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# Multidisciplinary Design Optimization of a Reusable Lunar Vehicle

*L. Beauregard<sup>1</sup>, Dr. A. Urbano<sup>1</sup>, D. Colbeck<sup>2</sup>, Dr. S. Lizy-Destrez<sup>1</sup>, and Prof. Dr. J. Morlier<sup>1,3</sup>*

*<sup>1</sup>Institut Supérieur de l'Aéronautique et de l'Espace (ISAE-SUPAERO), Toulouse, France*

*<sup>2</sup>Royal Melbourne Institute of Technology, Melbourne, Australia*

*<sup>3</sup> Université de Toulouse, ISAE-SUPAERO-INSA-Mines Albi-UPS, CNRS UMR5312, Institut Clement Ader, Toulouse, France*

*laurent.beauregard@isae-supero.fr*

## Abstract

The Lunar Orbital Platform-Gateway will be the successor to the ISS and will be placed around the Moon. To bring crew onto the lunar surface, a lunar lander must be designed and used. This work will present a system design tool for lunar landers which utilizes OpenMDAO, a multidisciplinary design optimization library. Moreover, different mission architectures will be compared independently. As a benchmark, a design for a one-stage LH2/LOX will be produced and compared to an existing design.

## 1. Introduction

With its end of life approaching, the international community is looking for a replacement to the International Space Station (ISS). The leading candidate is a space station, called the Lunar Orbital Platform-Gateway (LOP-G) or Gateway, placed in a L2 southern Near Rectilinear Halo Orbit (NRHO). The orbit would have a periapsis altitude of 1500 km, an apoapsis altitude of nearly 70,000 km and a period of roughly 6 days and 13 hours. This station would serve as a hub for human exploration and scientific progress. The NRHO is strategically chosen to have constant Earth communication, nearly no solar eclipses, low  $\Delta v$  maintenance cost, high observability of the lunar south pole, relative ease of access from the Earth and decent access to the lunar surface [20]. A dedicated vehicle will be required to bring crew and payload on the surface. However no design for this "lunar shuttle" has been chosen yet by the international community. Objectives for this vehicle are varied but often include the ability to bring 2 to 4 crew members unto the lunar surface for a duration of 2 to 14 days. The vehicle should have some level of reusability and be refueled at the LOP-G in between the missions.

The present work focuses on two objectives, first to establish a list of viable mission architectures and the second is the development of a tool for the preliminary sizing of a lunar lander. To tackle the design of such complex systems, Multidisciplinary Design Optimization (MDO) has gained significant traction in the past decades and has had a number of successes in the field of aerospace[12][8]. Application to the space sector has also seen a recent surge of interest [2][11]. However few research has focused on applying MDO to the design of a reusable lunar lander. In this work, a MDO software using the OpenMDAO library [9] is used as a preliminary design tool. In chapter 2, the concepts, nomenclature and mathematical framework behind MDO are introduced. In section 3, the different disciplines of propulsion, trajectory and structure are introduced along with the objective function. Chapter 4 will described the results obtained: first optimal trajectories will be analyzed, then mission architectures will be optimized independently and lastly a MDO will be performed for a one-stage LH2/LOx lunar lander and compared to an existing design.

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## 2. Multidisciplinary analysis and optimization

At the heart of a MDO is the solving of a nonlinear optimization problem, the formulation used in this article is

$$\begin{aligned}
 & \min_{x,y,z} f(x, y, z) \\
 & \text{Subject to } g(x, y, z) = 0 \\
 & \quad h(x, y, z) \geq 0 \\
 & \quad y = C(x, y, z) \\
 & \quad R(x, y, z) = 0
 \end{aligned} \tag{1}$$

where  $x \in \mathbb{R}^N$  are called the design variables,  $y \in \mathbb{R}^D$  are the coupling variables and  $z \in \mathbb{R}^S$  are the state variables

$f : U_f \subseteq \mathbb{R}^N \times \mathbb{R}^D \times \mathbb{R}^S \rightarrow \mathbb{R}$  is the objective function to minimize

$g : U_g \subseteq \mathbb{R}^N \times \mathbb{R}^D \times \mathbb{R}^S \rightarrow \mathbb{R}^{M_{eq}}$  are the equality constraints

$h : U_h \subseteq \mathbb{R}^N \times \mathbb{R}^D \times \mathbb{R}^S \rightarrow \mathbb{R}^{M_{ineq}}$  are the inequality constraints

$C : U_C \subseteq \mathbb{R}^N \times \mathbb{R}^D \times \mathbb{R}^S \rightarrow \mathbb{R}^D$  are the disciplines

$R : U_R \subseteq \mathbb{R}^N \times \mathbb{R}^D \times \mathbb{R}^S \rightarrow \mathbb{R}^S$  are the residuals

The design variables  $x$  are independent parameters that can be modified to optimize the system. The coupling variables,  $y$  are outputs of the disciplines. The states variables  $z$  are internal parameters that are necessary in the computation of the various models.

Considering the disciplines  $C(x, y, z)$  and residuals variables  $R(x, y, z)$ , the process of solving for  $z$  and  $y$  given a choice of design variables  $x$  is called a multidisciplinary analysis (MDA). Such system can be solved in a variety of ways; in general, a Newton's method or quasi-Newton's method is employed to this effect. For the discipline equations  $C$ , fixed point iteration schemes, such as Gauss-Seidel or Jacobi method can be used.

In this article, only monolithic architectures are considered, that is, only one system optimizer is present. Within monolithic architectures, several methods to optimize the system are applicable. The distinctive feature between these methods is how the variables  $y$  and  $z$  are handled by the optimizer, either solving for them explicitly at every step of the optimization or letting the optimizer handle the variables. These choices separates the architecture into 4 categories, see table 1

- Multidisciplinary Feasible - MDF
- Individual Discipline Feasible - IDF
- Simultaneous Analysis and Design - SAND
- All-At-Once - AAO

Table 1: Types of monolithic optimizer

	y is solved	y is free
z is solve	MDF	IDF
z is free	SAND	AAO

Whichever architecture is ultimately picked, an optimization algorithm must be chosen, those include BFGS, COBYLA, CG, Powell, Nelder-Mead and SLSQP among others. The choice of the algorithm depends on the nature of the problem, the constraints present and whether analytical derivatives are available. Refer to the documentation of the optimization toolbox of Python for more information[6].

## 3. Models

To perform multidisciplinary analyses and optimization of the system, models of the subsystems must be chosen, along with an objective function. For a lunar lander system, 3 main subsystems have been identified: propulsion, trajectory and structure. These disciplines are coupled through inputs and outputs.

### 3.1 Propulsion

The working principle of a chemical rocket engine is that reactants are combined in a chamber generating a vast amount of energy in the form of heat which is then converted to mechanical work by a nozzle. In this work, only liquid-fuel engines are considered. The objective of the propulsion discipline is to calculate the specific impulse  $I_{sp}$ , rocket engine mass  $M_{Engine}$  and tank mass  $M_t$  from the type of reactants, mixture ratio  $R_m$ , combustion chamber pressure  $P_c$ , expansion ratio  $\epsilon$  and total propellant  $M_p$ . The propulsion module is separated into 3 subdisciplines, the combustion analysis, the fluid expansion and the tank system.

The state of the art for combustion analysis is the NASA computer program Chemical Equilibrium with Applications (CEA). Given the fuel type,  $R_m$  and  $P_c$ , CEA computes the combustion temperature  $T_c$ , the adiabatic index  $\gamma$  and the average molecular weight  $M_w$  of the combustion products. Given an expansion ratio,  $\epsilon$ , one can solve for the pressure at the exit,  $P_e$

$$\epsilon = \sqrt{\frac{\gamma - 1}{\gamma + 1} \left( \frac{2}{\gamma + 1} \right)^{\frac{1}{\gamma - 1}}} \frac{1}{\sqrt{\left( \frac{P_e}{P_c} \right)^{\frac{2}{\gamma}} - \left( \frac{P_e}{P_c} \right)^{\frac{\gamma + 1}{\gamma}}} } \quad (2)$$

Where  $R = 8.314 \frac{J}{K mol}$  is the ideal gas constant. From the variables above, the speed of the exhaust  $v_e$  can be found to be

$$v_e = \sqrt{\frac{2\gamma}{\gamma - 1} \frac{RT_c}{M_w} \left( 1 - \left( \frac{P_e}{P_c} \right)^{\frac{\gamma - 1}{\gamma}} \right)} \quad (3)$$

The (vacuum) specific impulse can be found to be

$$I_{sp} = \frac{1}{g_0} \left( v_e + \frac{RT_c}{M_w v_e} \left( \frac{P_e}{P_c} \right)^{\frac{\gamma - 1}{\gamma}} \right) \quad (4)$$

In this work, the estimation of the mass of the rocket engines will be based on regression of existing engines. The choice of the model has an impact on the optimization possibilities, for example, some models includes the effects of the expansion ratio and some do not. Trying to optimize the expansion ratio with a model that does not accurately take it into account will result in nonsensical results.

$$M_{Engine}[kg] = T[N] \left( 7.81 \times 10^{-4} + 3.37 \times 10^{-5} \sqrt{\epsilon} \right) + 59 \quad (5)$$

$$M_{Engine}[t] = 0.0135 \times N_{eng}^{0.4118} \times T[kN]^{0.471} \times m_{propellant}[t]^{0.3574} \quad (6)$$

$$M_{Engine}[kg] = 0.00514 \times T[N]^{0.92068} \quad (7)$$

$$M_{Engine}[kg] = \frac{T[N]}{107.4 \ln(T[N]) - 792} \quad (8)$$

where  $T$  is the thrust of the engine,  $N_{eng}$  is the number of engines,  $m_{propellant}$  is the mass of propellant. With equations 5, 6, 7 and 8 coming from [1][16][21] and [18] respectively. Other regressions are considered in Castellini [4].

Given a propellant mass  $M_p$  and mixture ratio  $R_m$ , the mass of fuel  $M_F$  and oxidizer  $M_O$  are given by

$$M_F = \frac{1}{1 + R_m} M_p \quad (9)$$

$$M_O = \frac{R_m}{1 + R_m} M_p \quad (10)$$

Tank mass depends on the geometry of the vessel, usual shapes are spheres or cylinders (or shapes approximating those). The approximate relation between the tank mass,  $M_t$ , the pressure of the fluid  $P$ , the volume  $V$ , the density of the tank material  $\rho_t$  and the maximum allowable stress  $\sigma$  is given by

$$M_t = S_f c P V \frac{\rho_t}{\sigma} \quad (11)$$

where  $c$  is a constant that depends on the geometry of the vessel,  $c = 3/2$  for a sphere,  $c = 2$  for a cylinder and  $S_f \sim 2$  is a safety factor. Since the volume of the fuel/oxidizer is given by the total mass of fluid  $M_L$  divided by its density  $\rho_L$  a near linear relationship exist between the tank mass and the fuel/oxidizer mass.

$$M_t = \left( S_f c \frac{P}{\sigma} \frac{\rho_t}{\rho_L} \right) M_L = \alpha M_L \quad (12)$$

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This constant can be obtained from various existing tanks [1]

- Hydrogen,  $\alpha = 0.128$
- Liquid oxygen,  $\alpha = 0.0107$
- RP-1,  $\alpha = 0.0148$
- LCH4,  $\alpha = 0.0287$

Table 2: Propulsion discipline Input/Output

Inputs	Symbol	Outputs	Symbol
Type of fuel/oxidizer	-	Combustion temperature	$T_c$
Mixture ratio	$R_m$	Product molecular weight	$M_w$
Combustion chamber pressure	$P_c$	Adiabatic index	$\gamma$
Expansion ratio	$\epsilon$	Specific impulse	$I_{sp}$
Maximum thrust	$T$	Engine mass	$M_{\text{Engine}}$
Propellant mass	$M_p$	Propellant tank mass	$M_t$

### 3.2 Trajectory

Efficient ascent and descent trajectories to and from the surface of Moon are primordial for mass saving. The input/output of the trajectory discipline are summarized at the end of this section in table 3. The trajectory can be modeled in two phases as shown in figure 1.

- Phase 1: Near Rectilinear Halo Orbit (NRHO) to Low Lunar Orbit (LLO).
- Phase 2: LLO to lunar surface

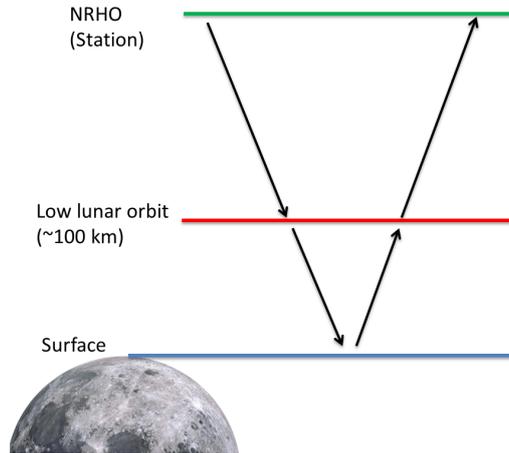


Figure 1: Schematic of the mission

Phase 1's dynamics can be modeled by the circular restricted three body problem (CR3PB) whose dynamics is given in a rotating frame by

$$\ddot{\vec{r}} = -\frac{\mu_E}{|\vec{r} - \vec{r}_E|^3} (\vec{r} - \vec{r}_E) - \frac{\mu_M}{|\vec{r} - \vec{r}_M|^3} (\vec{r} - \vec{r}_M) - \vec{\omega} \times (\vec{\omega} \times \vec{r}) - 2 \vec{\omega} \times \dot{\vec{r}} \quad (13)$$

where  $\vec{r}$  is the vectorial position of the spacecraft with respect to the Earth-Moon Barycenter,  $\mu_E$  and  $\mu_M$  are the standard gravitational constant of the Earth and Moon respectively and  $\vec{\omega}$  is the vectorial representation of the rotation of the Earth-Moon system. The thrust to weight ratio (TWR) of the lander is assumed to be greater than one because

it is necessary to lift-off from the lunar surface, this allows the maneuvers to be considered impulsive as a first order estimate. The problem then becomes a boundary value problem. Values of  $\Delta v$  from the NRHO to LLO varies between 730 m/s to 900 m/s depending on when during a lunar orbit the transfer is performed [20].

Phase 2's dynamics can be modeled by the following system of differential equations

$$\dot{r} = v_r \quad (14)$$

$$\dot{\theta} = \frac{v_t}{r} \quad (15)$$

$$\dot{v}_r = \frac{T}{m} \sin(\alpha) - \frac{\mu_M}{r^2} + \frac{v_\theta^2}{r} \quad (16)$$

$$\dot{v}_t = \frac{T}{m} \cos(\alpha) - \frac{v_t v_r}{r} \quad (17)$$

$$\dot{m} = -\frac{T}{w} \quad (18)$$

where  $r$  is the distance from the craft to the center of the Moon,  $\theta$  is the angle of the craft with respect to a reference point on the surface (usually the starting point),  $v_r$  is the radial component of the velocity,  $v_\theta$  is the tangential component of the velocity,  $\mu_M$  is the standard gravitational parameter of the Moon,  $m$  is the total mass of the system,  $T$  is the thrust generated by the engines,  $w = I_{sp}g_0$  is the effective exhaust velocity and  $\alpha$  is the pitch angle. Since gravity losses

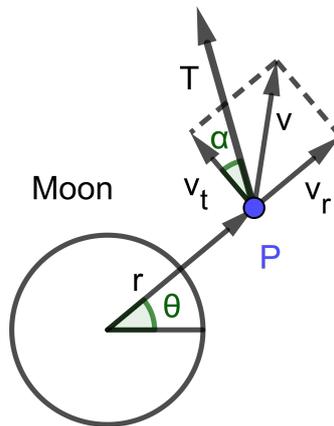


Figure 2: Geometry of the ascent/descent trajectory variables

are much more important in phase 2 than phase 1, the application of optimal control theory is of greater benefit.

The subject of optimal control is a vast and still is an ongoing area of research. The problem is usually formulated in the following manner

$$\min_u \{S\} \quad (19)$$

where

$$S = \phi(t_f, x_f) + \int_{t_i}^{t_f} \mathcal{L}(t, x, u) dt \quad (20)$$

subject to

$$\dot{x} = f(t, x, u) \quad (21)$$

$$h(t, x(t), u(t)) \geq 0 \quad (22)$$

$$\Phi(t_f, x_f) \geq 0 \quad (23)$$

where  $S$  is composed of two costs:  $\phi$ , the boundary cost and  $\mathcal{L}$ , the running cost,  $x$  are the state variables,  $u$  are the control variables,  $f(t, x, u)$  is the dynamics of the states,  $h(t, x, u)$  are path constraints and  $\Phi$  are boundary constraints. Software developed to solve such problems broadly fall into two main categories: indirect vs direct methods. In this work, only direct methods will be considered. Direct methods discretizes the dynamics/variables/cost of the system.

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This discretized system is then fed into a nonlinear optimizer such as SNOPT or IPOPT. A broad class of algorithms are called "Pseudospectral methods" where both the states and the control variables are discretized and approximated by orthogonal polynomials [10].

For the trajectory discipline, the objective function of the optimal control problem only has a boundary term: the amount of fuel necessary to achieve orbit  $S = M_p$ . The system dynamics is given by equations 14 to 18. The only constraint a priori is  $r \geq R_{\text{Moon}}$ , that is, the craft's trajectory must not intersect the Moon's surface. The optimal control was solved with the software "Dymos: Open Source Optimization of Dynamic Multidisciplinary Systems". Transcription methods in Dymos include Gauss-Lobatto Collocation and Radau Pseudospectral Method [7]. Two cases will be considered: without surface obstacles and with obstacles given by an inequality constraints on the trajectory. While there are many options to take into consideration the obstacles from the terrain [14], one simple way to implement a trajectory constraint is to set

$$r(\theta) \geq r_{\text{constr}}(\theta) = R + \frac{kh_{\text{min}}(R\theta)}{k(R\theta) + h_{\text{min}}} \quad (24)$$

This function has two defining properties: the slope  $k$  and the minimum height,  $h_{\text{min}}$

$$\left. \frac{dr_{\text{constr}}}{d(R\theta)} \right|_{\theta=0} = k \quad (25)$$

$$\lim_{\theta \rightarrow \infty} r_{\text{constr}} = R + h_{\text{min}} \quad (26)$$

The geometry of the constraint is shown in figure 3.

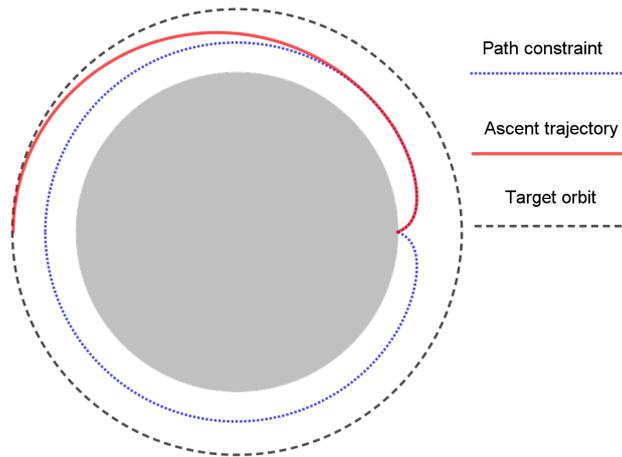


Figure 3: Geometry of the ascent trajectory with constraints (Not to scale)

Table 3: Trajectory discipline Input/Output

<b>Inputs</b>	Symbol	<b>Outputs</b>	Symbol
Thrust	$T$	Propellant mass	$M_p$
Total mass	$M_{\text{Total}}$		
Specific impulse	$I_{sp}$		

### 3.3 Structure

The purpose of this discipline is to evaluate the structural mass of the lander. Several models can be employed to model structural mass, among the simplest one is a fixed proportionality between the structure mass and the total mass of the system, usually called the structural efficiency  $\sigma$ . Typical values for structural efficiencies of Earth based launchers are

$\sigma \sim 10\%$  [17]. For lunar landers, different regression models are considered

$$M_{\text{Structure}}[t] = 0.0684 \left( m_{\text{dry}}[t]^{1.7851} F_D^{-0.0645} + m_{\text{payload}}[t]^{0.9062} \right) \left( \frac{H}{D} \right)^{0.6921} + 0.4528 \quad (27)$$

$$M_{\text{Structure}}[t] = 0.1882 \left( m_{\text{dry}}[t] \times F_D \right)^{0.2834} + 0.1055 \left( \frac{H}{D} \right)^{0.5179} + 7.0574 \times 10^{-4} v_H [m^3]^{1.1652} + 0.057 \quad (28)$$

$$M_{\text{Structure}}[kg] = 0.04928 M_{\text{Total}}[kg] + 390 \quad (29)$$

where  $v_H$  is the habitable volume,  $H/D$  is the height to diameter of the system,  $F_D = \frac{\rho_B}{1150 \text{ kg/m}^3}$  and  $\rho_B$  is the bulk density given by

$$\rho_B = \frac{1 + R_m}{\frac{1}{\rho_F} + \frac{R_m}{\rho_O}} \quad (30)$$

where  $\rho_F$  and  $\rho_O$  are the density of the fuel and oxidizer respectively. The regressions in equations 27, 28 and 29 can be found in [16], [16] and [19] respectively.

Table 4: Structure discipline Input/Output

Inputs	Symbol	Outputs	Symbol
Payload mass	$T$	Structure mass	$M_S$
Engine mass	$M_{\text{Engine}}$		
Tank mass	$M_t$		
Propellant mass	$M_p$		
Mixture ratio	$R_m$		
Habitable volume	$v_H$		
Geometric factors	-		

### 3.4 Objective function

In theory, the system should be optimized for the lowest possible cost while respecting the mission constraints. Although cost models have been considered before for launch vehicles [13], they carry larger uncertainties than engineering models, this is why surrogate variables are usually preferred. For Earth launchers, a common optimization metric is the total mass of the system  $M_{\text{Total}}$  also known as the gross liftoff weight (GLOW) for a fixed payload [3]. For reusable lunar landers, there seems to be no universally agreed upon metric. To identify a relevant metric, two limiting cases are considered: when the lunar lander is fully expendable, the metric should scale as the total mass of the system  $M_{\text{Total}}$  and considering an idealized case where the lander is infinitely many times reusable, then the cost should scale as the propellant mass used  $M_{\text{Propellant}}$ . In general, the metric is chosen to be the average mass used per mission, which is a linear combination of  $M_{\text{Total}}$  and  $M_{\text{Propellant}}$

$$f_{\text{Objective}} = \frac{M_{\text{Total}} + N_{\text{Reuse}} \times M_{\text{Propellant}}}{1 + N_{\text{Reuse}}} \quad (31)$$

where  $N_{\text{Reuse}}$  is the number of times the lander can be reused.

## 4. Results

### 4.1 Trajectory

The two most important parameters that influences the performance of an ascent/descent trajectory are the thrust to weight ratio (in lunar gravity) and the specific impulse of the vehicle. Without any constraints, the most efficient ascent trajectory is the one that grazes the lunar surface up to orbital speed followed by a Hohmann transfer to the correct height. While unreasonable trajectories in themselves, they still provide a lower bound for the  $\Delta v$  or fuel necessary. The fuel mass ratio as a function of initial thrust to weight ratio (TWR) and specific impulse is depicted in 4. When physical obstacles are taken into considerations, the  $\Delta v$  to reach orbit increases. The results of the optimal control simulation in Dymos is summarized in figure 5

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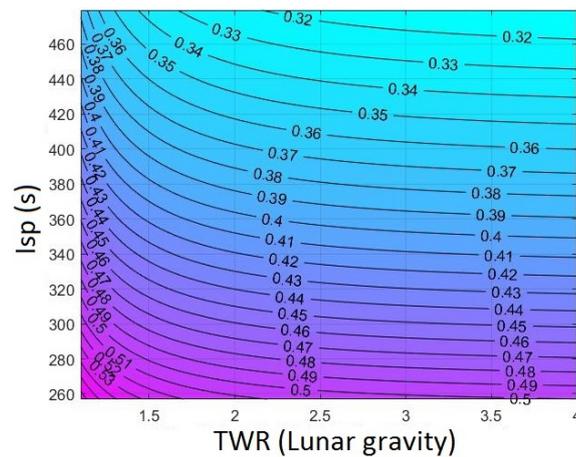


Figure 4: Fuel fraction necessary to perform an ascent trajectory without constraints

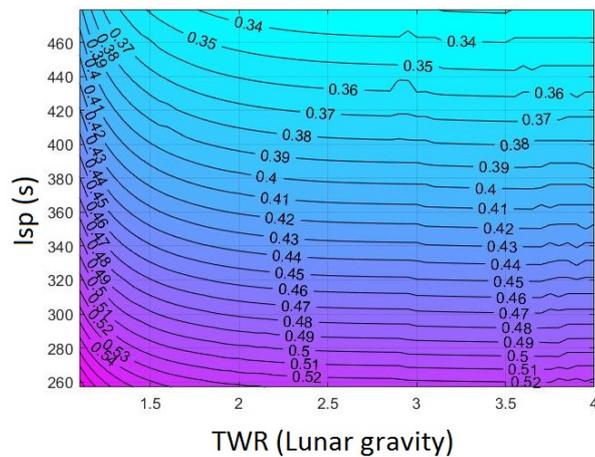


Figure 5: Fuel fraction necessary to perform an ascent trajectory with constraints

## 4.2 Mission analysis

The primary goal of the vehicle is to bring a payload from the NRHO to the lunar surface and back in orbit, schematically this is represented by figure 1. Even more abstractly, this mission architecture can be represented by figure 6, where "1" refers to the first (and only) vehicle. Up to now, only one-stage landers have been considered, however mission architectures with multiple stages can reduce the total mass of propellant used. The inclusion of multiple stages implies the non-trivial choice of the best mission architecture. An example of a two stages mission architecture is shown in figure 7, where double (or triple) lines indicates that two (or three) stages are moving together with the first number indicating which vehicle is providing the thrust. The number of possible mission architecture depends on the number of allowed transfers, for example, 5 transfers mean each craft can do a maximum of 5 transfers during the mission (including being ferried). A minimum of 4 transfers is necessary to perform the mission as one craft must at least perform the following sequence of transfers:

- 1) NRHO  $\rightarrow$  LLO
- 2) LLO  $\rightarrow$  Surface
- 3) Surface  $\rightarrow$  LLO
- 4) LLO  $\rightarrow$  NRHO

The number of possible mission architectures grows very rapidly with the number of allowed transfers as shown in table 5. However, the way the enumeration of valid mission architecture is done in this work leads to a large number of equivalent architectures which only differs by rearrangements. The mission architectures should be ranked with respect to an appropriate objective function (which will be described down below). However, the sheer size of the

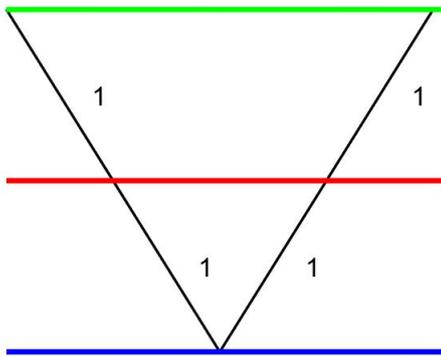


Figure 6: Representation of a one-stage lander mission

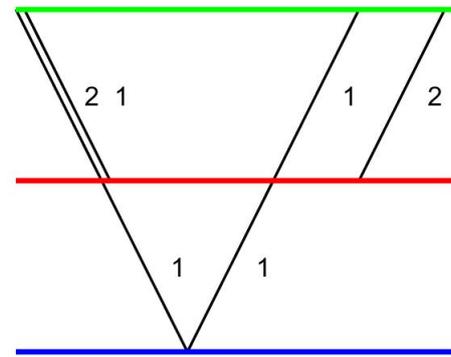


Figure 7: Example of a mission architecture with two stages

Table 5: Number of valid mission architectures

	2 stages	3 stages
# of transfer: 4	419	240,168
# of transfer: 5	4,539	6,749,884
# of transfer: 6	47,110	271,575,898

most numerous case considered (5 transfers, 3 stages) imposes the use of a simple and fast model so as to limit the computational time. A linear model is finally chosen as described here

$$M = P + T + F + S \quad (32)$$

$$T = \alpha F \quad (33)$$

$$F = M \left( 1 - e^{-\Delta v/w} \right) \quad (34)$$

$$S = \beta M \quad (35)$$

Where  $P, T, F, S, M$  are the payload mass, the fuel tank mass, the fuel mass, the structure mass and the total mass,  $\alpha, \beta, w$  are the tank mass to fuel mass ratio, the structural index and the effective exhaust velocity respectively. The advantage of such a simple model is that the whole system can be solved for analytically, the total mass is

$$M = \left( \frac{1}{(1 + \alpha)e^{-\Delta v/w} - \alpha - \beta} \right) P \quad (36)$$

As mentioned previously, to rank different mission architectures, an objective function must be introduced. The one chosen in this work is the mass necessary to bring a unit payload from the NRHO to the lunar surface and back. The mass used depends on whether a given vehicle is reusable or non-reusable, examples are shown in figure 8 and 9. For reusable vehicles, the mass penalty is given by the mass of propellant used for the mission, for non-reusable vehicles, the metric is the total mass of the vehicle. While this method attributes the same "cost" to bringing any type of mass to the NRHO and a more accurate model would give different weights to the type of material that is brought to orbit, it is reasonable, at first order, to assume the cost of bringing something to the NRHO is dominated by the launch cost. For this analysis, conservative values for  $\alpha, \beta$  and a methalox value for the effective exhaust velocity  $w$  were chosen

$$\alpha = 0.05 \quad (37)$$

$$\beta = 0.1 \quad (38)$$

$$w = 3.6 \text{ km/s} \quad (39)$$

The ordering of the best mission architectures are shown in figure 10 and 11, with the best architecture for a 2-stages and 3-stages architecture shown explicitly in figure 12 and 13. The lowest achievable mass fraction for different vehicle numbers are given in table 6. While the ordering of mission architectures depends on the values of  $\alpha, \beta$  and  $w$ ,

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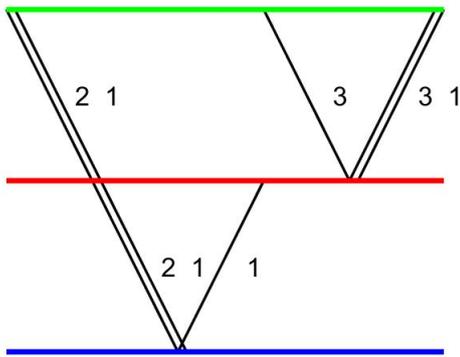


Figure 8: Non-reusable architecture

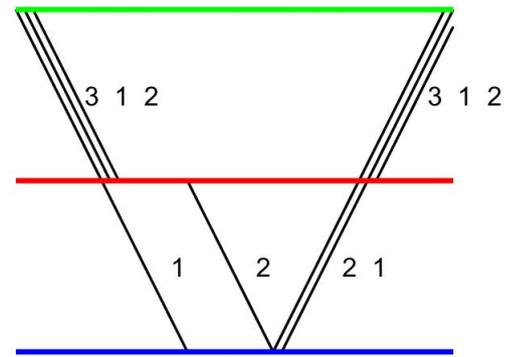


Figure 9: Reusable architecture

Table 6: Best mission architectures

	Normalized mass used per mission
<b>1 stage</b>	9.217
<b>2 stages</b>	5.096
<b>3 stages</b>	4.522
<b>Theoretical minimum (<math>\alpha = \beta = 0</math>)</b>	3.482

it is expected that there is no dramatic change in the optimal ordering if these coefficients are changed. The analysis can be performed again for a different set of constants  $\alpha, \beta, w$  but only considering the  $N$  best scenarios found for the original set of parameters. This process essentially allows the "weeding out" of millions of unreasonable mission architecture. A more thorough study can then be done on the set of optimal mission architecture.

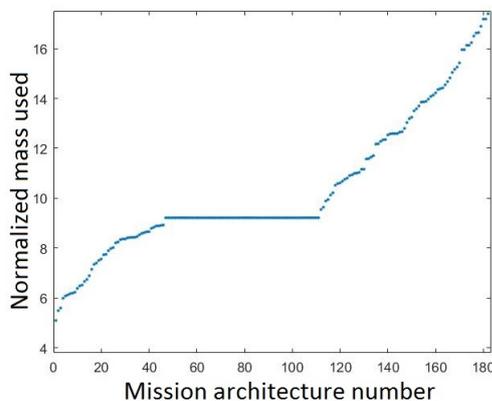


Figure 10: Best 180 mission profile for 2 stage architecture

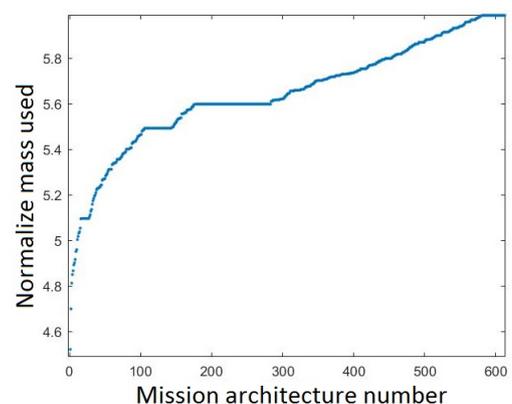


Figure 11: Best 630 mission profile for 3 stage architecture

### 4.3 Sizing

Lockheed Martin (LMT) has recently released the design of a one stage LH2/LOX reusable lunar lander [5]. The MDO tool developed in this work is used to size a one stage LH2/LOX lunar lander to compare to the LMT design. Since the LMT lander is planned to be used approximately 10 times, the objective function of the MDO will consider  $N_{\text{Reuse}} = 10$ . The payload of the lander is chosen to be similar to the mass of the Orion capsule plus one ton[15]. The engine chamber pressure and expansion ratio is set to the RL-10 engines[4]. The optimization variables are the thrust and mixture ratio. After convergence, a preliminary design is obtained. Table 7 summarizes the results and compares them with the Lockheed Martin lander.

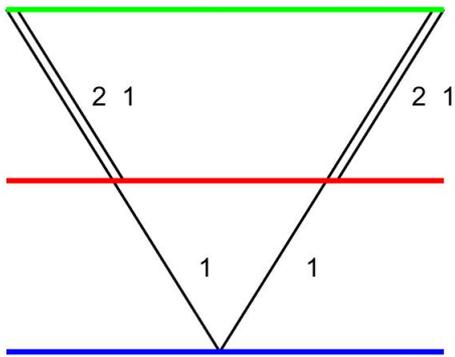


Figure 12: Best 2 stage architecture  
(cost: 5.096)

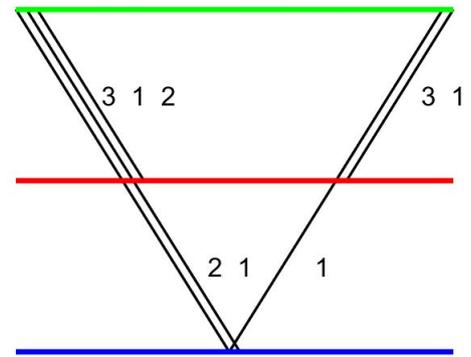


Figure 13: Best 3 stage architecture  
(cost: 4.522)

Table 7: Single stage, LH2/LOX lunar lander

Type	Parameter	Symbol	Value	Units	LMT Lander
Objective	Average mass used	$f_{\text{Objective}}$	40.64	t	42
Fixed	Payload	$M_{\text{Payload}}$	12.0	t	-
Fixed	Combustion pressure	$P_c$	24.0	bar	-
Fixed	Expansion ratio	$\epsilon$	40.0	-	-
Optimized	Thrust	$T$	230.2	kN	266.8
Optimized	Mixture ratio	$R_m$	5.17	-	-
State	Total mass	$M_{\text{Total}}$	56.92	t	62
State	Thrust to weight ratio	TWR	2.27	-	2.66
State	Propellant mass	$M_P$	39.01	t	40
State	Structure mass	$M_S$	3.195	t	-
State	Tank mass	$M_t$	1.570	t	-
State	Engine mass	$M_{\text{Engine}}$	1.148	t	-
State	Dry mass	$M_{\text{Dry}}$	17.91	t	22
State	Velocity increment	$\Delta v$	5148	m/s	5000
State	Specific impulse	$I_{sp}$	453.9	s	450+
State	Combustion temperature	$T_c$	3235	K	-

## 5. Conclusion

A tool to obtain preliminary designs of lunar landers was developed based on techniques of multidisciplinary design optimization. The three disciplines of propulsion, trajectory and structure were discussed and several models were presented. To compare with the existing Lockheed Martin design, an optimization with respect to the thrust  $T$  and mixture ratio  $R_m$  was performed on a one stage LH2/LOX lunar lander, the comparison can be seen in table 7.

An exhaustive search for the best mission architecture was done in parallel to the MDO using a simpler model to limit the computational time. Scenarios were ordered by the amount of mass necessary to perform the mission. Scenarios for 2 and 3 stages architecture using the least amount of mass were identified. While the optimality of the scenario depends on the coefficient  $\alpha, \beta$  and  $w$  of the model, a more thorough analysis can be done on the best mission architecture obtained so far.

Future work will include higher fidelity models, the addition of more disciplines such as the reaction control system, fuel feed system (turbopump, pressurization system), cooling system, three dimensional ascent/descent trajectories,

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more accurate gravity models and more accurate vehicle body dynamics, among others. In this work, the optimization of the mission architecture and the vehicle itself were done independently, the most significant step forward will be the combining of the mission architecture with the multidisciplinary design optimization. To this end, multi-stage systems with a framework general enough to include any mission architecture will have to be implemented in OpenMDAO.

## 6. Acknowledgments

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