Tool estimating the mass and sizing the structure of a reusable lunar lander

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March 31, 2020

Abstract

This research project aims at providing a tool computing the mass of a lunar lander and sizing its structure. With the constraint of using only open-source softwares. The paper first focuses on the design of the lander, and then, describes the tool that has been developed by implementing several studies made in the field of launchers and landers. The different results are analysed, mass results are compared to already existing designs, in order to validate these results and the tool. In conclusion, basic computations using Python language appears to be sufficient for the preliminary mass estimation of a reusable lander. However, the optimization of the structure using Nastran95 appears to be limited. An external optimizer using the Nastran95 outputs would be a solution.

Keyword : Reusable lunar launcher, Fuel estimation, Mass estimation, Structure sizing, Open-source

Glossary

I_{sp} Specific Impulse. 5

FPR Flight Performance Reserve. 6

LH2 Liquid Hydrogen. 3

LLO Low Lunar Orbit. 6

LOP-G Lunar Orbital Platform-Gateway. 1

LOX Liquid Oxygen. 3

MDO Multidisciplinary Design Optimization. 1

NRO Near Rectilinear Orbit. 6

SoA State of the Art. 3

Introduction

The U.S. (with NASA) wants to go back to the moon and create a Lunar Orbital Platform-Gateway (LOP-G). Which can be considered as a new International Space Station but around the Moon. In this context, supplies will have to be transferred by a vehicle acting as a shuttle between the surface of the Moon and this Space Station. The objective of the global project is to design a reusable launcher capable of doing this round trip with a given payload. All the subsystems; trajectory, propulsion, structure, etc, of the vehicle must be optimized to obtain the best performances. Such a problem is best handled by the formalism of multidisciplinary design optimization (MDO) for which OpenMDAO has been designed for.

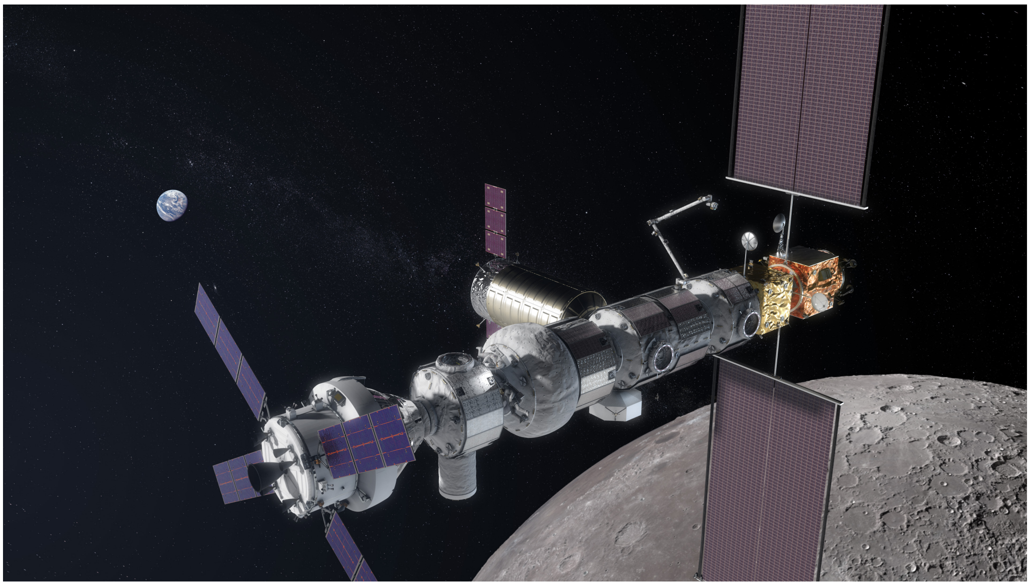

Figure 1: Lunar Orbital Platform-Gateway

Multidisciplinary Design Optimisation allows to optimise a whole system thanks to connection and feedback loops between the subsystems and their components. The connections and the structure of the MDO tool for the reusable launcher are found in the figure below from L. Beauregard.

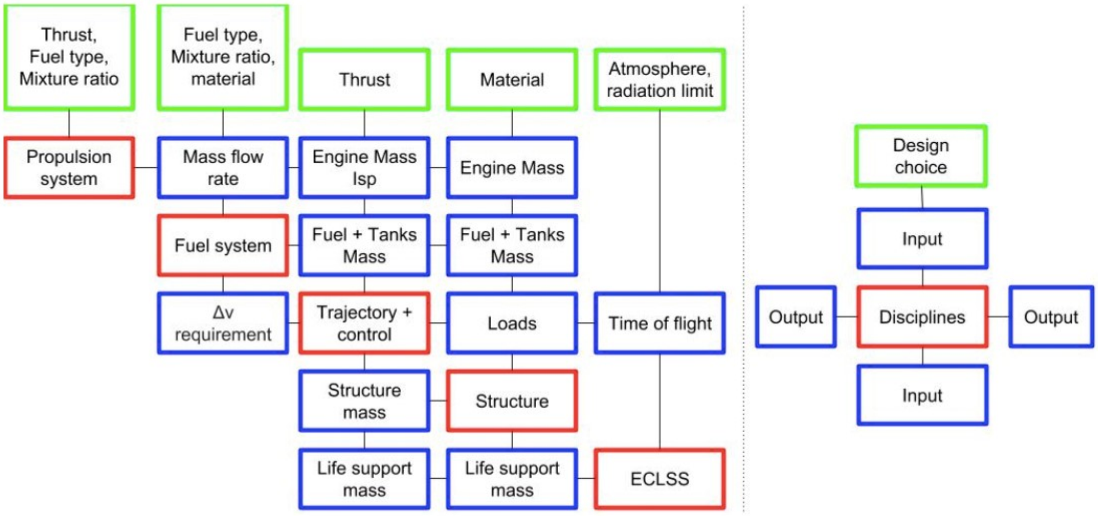

Figure 2: MDO structure of the whole project

The research project you are currently reading the paper from is focused on providing a tool relative to the structure "block". That is to say the tool will have to compute the dimensions and the mass of the structure. As you can see on Figure 2, this structure "block" has 4 inputs (inputs are on the vertical axis). Engine mass, life support mass (as well as the payload) will all be considered as a blackbox with one mass value. This blackbox will be entered in the tool as an input as well as the loads. However, fuel and tanks mass will be computed within the tool.

First, the paper will present the research done about the design of landers. In a second part, it will present the fuel and structure mass computations, with the results. Then, will be explained the sizing of the lander and its optimization. And finally, will be presented the conclusion and the perspectives of this project.

1 Design

1.1 State of the Art

You can see on Figure 3[1] the basic shape of a lunar lander, this lander is a conceptual model designed by NASA. It will be seen in this State of the Art (SoA) that every lander looks quite the same, except from some specificities. This SoA will help in the design choices that will have to be made.

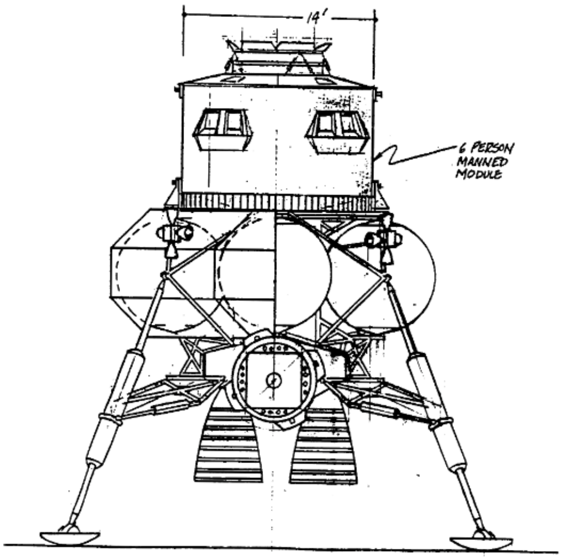

Figure 3: Reusable Lunar Lander Concept from NASA

1.1.1 NASA - Apollo Lander Module

The Apollo module, used by the USA to land astronauts on the moon, was the first of its kind and paved the way for future planetary and lunar lander. Indeed, the lander from NASA presented a basic configuration and the landers designed today are very similar. The lunar module used 4 landing legs with a truss structure. The damper where made of honeycomb crushable absorbers, because it was landing only once. The lander/launcher was made of two separate stages, one landing the whole structure and staying there, the other launching the astronauts back to their spacecraft. The fuel used was the couple N2O4/Aerozine 50. Speaking of the mass, the module had a dry mass of 2,000 kg and a total mass of 15,000 kg.

1.1.2 Lockheed Martin - Lunar lander

The lander, presented by Lockheed Martin during the 69th International Astronautical Congress[2], is one of the most recent reusable lander design existing. With a dry mass of 22 metric tons and a launch mass of 62 metric tons, this module is capable of transporting a 1,000 kg payload. The couple Liquid Oxygen/Liquid Hydrogen (LOX/LH2) has been chosen for propulsion, and this for two main reasons. According to them, it is the only fuel couple that allows a round trip for a single stage lander/launcher operating from the gateway orbit thanks to its high I_{sp}. Indeed, a single stage lander means full reusability, which is a big asset for regular operations. Although LOX/CH4 could also work for a single stage lander, it would imply Research and Development as "the mass fraction required implies very lightweight system", which is not preferable. The second reason to choose LOX/LH2 is that it can be made out of water, and water is known to be present on the moon. Therefore, it would allow in situ production of fuel for the lander. Concerning the structure, it has as well 4 legs but each made of one main strut and one secondary one. 4 thrusters are needed to propulse this heavy lander.

A new design in 2019 has been presented. It follows the new deadline for the American return on the moon, set by the government. In order to reduce the development time this new lander is composed of two stages and is not reusable.

1.1.3 Blue Origin - Blue Moon

The main particularity of this lander, designed by Blue Origin (founded by Jeff Bezos), is its adaptability. Indeed, the lander is strictly a lander, it does not have a pressurized module or anything like this. It has a platform where customers can put whatever payload they want on. By doing that, Blue Origin made its lander very versatile. As the previous lander from Lockheed Martin, Blue Origin chose to use LOX/LH2 fuel and designed its lander with 4 landing legs. However the "Blue Moon" is not reusable and will only land once.

1.2 Design choices

Thanks to this SoA several design choices can already be made. It is clear that LOX/LH2 is the best choice for fuel. As Lockheed Martin explained for their lander, it's one of the only propellant allowing for a single-stage lander/launcher to perform a round trip to the moon from the LOP-G. It has also proved its worth, and as it has been widely used, the design of the propulsion system won't be costly. For the structure, 4 legs with a truss configuration seems to be the best choice, it offers a great stability for the landing phase.

Regarding materials, a lot can be used. Each having its specificities. The most common one is aluminium, followed by steel. Aluminium exhibits a good strength-to-mass ratio and the large variety of alloys allows for a large variety of applications.
Nevertheless, steel is used when higher strength and/or temperature resistance are required. Stainless steel is the main choice among all the types of steel because of its chemical properties. It does not rust or corrode, it is preferable during manufacturing and test phases.

After reviewing the only manned lunar lander and the current usage of aluminium alloy in space structures, it has been decided that Aluminium 7075-T6 will be used for the structure of our lander, like on the Apollo module[3]. And Aluminium 2219-T851 will be used for the pressurized module, as for the space shuttle fuselage and other spacecraft project.[4].

2 Mass estimation

For the mass computation it has been decided to use a Python code. It is widely used and it allows to use batch processing for the next phase concerning structure dimensions. On the figure below an overview of the mass calculation loop.

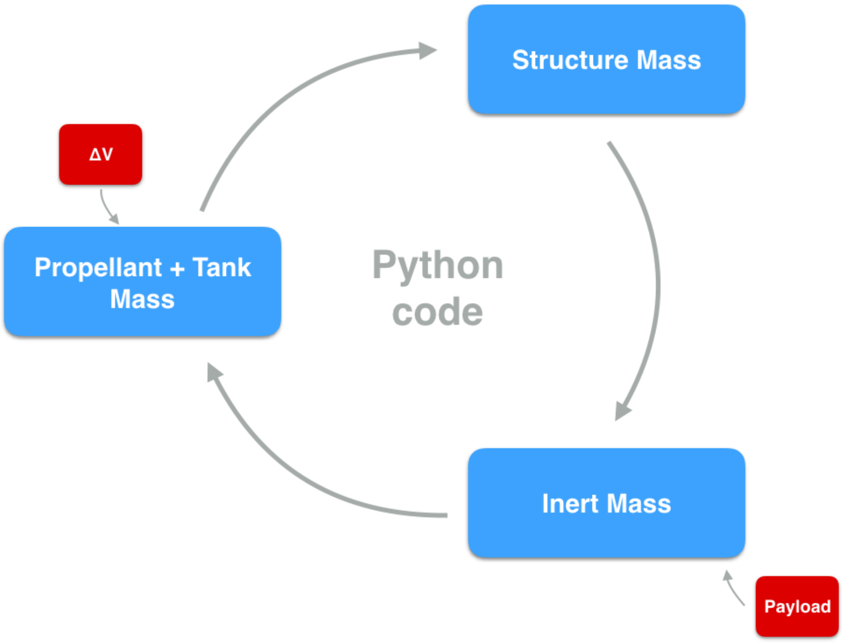

Figure 4: Scheme of how the mass calculus works

The mass estimation consists of a loop computing the fuel mass and the structure mass needed for a specific ΔV and a specific payload. These values are entered as inputs by the user. The computation works as follow : for the first loop the code put the dry mass (which is the total mass without fuel in this study) equal to the payload, because it is the only known mass at this moment. Then the amount of fuel needed to propel this dry mass is computed as well as the mass of the tanks containing it. After that, the mass of the structure needed to support all of this is computed. But the structure is part of the dry mass, hence a new dry mass is obtained, needing a new amount of fuel to propel it, and this new overall mass needs a new structure and so on.
After a few iterations the total mass converges to a certain value. The loop stops when the new total mass value is less than 1 kg different from the previous one.

The subsection below provides a detailed explanation of the calculations.

2.1 Fuel and tanks mass computation

2.1.1 Rocket equation

Propellants represent a very big proportion of launchers' mass. It is crucial to have a good estimation. By using the rocket equation (1)[5], we can retrieve the propellant mass of the spacecraft, depending on mission maneuvers.

$$\Delta v = I_{sp} g_0 \ln\left(\frac{M_{tot}}{M_{dry}}\right) \longrightarrow \frac{M_{tot}}{M_{dry}} = \exp\left(\frac{\Delta v}{I_{sp} g_0}\right) \tag{1}$$

As you can see in these equations the inputs would be I_{sp} and Δv, the first depends on the propulsion chosen, and the second depends on the maneuvers the spacecraft needs to achieve. The values can be found in Table 1. The launcher will have to go from the moon surface to the Low Lunar

Orbit and then to a Near Rectilinear Orbit (NRO) where the LOP-G is. This corresponds to a ΔV of 5,100 m/s.

Table 1: Δv for lunar mission [6]

Mission phase	Δv km/s
LLO to moon surface	1.9
Moon surface to LLO	1.8
To/from LLO from/to NRO	0.7

With this, is retrieved the ratio $\frac{M_{tot}}{M_{dry}}$. Then, by multiplying the dry mass to this ratio is obtained the total mass. And finally, by subtracting the dry mass from the total one, is obtained the fuel needed.

These first computations are as follow in the python code.

Listing 1: Rocket equation code

```
# rocket equation
R = exp(deltav / (I_sp * g))

M_tot = M_dry * R
M_prop = M_tot - M_dry

M_prop = M_prop * 1.07 #4% for FPR Propellant, 3% for Unusable Propellant
```

The last line adds propellant for Flight Performance Reserve (FPR) and for unusable propellant. The reason is because you cannot fly with only the theoretical amount of fuel. You need more in order to be sure you will not be out of fuel and to have precise changes of trajectory.

2.1.2 Tanks sizing and mass estimation

In this part, equations from a University of Maryland's paper[7] have been applied, this paper gathers several works from various authors.

With these equations are found the mass of LH2 and LOX tanks, as well as the mass of their insulation and their radius.

Listing 2: Tanks mass code

```
# mixture ratio LOX/LH2, 6:1
M_LH2 = M_prop / 7
M_LOX = 6 * (M_prop / 7)
M_LH2_1 = M_LH2 / 2  # 2 tanks for mass distribution
M_LOX_1 = M_LOX / 2  # 2 tanks for mass distribution

# LOX Tank
M_LOX_Tank = 0.00152 * M_LOX_1 + 318
V_LOX_Tank = M_LOX_1 / rho_LOX
r_LOX_Tank = (V_LOX_Tank / (4 * pi /3))**(1 / 3)  # radius of LOX tank
A_LOX_Tank = 4 * pi * (r_LOX_Tank**2)  # Area LOX Tank
M_LOX_Insu = 1.123 * A_LOX_Tank  # Mass LOX insulation
M_LOX_Tanks = 2 * M_LOX_Tank
M_LOX_Insus = 2 * M_LOX_Insu

# LH2 Tank
M_LH2_Tank = 0.0694 * M_LH2_1 + 363
V_LH2_Tank = M_LH2_1 / rho_LH2
r_LH2_Tank = (V_LH2_Tank / (4 * pi  /3))**(1 / 3)  # radius of LH2 tank
```

```
A_LH2_Tank = 4 * pi * (r_LH2_Tank**2)  # Area LH2 Tank
M_LH2_Insu = 2.88 * A_LH2_Tank  # Mass LH2 insulation
M_LH2_Tanks = 2 * M_LH2_Tank
M_LH2_Insus = 2 * M_LH2_Insu
```

2.2 Structural mass estimation

In this tool, the structural mass percentage follows a linear curve that decreases. This is due to the mass efficiency that improves while the total mass increases. This linear curve is function of the sum of the masses of the fuel, the tanks and the blackbox. This curve has been built empirically, based on the mass budget of existing lunar landers. The empirical method is the main method used in the space industry to estimate the mass budget of a future spacecraft during preliminary phases. Not many landers have been built at the time I am writing, the curve could be improve with more data in the future.

Listing 3: Calculus of the structure mass

```
def structure(M_):

y = -M_*5.92E-07 + 0.081

M_Struct = M_ * y

return M_Struct
```

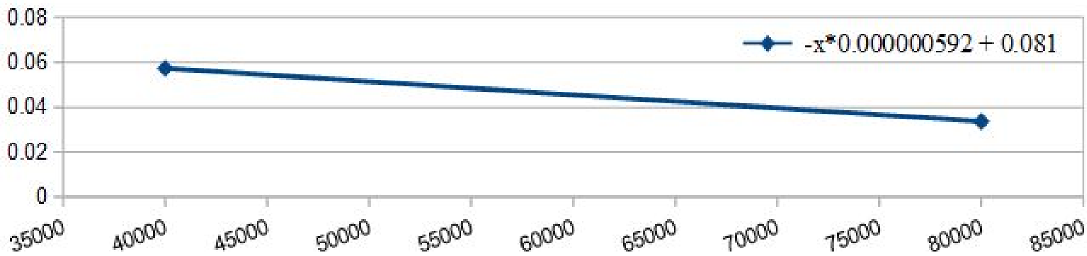

Figure 5: Evolution of the percentage of the structural mass

2.3 Results

In order to validate the output results of the tool, masses and mission parameters from two existing reusable landers have been used as inputs. The first one is a conceptual reusable lander from NASA. With a total ΔV for the mission of 4,380 m/s. The blackbox, which is the second input in the tool, has been computed as follow thanks to the detailed mass budget : $Blackbox = Payload + Inert\ Mass - Structure - Landing\ Syst. - Tanks$. Which gives a blackbox of 10,333 kg. With these two values the tool is able to provide a mass estimation. In the table below are found the results from the tool besides the actual mass budget of the lander.

Table 2: Validation results

	NASA concept (kg)	Code results (kg)
Dry Mass	15,823	14,561
Structure	2,465	2,314
Tanks Mass	3,025	1,914
Blackbox	10,333	10,333
Prop. Mass	32,395	28,413
Total Mass	48,218	42,974

As it can be seen, all the values from the code are slightly lighter, which in the end gives a total mass lighter than the NASA one. With a difference of 12%.
It can be explained by several things. First, the tanks in our model are strictly sized for the amount of fuel needed. Which is not the case in real life where the tanks are bigger than needed in order to be filled more or less depending on the mission.
Also, the concept from NASA was made in 1988, we can assume that in over 30 years some improvements have been made in the field.

A second example has been used to verify and validate the code: the Lockheed Martin lander. The ΔV is known, 5,000 m/s, and was used for input. However, the detailed mass budget is not provided by Lockheed Martin, only the dry and total mass is given, which are respectively 22,000 kg and 62,000 kg. Therefore, in order to verify the code, arbitrarily chosen blackboxes were entered in the code, until was reached the dry mass. When the dry mass was finally reached, the total masses could be compared. The tool provided a total mass of 70,000 kg. Which is 8,000 kg heavier than the real one. These extra kilograms found by your code are propellants.

To conclude on this part, the tool provides results with some differences from the expected and real values. But the goal is to provide an estimation of the mass. Furthermore, the specificities of the launchers we can compare with are not known. Hence, differences of 12.1% and 11.5%, for two very different landers, are acceptable results for an estimation.

3 Structural analysis and optimization

After computing the mass of the lander, another goal of the tool is to analyse and optimize its structure. Thanks to the optimization, the tool provides an estimation of the structural elements' dimensions and also gives a second estimation of the structural mass. To do so, two open-source softwares are used : GMSH, for the modelization and nastran95, for the analysis.

3.1 Modelization and meshing

The model of the lander presents 4 legs with a truss structure and a pressurized payload, in accordance with the design choices made after the State of the Art. It uses 1D and 2D elements only. Indeed, a lander is basically made of bars, beams and shells. 3D elements would just increase the analysis time and add no value.

Two models have been built, one for visualization, where the architecture can be seen with the tanks. And one for the analysis, where tanks are replaced by bars. This approximation comes from the fact that tanks are very complex structures and it does not fall into the scope of a preliminary analysis. Only the mass of the filled tanks and their fixations on the lander are needed. These two models are found in Figure 6 thereafter.

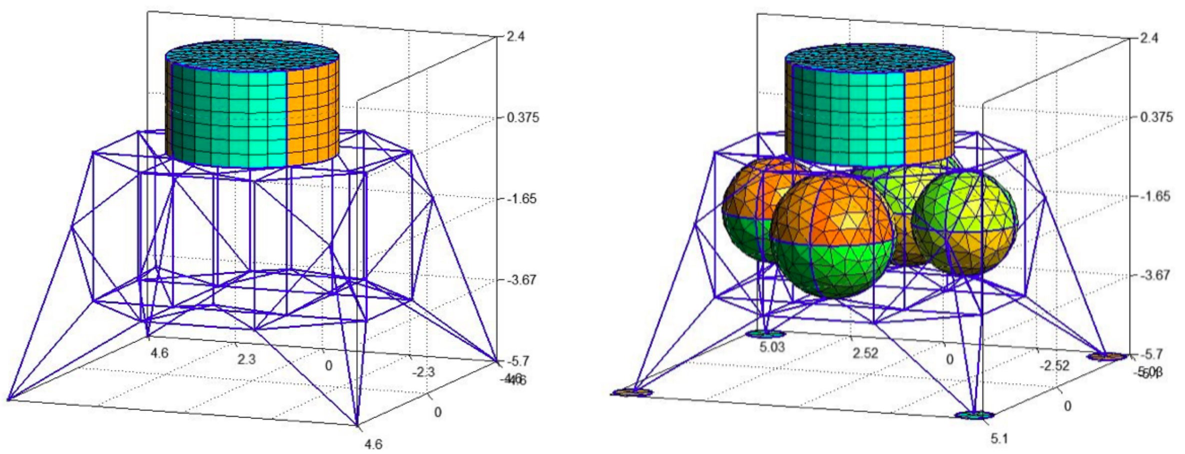

Figure 6: From left to right: Analysis model and Visualization model (scale in meter)

3.2 Structural analysis

A mesh file is retrieved from the GMSH model and is used for analysis. When the tool is run and mass calculation is over, it writes new input values in the mesh file, along with analysis parameters (type of analysis, boundary conditions, materials, etc...). The input values are the payload mass, the tanks mass (with the propellant) and their radius, and the g-force entered by the user.

It has been decided to run a static analysis, because quasi-static loads are the main used in first preliminary designs of spacecrafts[8][9]. Low frequency sinusoidal loads can also be applied and analysed with few modifications in the code.

Two subcases are used in order to analyse the lander properly. These subcases correspond to the launch phase from the Earth and to the landing on the Moon. It is where the biggest stresses are found.

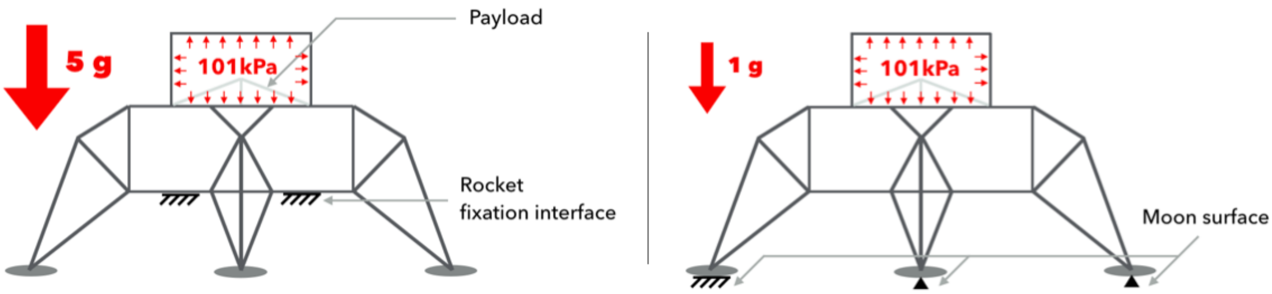

Figure 7: From left to right: Launching phase and Landing phase

The g-force is up to the user, and is chosen prior to the analysis.

The optimization is done thanks to the nastran function *POPT*, which uses the fully stressed design method. This method consists of making the stress tend towards the set stress limits by modifying geometrical properties of elements. Therefore, the stress limits entered in the tool are the yield stresses of the materials divided by the Factor of Safety. The Factors of Safety have been chosen taking into account NASA's standard[10] for human spaceflight.

Table 3: Stress limits

	Yield Stress (MPa)	Factor of Safety	Stress limit (MPa)
Aluminium 7075-T6	503	1.25	402
Aluminium 2219-T851	352	1.5	235

3.3 Results

To analyse the results of the optimization, the conceptual lander from NASA was again used as a standard with which to compare. The structural mass will also be compared with the value retrieved by the tool. For this analysis, an acceleration of 4 g has been used (corresponding to the g-force experienced in a Saturn V rocket) for the earth launching phase, and 1 g for the landing phase.

During the first loop, the structural mass comes from the initial elements' dimensions already implemented in the file. This mass is 5,618 kg, which is twice more than the actual one from NASA and from our tool. But after optimization, this mass drops to 2,282 kg, which is about 32 kg lighter than the tool's value.
This result, with a difference of less than 1.4% with the computed value and 9% with NASA's mass value, validates the geometric model of the lunar lander.

To verify the sizing results, other inputs were used. These inputs correspond to the openMDAO project (i.e. a blackbox of 10,000 kg and a ΔV of 5,000 m/s). Only few examples will be presented. One can dive into the output file of nastran and view all the results.

For instance, the main landing legs, with an initial surface of 0.0028 m2, has a surface of 0.00045 m2 after optimization. Other elements of the landing legs have bigger surfaces after optimization.
Another example: the horizontal beams supporting the payload. The properties, before and after optimization, are found in the table thereafter.

Table 4: Horizontal beam optimization

	surface (m2)	Iz (m4)	Iy (m4)	J (m4)
initial geometry	0.0046	1.348E-5	6.953E-6	1.381E-5
optimized geometry	0.00157	4.604E-6	2.375E-6	4.717E-6

Regarding the stress results, the values are all under the limits. However, because the *POPT* function does not provide control over the displacement, a few displacements between 1 and 2 decimeters are found.

4 Conclusion and perspectives

This project has fulfilled its objectives: providing a tool that estimates the mass of a lunar lander and that provides structural elements' dimensions. Of course, this tool does not provide an exact value of the mass of a lander, as it was seen when compared to actual lander. It provides an estimation, which can help during preliminary design phases to anticipate future constraints or issues. The tool can be implemented in the openMDAO project for which it has been built.

However, a criticism against this tool can be made : its versatility. Indeed, if one wants to make changes in the geometric model, he will have to change a lot of things in the code and follow several steps, described in a tutorial provided with the tool. These limitations in terms of modifications are due to GMSH, which does not provide a wide control over the mesh file. For instance it only gives CBARs, so if one wants CRODs in his model he will have to do it manually or use the corrector file (provided as well, as a template) with some modifications. These limitations also come from the optimization function of Nastran which is not compatible with the AUTOSPC function, which is used to avoid singularities in a FEM model. Nevertheless, if no modifications in the geometric model are made, these issues are not met and the tool can be used easily.

Finally, an improvement related to the accuracy is possible. The optimization part could be more accurate with an external optimizer. It would be better in terms of results, because the optimization function of Nastran is more than 40 years old and is only compatible with old elements : CTRIA2, CQUAD2. These elements are less accurate than the new CTRIA3 and CQUAD4. Furthermore, an external optimizer would provide a greater control over the optimization. For example, displacement limitations could be implemented, which is not possible with the *POPT* function.

All the documentation along with the tool are available following this link :
$https://github.com/mid2SUPAERO/RP\_MAE\_Arthur\_GUY$

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