Rapid Lunar Transporter

Group Echo

AA 462 (Rocket Propulsion) Design Report

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The Rapid Lunar Transporter is a spacecraft and mission designed to execute soft landings on the moon quickly, reliably, and safely. After launch from one of a variety of potential modern launch vehicles, the two stage vehicle detailed in this report can travel from low Earth orbit to the surface of the moon at Sinus Medii in less than forty hours. The first stage features a LOX/LH2 engine with a specific impulse of 448.64s and thrust of 100 kN that inserts the upper stage in a translunar injection trajectory at a speed of 11.47 km/s. The second stage employs a reliable hypergolic bipropellant engine with a specific impulse of 328s and a thrust of 55.4 KN that will slow the spacecraft and lower it to the surface of the moon. The first stage uses a turbo-pump fed system to achieve a high insertion velocity while the second stage utilizes the simplicity of a pressure fed system during orbit maneuvering and landing phase of the mission. The total ΔV capability of the design is 8.60 km/s.

Nomenclature

A	Area [m^2]
c	Effective Exit Velocity [m/s]
c^*	Characteristic Velocity [m/s]
F	Thrust [kN]
L^*	Characteristic Chamber Length [m]
m	Mass [kg]
$\dot{m}$	Mass Flow Rate [kg/s]
P	Pressure [kPa]
r	Mixture Ratio
t_{bo}	Burn Time [s]
T	Temperature [K]
v	Velocity[m/s]
ρ	Density [kg/m^3]
γ	Heat Capacity Ratio
ϵ	Structural Mass Ratio

Subscripts:

c	Combustion Chamber
e	Nozzle Exit
f	End of Stage
i	Injector
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
o	Beginning of Stage
p	propellant
pl	payload
t	Nozzle Throat

I. Introduction

A multi-national consortium has made a call for proposals for a spacecraft system that deliver and softly land a scientific payload from Low Earth Orbit (LEO) to the surface of the moon at Sinus Medii. The payload consists of a rover, science instruments, and communication equipment that will explore and study the natural satellite. The following design report is a response to the consortium's call.

The consortium established three primary mission requirements:

1. The spacecraft begins from a circular Low Earth Orbit altitude of 400 kilometers.
2. The landing site on the moon is Sinus Medii, at the confluence of the lunar equator and prime meridian.
3. The total payload mass is 800 kilograms.

Furthermore, the design proposal was permitted to adhere to the following general simplifications and assumptions:

1. Ideal rocket theory and ideal gas theory can be used.
2. The burns used for orbital maneuvering may be treated as non-impulsive and the method of patched conics may be implemented .
3. Laminar flow may be assumed in the propellant feed lines.

Beyond these constraints and assumptions the only other mission design requirement is that characteristics of the spacecraft follow fundamental physical and chemical laws using existent/realistic technology. As such, considerable freedom is given to proposers and a plethora of potential mission and craft designs exist.

Group Echo has designed a mission that balances ambition with plausibility. The basic goal in our design was to answer the question: **how quickly can we send a payload from LEO to the moon realistically?** Often due to the enormous costs of space missions, efficiency reigns as the most important factor of a project, but what if time were the most important factor? Perhaps there are humans starving or dying of thirst on a lunar colony and needed food and water as soon as possible. Perhaps a biological sample that perishes rapidly is sent to the moon to be studied. In scenarios like these, time is of the essence and the quickest spacecraft is needed. With this goal in mind, the name of our proposed craft is the Rapid Lunar Transporter (RLT). To begin to address the issue realistically, we established that the initial mass of our spacecraft was to be a mass that can be delivered to LEO by typical contemporary or upcoming American heavy launch vehicles such as the SpaceX Falcon Heavy and the United Launch Alliance Delta IV Heavy and Vulcan Centaur [1] [2]. Secondly, we established that our initial orbit around the Earth was in the ecliptic plane with the moon so that we could ignore inclination and treat the orbital mechanics as two-dimensional. With these initial parameters, a trajectory was developed that optimized for minimal transfer time to lunar orbit. Rather than optimize for the absolute minimum mission time to Sinus Medii, our design optimizes for the minimum transfer time for an *assured* landing at Sinus Medii. That is, the mission allots extra propellant to address trajectory adjustments and the trajectory includes an insertion into lunar orbit so that in the case of an emergency, the craft could safely orbit the moon as much as necessary before decent while corrections are made.

An alternative trajectory to this would be the circumlunar free return trajectory like that used in the Apollo missions, however since this mission is unmanned and unequipped to land in an atmosphere, this trajectory would not be as sensible. Of course, there are other more efficient trajectories, but the goal is to practically minimize mission time which generally comes at the cost of efficiency [3].

A. Spacecraft System Overview

The Rapid Lunar Transporter, with an initial mass of 29,200 kilograms, has two stages. The first stage uses liquid hydrogen and liquid oxygen propellant to conduct the translunar injection maneuver while in low Earth orbit which accounts for 44.2% of the total ΔV. The second stage with an initial mass of 10,044 kg uses the hypergolic propellants dinitrogen tetroxide (N_2O_4) and Aerozine-50 to conduct the three other maneuvers, accounting for 55.8% of the total ΔV. Many alternative fuels and staging configurations could have been used, however decisions made for the RLT balanced performance with reliability. A wealth of knowledge exists for similar staging configurations (ULA Centaur, McDonnell Douglas/Aerojet Delta-K [4][5]) that informed our design. The total time for the mission is 39.79 hours, a significant decrease in time compared to the landing portion of the Apollo missions and other lunar landing missions.

II. Design Description

A. Trajectory

A schematic of the mission trajectory appears in Figure 1. The mission includes three main phases and involves four orbital maneuvers. The first phase begins in low Earth orbit with the translunar injection maneuver ($\Delta V_1 = 3.8$ km/s) and lasts for 38.44 hours until the RLT arrives at the periapsis of the hyperbolic trajectory around the moon. The RLT enters the Moon's sphere of influence of 62,526 km after 18.15 hours, and arrives at the low lunar orbit altitude of 1838 km after the additional 20.28 hrs. The second phase begins with the low lunar orbit insertion maneuver ($\Delta V_2 = 3.07$ km/s) and lasts only until the RLT arrives at the far side of the moon with respect to the Earth, a travel time of 0.45 hours. The final phase is the half ellipse trajectory from low lunar orbit that begins with the descent burn ($\Delta V_3 = 0.023$ km/s) and terminates with the landing burn ($\Delta V_4 = 1.70$ km/s), for a travel time of 0.90 hours. This brings the total time of the trajectory to 39.79 hours or 1 day 15 hours 47 minutes and 28 seconds. The (ΔV) values, duration of the phases, and other orbital parameters were calculating following procedures presented in *Orbital Mechanics for Engineering Students* [6]. Note that to decrease the time to the moon, we have established the launch to occur such that the encounter with the moon occurs when the moon is at the periapsis in its orbit around the Earth.

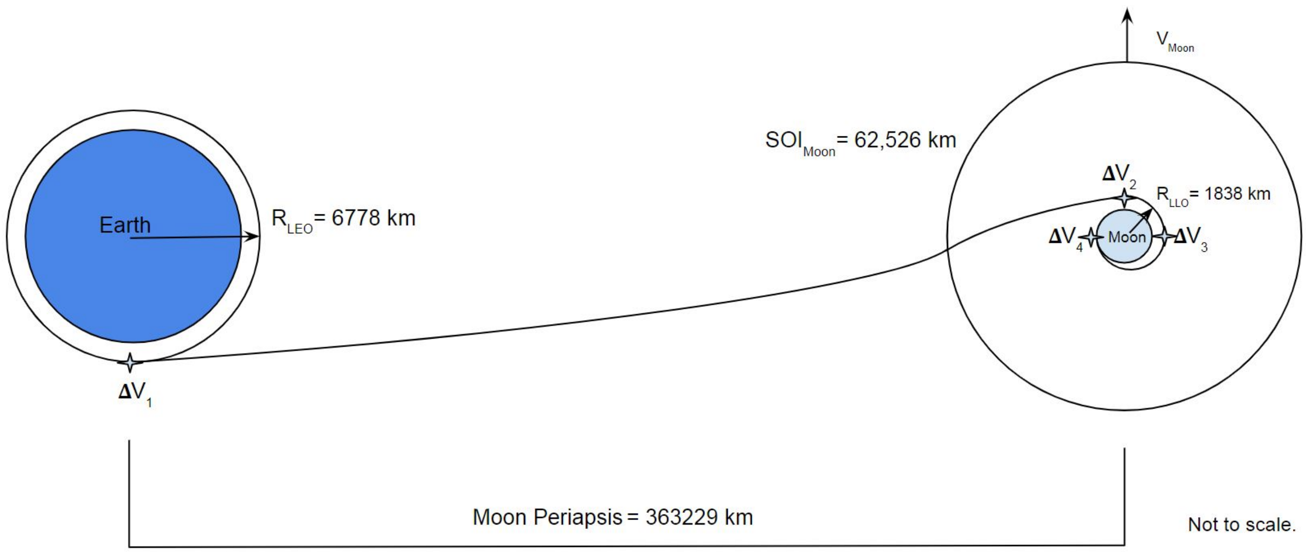

Fig. 1: Trajectory of the Rapid Lunar Transporter.

B. First Stage

The first stage of the rocket will boost the spacecraft from LEO onto a high-energy lunar injection trajectory. Due to the need for a large ΔV, as well as the short orbital duration of the rocket, liquid hydrogen and oxygen were chosen as the propellants. The piping and instrumentation diagram for the stage can be seen in Figure 2.

With the selection of fuel and oxidizer, an expander-cycle turbo-pump fed system is chosen for the first stage. The use of turbo-pumps enables the engine to operate at high chamber pressure to provide the necessary thrust for the first stage. The tank pressure practically becomes independent of the chamber pressure using the turbo-pump. This allows the propellants to be stored in a relatively low pressure environment, resulting in a reduction in pressure requirements on the propellant tanks and their feed lines. As a result, the use of turbo-pump can satisfy the high insertion velocity requirement during the first stage of the mission without increasing the vehicle tank weight significantly.

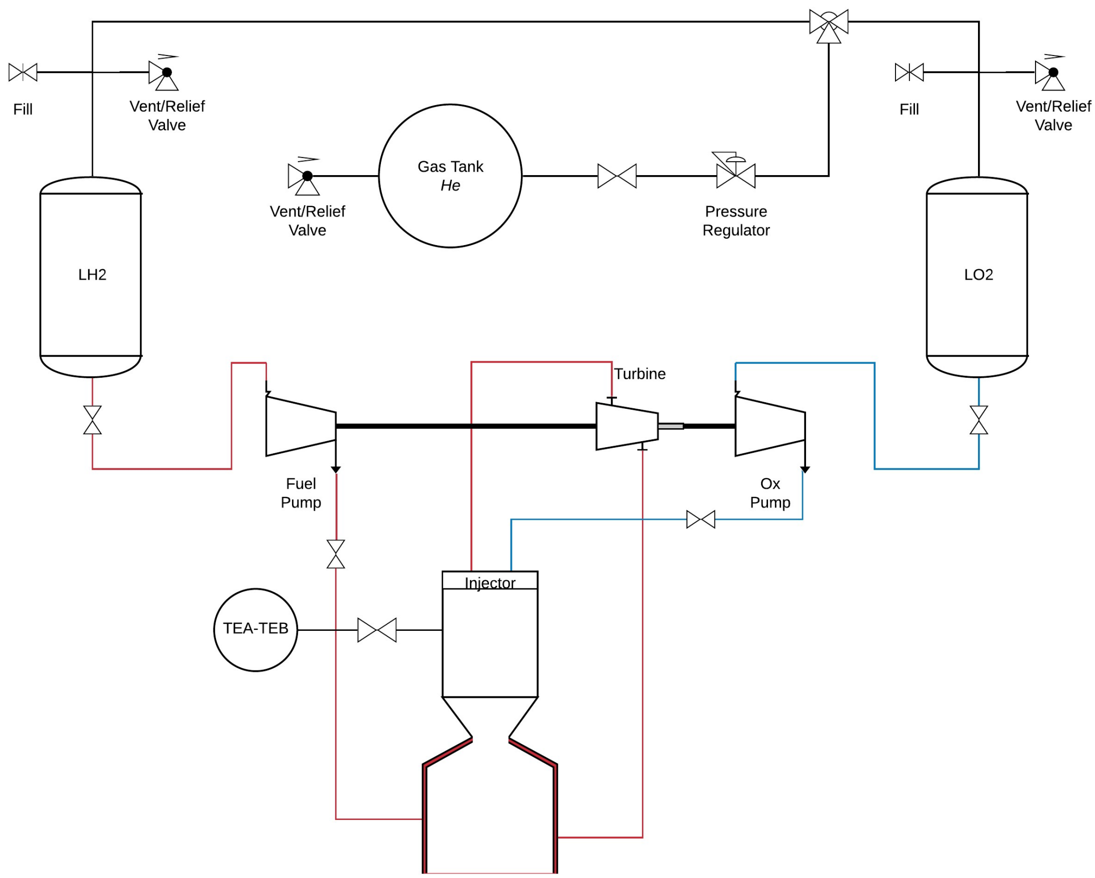

Fig. 2: P&ID of 1st Stage

With the choice of LH_2 and LO_2 as the fuel and oxidizer along with the estimation on the sizing of tanks and pipes, the pressure profiles of each propellant across each component is showed in Figure 3.

Based on the pressure difference and Eq. 1, the turbine needed to provide a power of 169.6 kW to the pumps for pressurization of both propellants to the desired injection pressure.

$$\hat{P}_{turbine} = \frac{\dot{m}_O \Delta P_O}{\rho_O} + \frac{\dot{m}_F \Delta P_F}{\rho_F} \tag{1}$$

The subscript O stands for oxidizer and subscript F is fuel. The $\hat{P}$ is the total power needed.

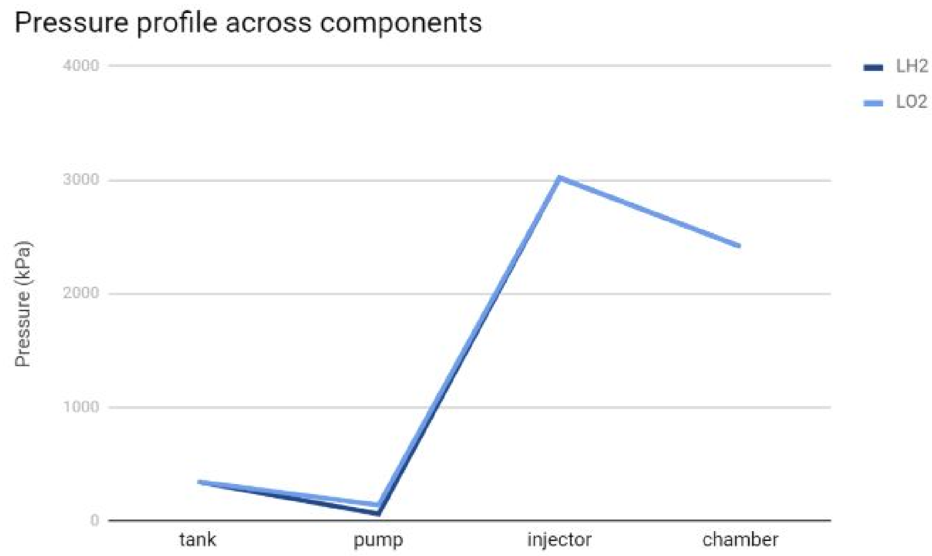

Fig. 3: Pressure profile across each component

The pressure of the helium tank was chosen to be $24.1MPa$. The required mass of helium needed to maintain pressure in the tanks was calculated using Eq. 2. This resulted in a helium mass of $43.4kg$, and a volume of $1.12m^3$

$$m_{He} = \frac{V_p P_p}{R_{He} T_{He}} \left[\frac{\gamma_{He}}{1 - \frac{P_p}{P_{He}}} \right] \tag{2}$$

Startup will be achieved by flowing the pressurized cryogenic hydrogen into the combustion chamber. The heating of the sub-20K hydrogen by the ambient-temperature combustion chamber and nozzle will provide energy to start the turbine, and oxygen will begin to flow. Ignition will be achieved by flowing Triethylaluminum-Triethylborane starter fluid into the combustion chamber, which is hypergolic with oxygen.

The first stage combustion chamber was designed to operate at a chamber pressure of $P_c = 2413$ kPa and a chamber temperature of $T_c = 3349$ K. From the selection of propellants with a mixture ratio of r = 5.9 and specific impulse of Isp = 448.64 s, the corresponding performance parameters was calculated based off a desired thrust of F = 100 kN and a area ratio of 84 [7]. The result were tabulated in Table. 1 below.

Table 1: First Stage Performance Parameters

P_i	P_c	P_e	T_c	T_e	γ_c	A_c
3016 kPa	2413 kPa	1.574 kPa	3349 K	1169 K	1.134	0.0648 m^2
A_t	A_e	L^*	V_e	**C**	C^*	C_F
0.0216 m^2	1.8144 m^2	0.85 m	3866 m/s	4397 m/s	2293 m/s	1.9176

The propellants of the first stage will be stored in 301 stainless steel balloon tanks, and will make up the main structure of the rocket. This structure can be seen in Figure 4. The hydrogen will be stored in the upper tank. Due to its spherical design, the stress in the walls is calculated as $\sigma = \frac{Pr}{t}$, with a pressure of $344kPa$ and a tank radius of $2m$, the tank walls can be 0.7mm thick while maintaining a factor of safety of 1.3. The liquid hydrogen and oxygen share a common bulkhead. Due to the temperature differential between the two propellants, fiberglass honeycomb will be used to insulate between the tanks. The helium for pressurization will be stored in tanks above the hydrogen tank at $24MPa$, and the TEA-TEB starter fluid is stored below the oxygen tank at $3.5MPa$.

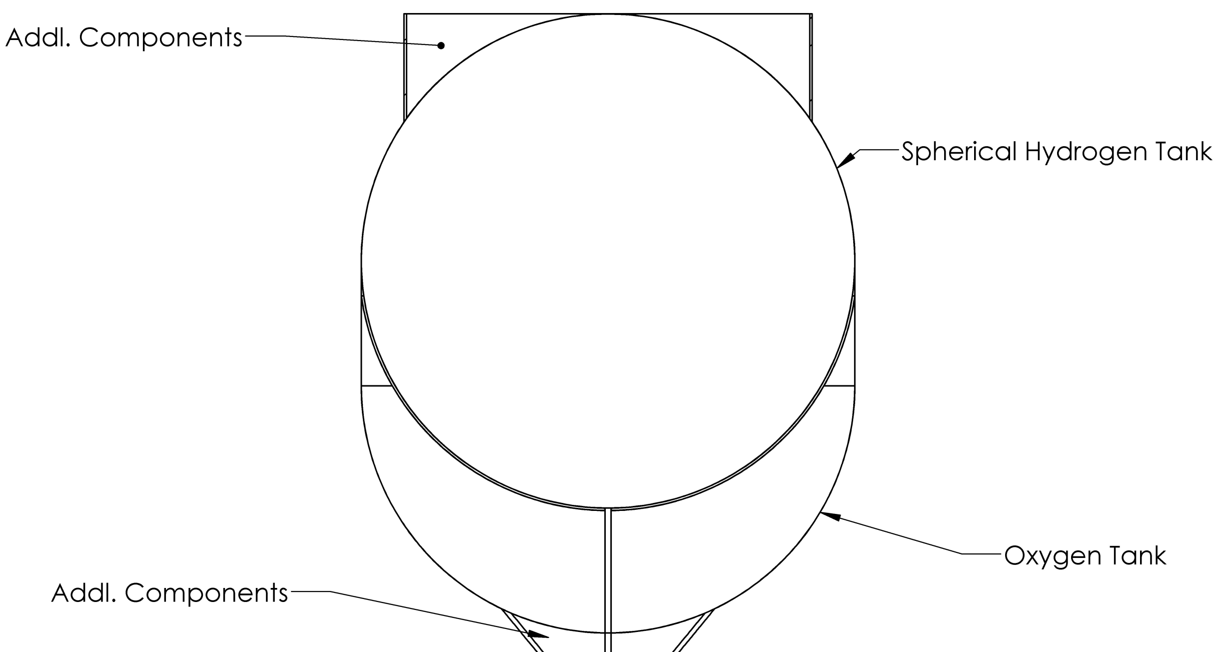

Fig. 4: Stage 1 tanks and structure

C. Second Stage

The purpose of the second stage rocket is to slow the spacecraft down into capture orbit with the moon, and then safely land on the surface to deliver the payload. This maneuver requires $\Delta V = 3.07$ km/s for the capture, a small burn of $\Delta V = 23.1$ m/s for a Hohmann transfer, and $\Delta V = 1.70$ km/s for landing, giving a total budget of $\Delta V = 4.80$ km/s. The upper stage of the rocket has an initial mass of $m_0 = 10044$ kg, a majority of which consists of propellant mass, with a value of $m_p = 8075$ kg. A mass breakdown of the remaining structural components is listed in Table 2 below.

Table 2: Structural components and their associated mass

Component	**Mass (kg)**
Fuel tanks	53.36×2
Oxidizer Tanks	65.8×2
Pressurant	20.55
Pressurant tank	106.77
Engine Components	104
Batteries	61×4
RCS System	50
Avionics	272.87
Wiring	135.71
Total	**1172.22**

For the second stage of the rocket, a storable liquid bipropellant system was chosen for the engine. It uses a hypergolic propellant combination of Aerozine 50 as fuel, which consists of 50% hydrazine and 50% unsymmetrical dimethylhydrazine (UDMH), and dinitrogen tetroxide as its oxidizer. An oxidizer/fuel mixture ratio (by weight) of 2 was determined to be optimal in providing sufficient thrust without requiring too large of an oxidizer tank. A pressure-feeding system is used to deliver the propellants to the combustion chamber, eliminating the complexity involved in using turbo-pumps and allowing for easier implementation of throttling control at the cost of incurring additional mass to store the pressurized helium. As such, chamber pressure was chosen to be lower so as to minimize the size of the feed system. Throttling is controlled with a series of electronically operated valves, allowing for multiple burns simply by opening or closing the valves. Additionally, more precise maneuvers requiring greater thrust control can be done by regulating gas flow, giving up to 70%

throttability without stability issues due to unwanted pressure drops across the injector. The rocket is configured such that there are two of each fuel and oxidizer tanks, oriented in a circle around the helium tank.

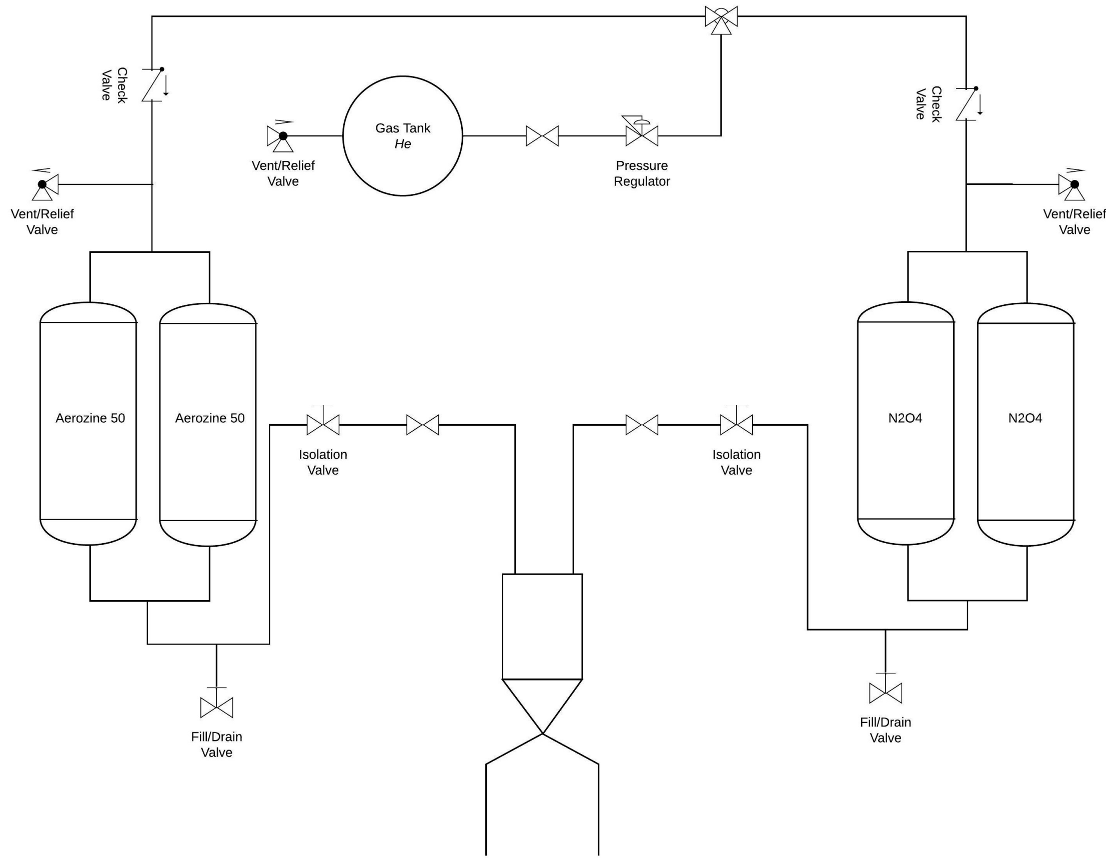

Fig. 5: P&ID of 2nd Stage

The combustion chamber was designed to be at P_c = 896.32 kPa, and is lined with ablative material in order to withstand the high temperatures during operation, rather than the regenerative cooling method used in the first stage typically used with gas generator cycles. A chamber contraction ratio of 3 was chosen to keep chamber weight as low as possible without risking incurring greater Rayleigh losses. Similarly, the supersonic nozzle area ratio of 65 was selected to provide sufficient Isp values without incurring too much additional weight.

Based on the design choices made for the type of propellant, mixture ratio, chamber pressure, area ratios, and a desired thrust of F = 55.4 kN, the specific impulse for the second stage is Isp = 327.81 s, and propellant mass flow rate $\dot{m}$ = 17.22 kg/s. A complete analysis on the resulting performance parameters are calculated and tabulated below.

Table 3: Second Stage Performance Parameters

P_i	P_c	P_e	T_c	T_e	γ_c	A_c
1120.4 kPa	896.32 kPa	0.997 kPa	3109.78 K	1158.8 K	1.131	0.09774 m^2
A_t	A_e	L^*	V_e	**C**	C^*	C_F
0.03258	2.1177	0.8 m	3217.18 m/s	3215.9 m/s	1695.9 m/s	1.8962

Fuel and oxidizer will be stored in a pair of spherical composite-overwrap pressure vessels each arranged radially around the central rocket motor and helium pressurant tank. These tanks will be mounted to an aluminum-space frame structure. The payload will be mounted above the tanks. This configuration can be seen in Figure 6.

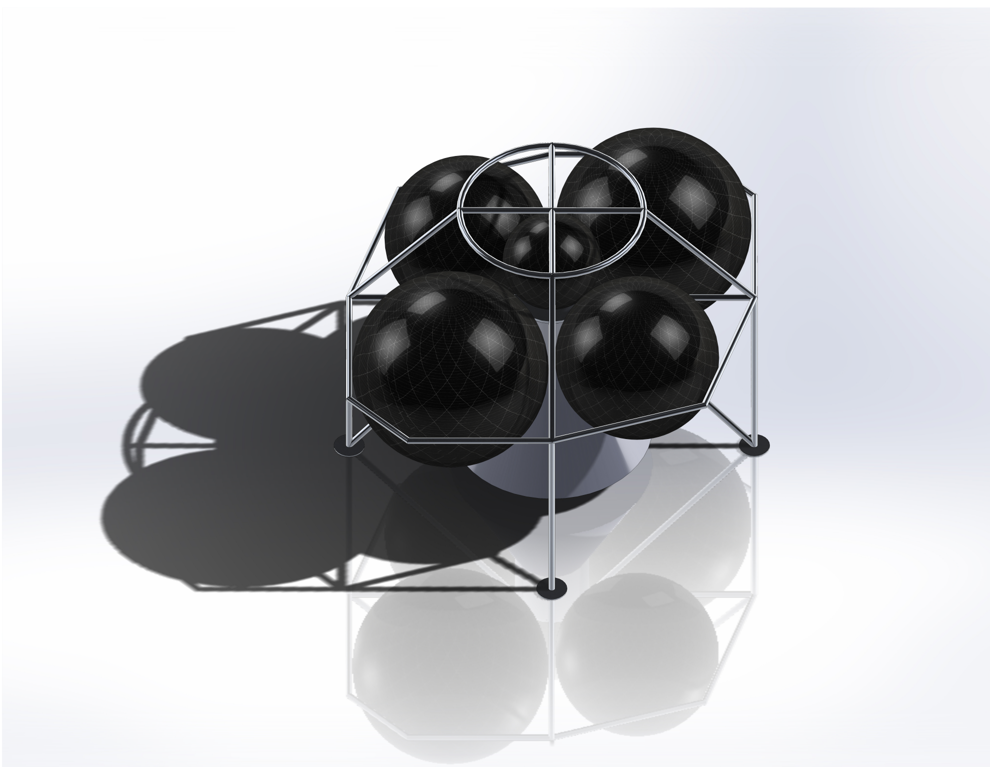

Fig. 6: 2nd stage structure and tank configuration

The pressures in the combined fuel tanks and oxidizer tanks are set to be at 1154.4 kPa and 1184.4 kPa respectively. With 2.5 m tubing running from the tanks to the injector as well as passing through valves, a combined pressure drop of roughly 100 kPa exists between the tanks and the injector. This leaves the pressure at the injector to be 1120.4 kPa.

D. Injector Design

The injector of choice for both our engines is a pintle injector exhibiting discrete axial flow for the oxidizer and swirl radial flow for the fuel. A visual representation of these flows, as well as the combined flow, are given in Figure 7. The injector diameter and length from the injector face will be about 1/3 the diameter of the combustion chamber, or 0.10m for the first stage and 0.12m for the second stage. Considering hydraulic flow of an incompressible fluid through an orifice and an approximate pressure drop across the injector of 25%, the total area of the metering orifices should be 0.233m^2 for the first stage and 0.078m^2 for the second stage.

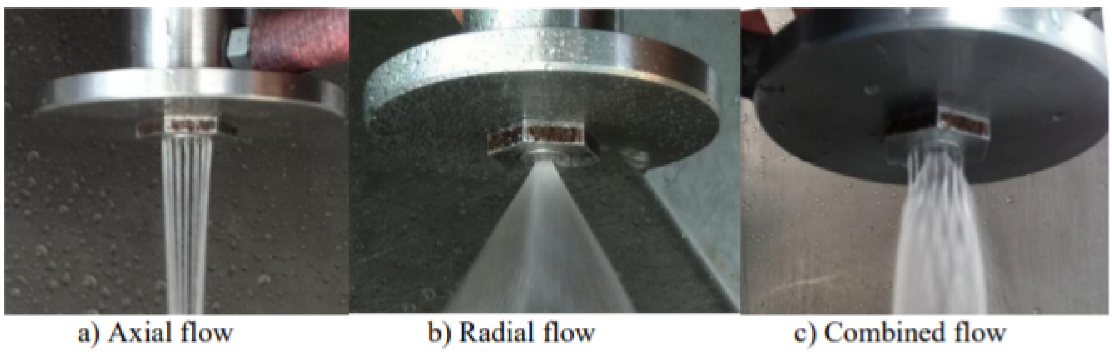

Fig. 7: A pintile injector showing fluid flow of a) oxidizer alone b) fuel alone and c) both flows combined [8]

E. Other Design Considerations

Launch

The RLT is intended to be launched to LEO by a single rocket. Because of this, it is packaged to fit inside a fairing. Figure 8 shows the RLT inside the payload volume of a Falcon 9 fairing [9]. This configuration would allow for a payload volume of 8.85m^3, which results in a payload density of 102$\frac{kg}{m^3}$. If the payload is larger, a stretched fairing could be used during launch.

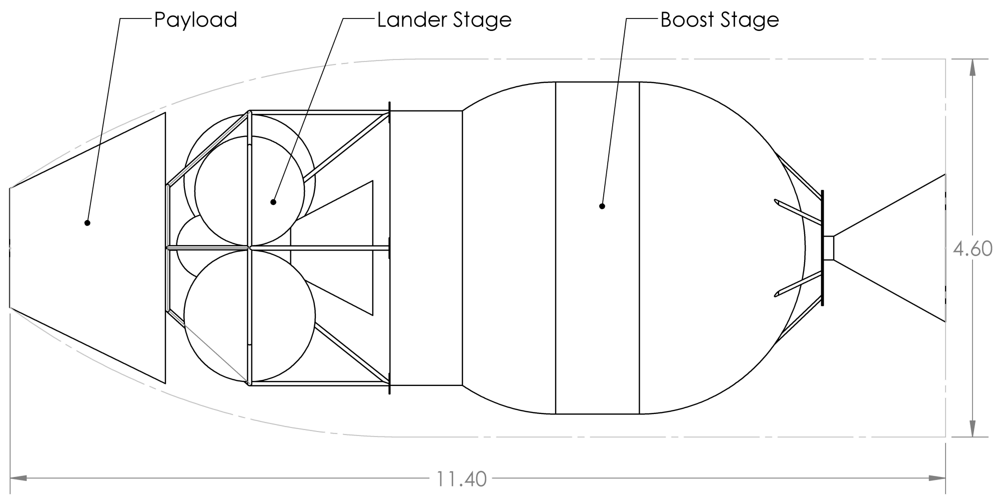

Fig. 8: RLT stack inside Falcon 9 fairing

Thermal Protection Systems

The Rapid Lunar Transport employs active and passive thermal protective systems. The primary system for the first stage is the Earth: the ignition of the first stage will occur shortly after ground launch and will occur at night to ensure minimal boil off of the cryogenic propellants. Secondly, the nozzles will be cooled to avoid heating components of the spacecraft and contain as much energy as possible in the exhaust. The nozzle of the first stage engine will be regeneratively cooled by the liquid hydrogen propellant as can be seen in the red lines in Figure 2. The second stage engine nozzle is made of a niobium alloy capable of radiative cooling, similar to that used in the SpaceX Merlin Vacuum 1C Engine [10]

Second, the combustion chambers for both the first and second stage are made of ablative materials that can shed some of the heat that would have been delivered to the spacecraft. For best heat transfer, the first stage will be made from a copper alloy, and the second stage will be aluminum.

Control Systems

The spacecraft is controlled via an onboard computer in the payload and a preloaded program that makes any necessary corrective maneuvers. Diagnostics and telemetry are monitored on the ground via payload communication devices. Additionally, there is on board power in the form of small solar panels and battery storage, adding 273 kg to the mass of the rocket.

Alongside the gimballing capability of each stage's engine, the second stage also includes a reaction control system (RCS). This consists of four sets of monopropellant thruster quads that are embedded internally into the frame of the second stage around the main propellant tanks. Although not shown in the Figure 5 or Figure 8, the thrusters are controlled by piping and a small Aerozine-50 tank that altogether have a mass of 50 kg.

III. Assessment of Design

A. Mission Assessment

The chosen trajectory is neither the fastest nor most efficient method to get to the moon. However, it does significantly reduce the mission time to under 40 hours, while still using proven, reliable technologies, and a single launch. The integration of the RLT to booster structure was not investigated, but a system of struts and connectors could be used to achieve this.

The required ΔV value for each stage is 3.8 km/s and 4.8 km/s for the first and second stages, respectively. To achieve this, the first stage has $16700kg$ of propellant and a structural mass of $2450kg$, and the second stage has a propellant mass of $8175kg$ and a structural mass of $1070kg$. Using Eq. 3, the final ΔV value of each stage is 3.90 km/s and 5.4 km/s. This results in a 2.6% ΔV margin for the first stage, and a 12.5% margin for the second stage. This large margin for the second stage can be used for course corrections, as well as possible adjustments during landing.

$$\Delta V = V_e \, ln\left(\frac{m_0}{m_f}\right) \tag{3}$$

The orbital maneuver calculations made for this design are inherently inaccurate considering they use the method of patched-conics and assume instantaneous changes in velocity (non-impulsive). While this provides an accurate estimate of the actual ΔV values, a more precise method (such as a restricted 3-body probelm) should be developed if the consortium were to select the design.

B. First Stage

Due to the simple needs of the first stage, the design focus was on performance and simplicity. The use of cryogenic propellants without thermal management would result in a large performance loss if used for long-duration missions, due to propellant boil-off. However, because the stage is only being used for a single burn out of LEO, boil-off is unimportant. This allows for mass to be saved that would be used for thermal insulation, but does limit the usefulness of the first stage to boost burns, preferably in the night sky. Additionally, care must be taken to keep the tanks inflated at all times, or the structure will likely collapse. This is not unprecedented in rocketry though, as both the atlas rocket and centaur upper stage use balloon tanks.

C. Second Stage

Throughout the process of designing the second stage, the main objective in mind was to get the payload to the surface of the moon quickly, reliably, and efficiently. Keeping the structural mass as low as possible without compromising thrust was essential in achieving that goal. In addition to a faster mission completion time, the overall mass of this stage of the rocket is less than that of the Apollo lunar module [3]. This lighter and faster spacecraft results in a lower required propellant budget and is consequently more cost efficient. There are however drawbacks to this system, as the throttling capabilities of this second stage is much less precise than that of the lunar descent module, which was capable of up to 10% of maximum thrust. Furthermore, the use of a bipropellant hypergolic engine inherently results in lower thrust and efficiency when compared other engine types, however the reliability and simplicity of hypergolic propellants is worth the loss in performance for.

IV. Conclusions and Recommendations

The RLT allows for rapid transport to the moon, while still being attainable with current technology. The LH2/LOX first stage allows for a single boost towards the moon, while the hypergolic second stage slows and lands on the lunar surface. Both of these systems rely on well-understood and proven technologies. However, there may be room to incorporate more advanced technologies and materials in the propulsion system to reduce weight and improve efficiency. Additionally, the structure could benefit from additional analysis and optimization. The RLT shows what is achievable with current capabilities, and can provide a pathway to the moon for time critical cargo.

Acknowledgements

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References

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Appendix A: Orbital Mechanics MATLAB Script

Contents


```
close all; clear all;

Rpm = 363228.9*10^3;
em = 0.0549;
Mm = 7.342*10^22;
rm = 1737.4*10^3;

Me = 5.972*10^24;
re = 6378.1*10^3;

G = 6.67408*10^-11;
mue = G*Me;
mum = G*Mm;
soim = Rpm*(mum/mue)^(2/5);
Rsoi = (mum/mue)^(2/5)*Rpm;
Encounter = Rpm-Rsoi;

Vpm = sqrt((em+1)*mue/Rpm);
v400 = sqrt(mue/(400*10^3+re));
```

Transfer burn

```
DV1 = 3800;

vpt = v400+DV1;
rpt = 400*10^3+re;
Et = vpt^2/2-mue/rpt;
ht = rpt*vpt;
```

```
et = rpt*vpt^2/(mue)-1;

vint = sqrt((Et+mue/Rpm)*2);
phit = acosd(ht/(Rpm*vint));
Vint = [vint*cosd(phit);vint*sind(phit)]; % [//,r]
thetainf1 = acosd(-1/et);
```

SOI transfer

```
Vinf = [Vint(1)-Vpm;Vint(2)];
Eins = norm(Vinf)^2/2;
```

circulization burn

```
RpIns = 1838*10^3;
e2 = 1+RpIns*norm(Vinf)^2/mum;
Beta = acosd(1/e2);
thetainf2 = acosd(-1/e2);

Vpins = sqrt((Eins+mum/RpIns)*2);
hins = RpIns*Vpins;
Vorbit = sqrt(mum/(RpIns));
DVins = Vpins-Vorbit;    % circular holding orbit delta-v
```

homann to surface

```
h = sqrt(2*mum)*sqrt(RpIns*rm/(RpIns+rm));
DVhomann = Vorbit-h/RpIns;
DVlanding = h/rm;

%DVh

DVsum = DVlanding+DVhomann+DVins+DV1;
DVs2 = DVlanding+DVhomann+DVins;
```

Time Calculations

```
% 2 Hyperbola
```

```
theta = @(h,r,mu,e)acosd((h^2)/(r*mu*e) - 1/e);

Mh = @(e,theta)((e*sqrt(e^2 - 1)*sind(theta))/(1+e*cosd(theta))...
    - log((sqrt(e+1)+sqrt(e-1)*tand(theta/2))/(sqrt(e+1)-sqrt(e-1)*tand(theta/2))));
F = @(e,theta)(log((sqrt(e+1)+sqrt(e-1)*tand(theta/2))/(sqrt(e+1)-sqrt(e-1)*tand(theta/2

t = @(Mh,mu,h,e)((Mh*h^3)/(mu^2))*(1/((e^2 -1)^(3/2)));


theta1 = theta(ht,Encounter,mue,et);
F1 = F(et,theta1);

Mh1 = Mh(et,theta1);
t1 = t(Mh1,mue,ht,et);
t1_hrs = t1/3600;
theta2 = theta(hins,Encounter,mum,e2);

Mh2 = Mh(e2,theta2);
t2 = t(Mh2,mum,hins,e2);
t2_hrs = t2/3600;
% Circular?
t3 = (2*pi/(sqrt(mum))*(rm+1000)^(3/2))/4;
t3_hrs = t3/3600;
% Hohmann to surface

a = (rm+(rm+1000))/2;

t4 = (2*pi/(sqrt(mum))*a^(3/2))/2;
t4_hrs = t4/3600;
ttotal = (t1+t2+t3+t4);
ttotal_days = ttotal/(24*3600);
```