agency (JAXA). The LE-5A engine operates with LH2 and LOX, providing a thrust of approximately 27.6 kN. We consider the LE-5A engine reliable and efficient within its intended application for the H-IIA rocket series. JAXA has successfully used it in multiple launches, contributing to the overall success of the H-IIA missions and further proving its efficacy for our purposes. We will refer to our new engine as the LE-5B.

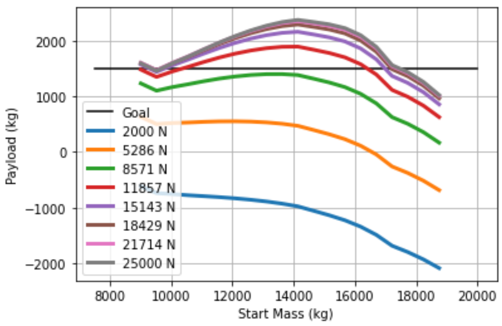

Figure 3.13: Pump-Fed Trade

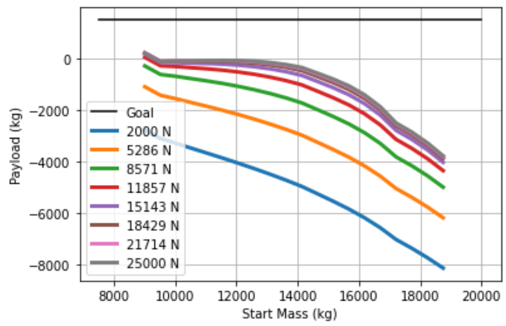

Figure 3.14: Pressure-Fed Trade

3.5 Tank Specifications

3.5.1 Number of Tanks

In Figure 3.15, we observe multiple options for our use of one or multiple tanks per propellant. We see in Figure 3.15 that the use of one tank per propellant is optimal to maximize the payload capacity of our lander. The graph clearly illustrates that with a single tank configuration, we achieve the highest payload capacity compared to configurations with two or three tanks. Furthermore, Figure 3.16 shows that the use of one tank also results in the lowest cost compared to configurations with multiple tanks.

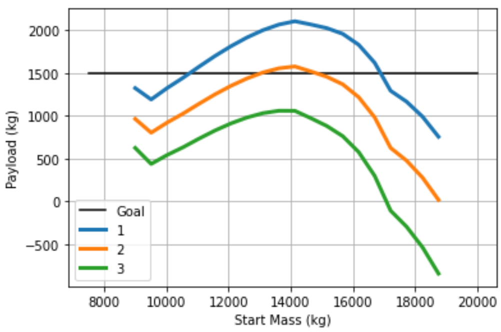

Figure 3.15: Tank Number Trade

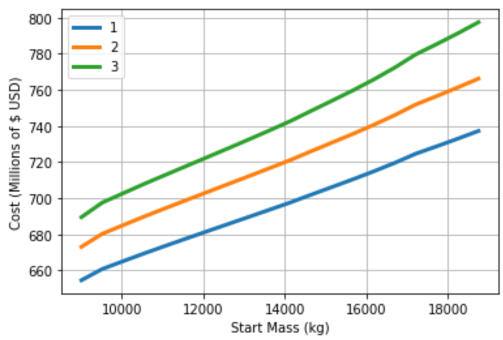

Figure 3.16: Tank Cost Trade

3.5.2 Tank Diameter

We determined the diameter of the tanks by accounting for several clearances and spacings: 0.2 meters for dynamic clearance, 0.024 meters for insulation, 0.15 meters for tank spacing, and 0.3 meters for landing gear and solar panels. Eq. 4 shows the maximum radius of the tanks, r_{Max}.

$$r_{\mathrm{Max}} = \frac{7\text{ m} - 0.2\text{ m} - 0.024\text{ m} - 0.15\text{ m} - 0.3\text{ m}}{2} = 3.16\,\mathrm{m} \tag{4}$$

Using our propellant totals, dividing by their respective densities, and adding a 10% ullage, we obtain our volume requirements per tank.

Using this, we calculate the optimal tank structure by comparing the maximum volume capacities of sphere vs. pill-shaped tanks. Eq. 5 provides the volume of a sphere where V_{Prop} is the volume of propellant and r_{Max} is the maximum radius.

$$V_{\mathrm{Prop}} = \frac{4}{3}\pi r_{\mathrm{Max}}^3 \tag{5}$$

This yields a volume of 132 kg/m^3, greater than the necessary volume capacity of our oxygen (4.62 kg/m^3) and LH2 (14.87 kg/m^3) tanks. Therefore, we do not require any additional tank space outside of that given by a sphere of radius r_{Max}. As a result, we utilize a spherical (i.e. without a cylindrical body) tank configuration.

Since a sphere of radius r_{Max} is too large for our propellant needs, we calculate only the radius needed. Eq. 6 provides the radius where V_{Prop} is volume of propellant and r_{Optimal} is the radius.

$$r_{\mathrm{Optimal}} = V_{\mathrm{Prop}}\frac{4}{9}\pi \tag{6}$$

Inputting the required propellant volumes, we find the oxygen tank radius of 1.03 m and the LH2 tank radius of 1.53 m. Since we know our tanks will be spherical, we multiply the maximum radius by 2 to produce our tank lengths of 2.07 m and 3.05 m. Repeating this process once more for our monoprop tank results in a radius of .32 m and length of .65 m.

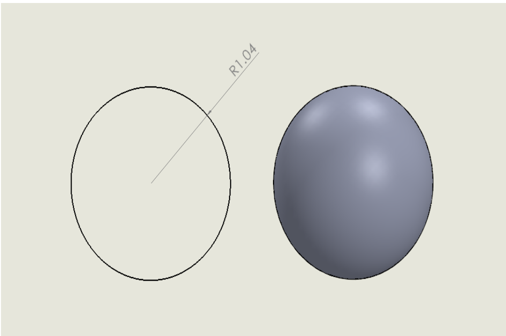

Figure 3.17: Oxygen Tank

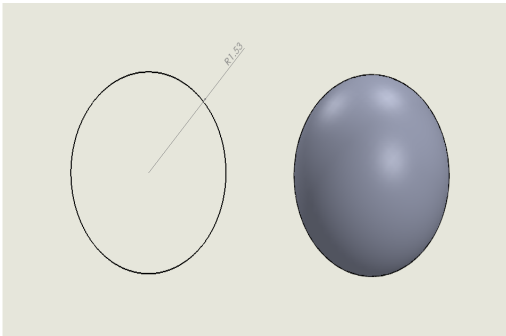

Figure 3.18: Fuel Tank

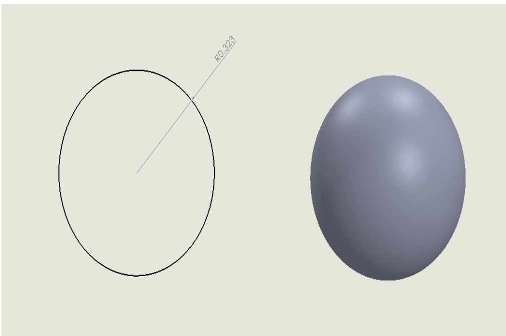

Figure 3.19: Monoprop Tank

3.5.3 Tank Material

To select the primary material for the tank construction of our spacecraft, we performed a detailed analysis using Figures 3.20 and 3.21. Figure 3.20 graphically illustrates that Al-Li offers the highest payload capacity among the materials considered, including Stainless Steel and Al2219, when plotted against the starting mass of the lander. Furthermore, Figure 3.21 confirms that Al-Li is not only optimal for payload but also the most cost-effective option among the materials studied. The favorable combination of high strength-to-weight ratio and affordability makes Al-Li the clear choice to ensure that our spacecraft meets performance targets while maintaining fiscal responsibility.

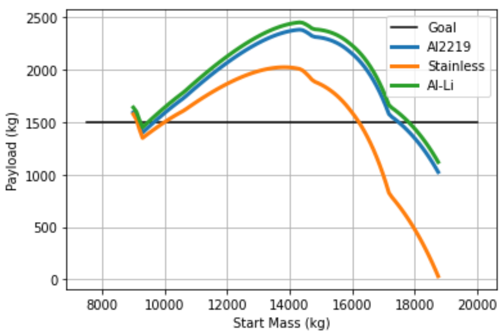

Figure 3.20: Tank Material Trade

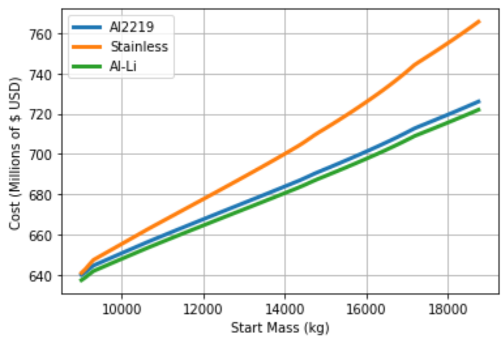

Figure 3.21: Tank Material Trade

3.6 Validation

Through the trade studies conducted in sections 3.2, 3.2, 3.3, 3.4, and 3.5, we arrived on a final design. The chosen configuration utilizes a Nike Rocket, powered by LOX and LH2 propellants, a deployable solar panel array, Al-Li alloy with a single tank design, and our custom pump-fed LE-5B engine.

We reanalyze Figure 3.1 using the newly found inputs outlined in the above paragraph. This updated graph confirms that the Nike Rocket Family was the only viable choice. Additionally, it demonstrates that our trade studies have successfully optimized the payload capacity.

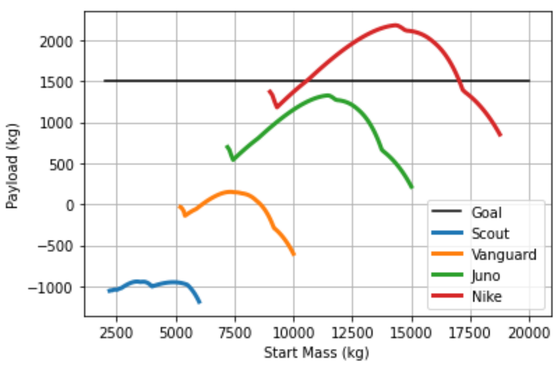

Figure 3.22: Optimized Rocket Trade

3.7 Mass Budget

Given Figure 3.22 and the trades we conducted, the lowest possible start mass that grants us at least a 1500 kg payload is 11000 kg. We assume this mass for our mass budget computations. For all of the following mass subsections, we utilize mass estimating relationships.

3.7.1 Avionics

To derive Eq. 7 we used the best-fit data from other rockets of similar class and their respective avionics system mass. Using this function, we estimated our avionics system mass as a function of the starting mass.

$$M_{\text{avionics}} = 8(M_0)^{0.361} = 230.2 \text{ kg} \tag{7}$$

3.7.2 Electrical

Using a 1200 W nominal power draw for our lander and a 1000 W power draw for our payload, we calculate the mass of all electrical components.

We start by calculating the total load accounting for a margin of 30%.

$$\text{Total margined load} = (1200 \text{ W} + 1000 \text{ W}) \cdot 1.3 = 2860\,\text{W} \tag{8}$$

Next, we determine the array mass using a deployable array's power density of 75 W/kg.

$$M_{\text{Array}} = \frac{2860 \text{ W}}{75 \text{ W/kg}} = 38\,\text{kg} \tag{9}$$

We then calculate the total energy needed for 8 hours of battery usage.

$$E_{\text{Tot}} = 2860 \text{ W} \cdot 8 \text{ hr} = 22880\,\text{W-hr} \tag{10}$$

Following this, we compute the battery energy needed assuming a depth of discharge of 30%.

$$E_{\text{Battery}} = \frac{22880 \text{ W-hr}}{0.7} = 32685\,\text{W-hr} \tag{11}$$

Subsequently, we find the mass of the battery using a 100 W-h/kg battery.

$$M_{\text{Battery}} = \frac{32685 \text{ W-hr}}{100 \text{ W-hr/kg}} = 326.9\,\text{kg} \tag{12}$$

To derive Eq. 13 we used the best-fit data from other rockets of similar class and their respective wiring system mass. Using this function, we estimate our wiring system mass as a function of the starting mass (M_0) and max tank length (h_{tank}).

$$M_{\text{wiring}} = 1.058\sqrt{(M_0)}h_{\text{tank}}^{\frac{1}{4}} = 146.66 \text{ kg} \tag{13}$$

Finally, we calculate the total electrical system mass, including the wiring and a 50kg converter:

$$M_{\text{Electrical}} = M_{\text{Array}} + M_{\text{Battery}} + M_{\text{Converter}} + M_{\text{Wiring}} = 561.6\,\text{kg} \tag{14}$$

3.7.3 Propulsion

Eq. 15 provides the mass of the engine where T is the thrust of our LE-5B engine (14000 N) and TW_{Engine} is our thrust to weight ratio (40).

$$M_{\text{Engine}} = \frac{1}{\left(\frac{TW_{\text{Engine}}}{\text{T}}\right) \cdot 9.81 \text{ m/s}^2} = 38.22\,\text{kg} \tag{15}$$

The following Eq. 16 provides the surface area of the tanks, where r is the radius of each tank.

$$\text{saDomesPerTank} = 4\pi \cdot r^2 \tag{16}$$

After we enter our values from section 3.5.2, we obtain the following results.

$$\text{saOxTank} = 13.41\,\text{m}^2 \quad (17) \qquad \text{saFuelTank} = 29.25\,\text{m}^2 \tag{18}$$

$$\text{saMonoTank} = 1.31\,\text{m}^2 \tag{19}$$

Given our use of cryogenic propellant, cooling is essential. We employ spray-on foam (0.005 m thickness, 50 kg/m^3 density) and multilayer insulation (0.001 m thickness, 80 kg/m^3 density). Using these values, we calculate the mass of our spray-on and multilayer insulation.

$$m_{\text{SOFIOx}} = 13.41 \text{ m}^2 \cdot .005 \text{ m} \cdot 50 \text{ kg/m}^3 = 3.35 \text{ kg} \tag{20}$$

$$m_{\text{SOFILH2}} = 29.25 \text{ m}^2 \cdot .005 \text{ m} \cdot 50 \text{ kg/m}^3 = 7.31 \text{ kg} \tag{21}$$

$$m_{\text{LMIOx}} = 13.41 \text{ m}^2 \cdot .001 \text{ m} \cdot 80 \text{ kg/m}^3 = 1.07 \text{ kg} \tag{22}$$

$$m_{\text{MLILH2}} = 29.25 \text{ m}^2 \cdot .001 \text{ m} \cdot 80 \text{ kg/m}^3 = 2.33 \text{ kg} \tag{23}$$

We find the necessary thickness for our tanks using Eqs. 24 and 25 where P_{Tank} is pressure, r is the maximum radius, σ is maximum stress, t is thickness, ρ_{Prop} is propellant density, a_{Max} is maximum launch acceleration, and l_{Tank} is the length of the tank.

$$\sigma = \frac{Pr}{2t} \tag{24}$$

$$P = 1.5\,(P_{\text{Tank}} + \rho_{\text{Prop}} a_{\text{Max}} l_{\text{Tank}}) \tag{25}$$

Solving for P in all tanks accounting for a 1.5 FOS yields the following.

$$P_{\text{Ox}} = 1.5\left(300 \text{ kPa} + 1140 \text{ kg/m}^3 \cdot 50 \text{ m/s}^2 \cdot 2.07 \text{ m}\right) = 626681.6 \text{ kPa} \tag{26}$$

$$P_{\text{LH2}} = 1.5\left(300 \text{ kPa} + 70 \text{ kg/m}^3 \cdot 50 \text{ m/s}^2 \cdot 3.05 \text{ m}\right) = 466019.3 \text{ kPa} \tag{27}$$

$$P_{\text{Mono}} = 1.5\left(300 \text{ kPa} + 70 \text{ kg/m}^3 \cdot 50 \text{ m/s}^2 \cdot .646 \text{ m}\right) = 491957.4 \text{ kPa} \tag{28}$$

By solving Eq. 24 for t in our Ox, LH2, and monoprop tanks, and using our minimum pressure totals above, we obtain our minimum thickness requirements.

$$t_{\text{Ox}} = \frac{P_{\text{Ox}} r}{2\sigma} = 4.6 \times 10^{-4} \text{ m} \tag{29}$$

$$t_{\text{LH2}} = \frac{P_{\text{LH2}} r}{2\sigma} = 5.1 \times 10^{-4} \text{ m} \tag{30}$$

$$t_{\text{Mono}} = \frac{P_{\text{LH2}} r}{2\sigma} = 2.7 \times 10^{-4} \text{ m} \tag{31}$$

Since the minimum stress-allowing thickness for each tank is less than the manufacturable thickness of Al-Li and Al2219 (.004 m), we will use a uniform thickness of 0.004 m for the oxygen, LH2 tanks, and monoprop tanks. We multiply by thickness and density to find the masses of each propellant tank. The LOX and LH2 tanks utilize Al-Li (density of 2700 kg/m^3) and the monoprop tank utilizes Al2219 (density of 2840 kg/m^3).

$$m_{\text{OxTank}} = .004 \text{ m} \cdot 13.41 \text{ m}^2 \cdot 2700 \text{ kg/m}^3 = 144.89 \text{ kg} \tag{32}$$

$$m_{\text{LH2Tank}} = .004 \text{ m} \cdot 29.25 \text{ m}^2 \cdot 2700 \text{ kg/m}^3 = 315.89 \text{ kg} \tag{33}$$

$$m_{\text{LH2Tank}} = .004 \text{ m} \cdot 1.31 \text{ m}^2 \cdot 2840 \text{ kg/m}^3 = 14.15 \text{ kg} \tag{34}$$

To account for welds, end caps, and other manufacturing processes, we add 20% to the tank masses.

$$m_{\text{OxTank}} \cdot 1.2 = 173.86 \text{ kg} \tag{35}$$

$$m_{\text{LH2Tank}} \cdot 1.2 = 379.07 \text{ kg} \tag{36}$$

$$m_{\text{MonoTank}} \cdot 1.2 = 16.98 \text{ kg} \tag{37}$$

For a 1.1 FOS, we multiply the tank masses one final time.

$$m_{\text{OxTank}} \cdot 1.1 = 191.25 \text{ kg} \tag{38}$$

$$m_{\text{LH2Tank}} \cdot 1.1 = 416.97 \text{ kg} \tag{39}$$

$$m_{\text{MonoTank}} \cdot 1.1 = 19.66 \text{ kg} \tag{40}$$

Our lander utilizes a 50 kg RCS, a 100 kg pressurization system, and 50 kg of feedlines. We sum up all of the propulsion system components to yield our total system mass.

$$\begin{aligned} m_{\text{propulsion}} = {} & m_{\text{OxTank}} + m_{\text{LH2Tank}} + m_{\text{MonoTank}} + m_{\text{SOFIOx}} \\ & + m_{\text{SOFILH2}} + m_{\text{LMIOx}} + m_{\text{MLILH2}} \\ & + m_{\text{RCS}} + m_{\text{Pressurization}} + m_{\text{Feedlines}} + m_{\text{Engine}} = 880.2 \text{ kg} \end{aligned} \tag{41}$$

3.7.4 Thermal

We used the best-fit data from other rockets of similar class and their respective thermal system mass to derive Eq. 42. Using this function, we estimate the mass of our thermal system as a function of the starting mass.

$$M_{\text{thermal}} = 0.03 M_0 = 330.0 \text{ kg} \tag{42}$$

3.7.5 Structures

We estimate the structure mass to be 20% of the dry mass and the landing gear to be 8% of the dry mass. Using these percentages, we calculate their masses to complete our mass budget.

$$m_{\text{DryWithoutStructure}} = m_{\text{Avionics}} + m_{\text{Electrical}} + m_{\text{Propulsion}} + m_{\text{Thermal}} \tag{43}$$

$$m_{\text{Structures}} = \frac{m_{\text{DryWithoutStructure}}}{(1 - .28)} \cdot (.28) = 778.6 \text{ kg} \tag{44}$$

3.8 Propellant and ΔV Budgets

We determined our propellant requirements to support our lander through each phase. To achieve this, we utilize the equations outlined in Sections 2.1 and 2.2. Using Eq. 2, we calculate the velocities at two critical points: the initial apoapsis and the target apoapsis. The difference between these velocities provides us with the necessary ΔV to reach our target apoapsis. Subsequently, we use ΔV to solve for m_f in Eq. 1 thereby determining the required propellant. This approach, based on our mission parameters, yields the following ΔV budget.

Phase Name	DV (m/s)	Mass0 (kg)	MassF (kg)	Impulse (kg)	Boiloff (kg)	Chill (kg)	RCS (kg)	Settling (kg)
Pre-TCM1 Settling	0.0	11000.0	10995.0	0.0	0.0	0.0	0.0	5.0
Pre-TCM1 Chill	0.0	10995.0	10985.0	0.0	0.0	10.0	0.0	0.0
TLI	353.4	10985.0	10174.5	810.4	0.0	0.0	0.0	0.0
Coast to TCM1	0.0	10174.5	10156.5	0.0	15.0	0.0	3.0	0.0
Pre-TCM1 Settling	0.0	10156.5	10151.5	0.0	0.0	0.0	0.0	5.0
Pre-TCM1 Chill	0.0	10151.5	10141.5	0.0	0.0	10.0	0.0	0.0
TCM1	20.0	10141.5	10097.6	43.9	0.0	0.0	0.0	0.0
Coast to TCM2	0.0	10097.6	10061.6	0.0	30.0	0.0	6.0	0.0
TCM2	5.0	10061.6	10038.3	23.3	0.0	0.0	0.0	0.0
Coast to TCM3	0.0	10038.3	10020.3	0.0	15.0	0.0	3.0	0.0
TCM3	5.0	10020.3	9997.1	23.2	0.0	0.0	0.0	0.0
Coast to LOI	0.0	9997.1	9988.1	0.0	7.5	0.0	1.5	0.0
Pre-LOI Settling	0.0	9988.1	9983.1	0.0	0.0	0.0	0.0	5.0
Pre-LOI Chill	0.0	9983.1	9973.1	0.0	0.0	10.0	0.0	0.0
LOI	850.0	9973.1	8293.9	1679.1	0.1	0.0	0.0	0.0
Coast to TCM4	0.0	8293.9	8284.9	0.0	7.5	0.0	1.5	0.0
TCM4	5.0	8284.9	8265.8	19.2	0.0	0.0	0.0	0.0
Coast to DOI	0.0	8265.8	8256.8	0.0	7.5	0.0	1.5	0.0
Pre-DOI Settling	0.0	8256.8	8251.8	0.0	0.0	0.0	0.0	5.0
Pre-DOI Chill	0.0	8251.8	8241.8	0.0	0.0	10.0	0.0	0.0
DOI	25.0	8241.8	8197.2	44.6	0.0	0.0	0.0	0.0
Coast to PDI	0.0	8197.2	8188.2	0.0	7.5	0.0	1.5	0.0
Pre-PDI Settling	0.0	8188.2	8183.2	0.0	0.0	0.0	0.0	5.0
Pre-PDI Chill	0.0	8183.2	8173.2	0.0	0.0	10.0	0.0	0.0
PDI	1978.1	8173.2	5321.7	2851.3	0.2	0.0	0.0	0.0

Table 1: Mission Phase Propellant Budget

Using Table 1 we find our propellant needs by category.

Category	Total (kg)	Oxidizer (kg)	Fuel (kg)	Mono (kg)
Impulse	5494.9	4593.9	835.3	65.6
Reserve	109.9	91.9	16.7	1.3
Boiloff	90.3	30.1	60.2	0.0
RCS	18.1	0.0	0.0	18.1
Settling	25.0	0.0	0.0	25.0
Residual	57.9	47.4	9.4	1.1
Total	5846.0	4788.4	946.6	111.1

Table 2: Propellant Budget by Category

3.9 Mass Budget

Category	Basic Mass (kg)
Avionics	230.2
Electrical	561.6
Propulsion	880.2
Thermal	330.0
Structures	778.6
Total Basic Mass	2780.6
MGA	417.1
Total Predicted (Basic + MGA)	3197.6
Margin	417.1
Total Allowable (Pred + Margin)	3614.7

Table 3: Propellant Budget by Category

Using the data found in Tables 1, 2, and 3, we determine that our final payload capacity exceeds our goal by 39.2 kg, resulting in a total capacity of 1539.2 kg.

Category	Mass (kg)
Allowable Dry Mass	3614.7
Total Propellant	5846.0
Payload	1539.2
Launch Mass	11000.0

Table 4: Mass Budget by Category

3.10 Monetary Budget

In developing our rocket, it is important to distinguish between nonrecurring engineering (NRE) and recurring engineering (RE) costs. NRE covers the initial design, development, and testing work that happens only once. RE represents the unit cost of producing each rocket. The following equations are predictive models using data from various lunar lander budgets. They provide the NRE and RE costs for our rocket, giving us an estimate of the project's financial requirements.

$$RE_{Lander} = \$30,000,000 \left(\frac{m_{\text{dry}}}{2000}\right)^{0.4} = \$38,013,526 \tag{45}$$

$$NRE_{\text{Engine}} = \$2,500 \cdot \text{thrust} = \$37,500,000 \tag{46}$$

$$NRE_{Lander} = 12RE_{Lander} + NRE_{engine} = \$456,162,317 \tag{47}$$

Category	Cost ($ USD)
Engine Estimate	$37,500,000
Lander Cost Estimate	$456,162,317
Rocket Cost	$150,000,000
Total	$643,662,317

Table 5: Project Cost

3.11 Conclusion

In conclusion, after thorough analysis and consideration of various design elements, we identified the Nike rocket as the most optimal lunar lander configuration to support a 1500 kg payload. The use of LOX/LH2 propellants offers superior efficiency and thrust, while AL-Li materials ensure the structure remains both robust and lightweight. Implementing a pump-fed engine system, along with a custom-designed LE-5B engine, enhances overall performance, reliability, and payload efficiency. Furthermore, the decision to use one tank per propellant streamlines the design and optimizes fuel management. This configuration not only meets payload requirements, but exceeds it by roughly 40 kg.

By successfully implementing this design, we could significantly enhance our capabilities for sustainable lunar exploration. These findings lay a solid foundation for developing a viable lunar lander, paving the way for our future lunar exploration and potential missions beyond.