Design of a sample return vehicle

Integrated Project Team

Team D

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Introduction

"Why should we return to the Moon?" That is the question asked by NASA to thousands of people. With the answers they gathered, NASA and thirteen world's space agencies are working together to develop a Global Exploration Strategy.

They found six themes that could answer this question. The first three ones are as follows: global partnerships, economic expansion and public engagement. The lunar exploration could create partnerships with other countries that have a common goal. Then, we have the three other themes: human civilization, exploration preparation and scientific knowledge.

First, going back to the Moon could help the scientists to answer fundamental questions about the Earth, Solar System and universe formation. In fact, the interior of the Moon retains a record of the initial stages of planetary evolution. Then, its crust has never been altered by plate tectonics like on Earth or planet wide volcanism like on Venus. The last time that moon samples got back to Earth was in 1976 with the Luna 24 mission. Therefore, the space agencies want to return to the Moon to have more information about the Solar System than they already have. Secondly, Moon exploration would be a guinea pig to test new technologies, new techniques to reduce the risks for the future missions to Mars or other planets. Thirdly, even if the Moon does not have an atmosphere like the Earth, we could produce the oxygen thanks to the Moon water (Northrop's mission planned before 2018) to supply with oxygen a permanent scientific lab.

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Presentation

I. Global description

The project goal is to develop a Lunar Lander mission for NASA to conduct different investigations at multiple sites on the Moon surface. Thanks to the preliminary requirements in the Concept Description Document (CDD), we have to design a Lunar Exploration Transportation System (LETS) that will go to the Moon on board of an Atlas V-4 launcher. After landing on the Moon, a rover will explore Moon surface and collect regolith samples. These samples will go back to Earth in order to be analyzed. To develop this mission, we are separated into three teams as follows: one team of nine students at University of Alabama in Huntsville (UAH) is responsible for the lunar lander design, a two-people team at Southern University is responsible for the mobility concept (rovers) and another two-people team at ESTACA will design the Return Sample Vehicle.

II. ESTACA Contribution

As written above, we have to design the vehicle that will bring the regolith samples safely back to Earth. For the mission to succeed, we have many constraints to respect. These are gathered in three main categories: performance, environments and interfaces. It means the SRV has to have a propulsion system able to escape the Moon velocity and an optimized trajectory in order to minimize the fuel quantity and then the fuel mass. This fuel economy will permit to deliver more than one kilogram of lunar regolith back to Earth. Then, the SRV must withstand the environment conditions anywhere in space, lunar surface or Earth and in any situation like launch, landing or atmospheric reentry. The SRV shall be designed to keep the regolith samples viable; we have to provide means to sustain samples as in lunar condition. Finally, during the entire mission, the SRV has to be designed in relation to the LETS and the rover in order to fit in the Atlas V-4 launcher. To see the entire preliminary requirements, you can check the appendix p.40.

Performance

I. Trajectory

The determination of the SRV trajectory is vital in order to develop our project. Indeed, the distance the module covers, influences the fuel consumption. As the SRV is going to pass from an orbit to another, the engine has to provide an impulsion that will depend on the change in velocity we need.
Before describing the trajectory in details, we will do a recap about the space mechanics.

I.1.Space mechanics

I.1.1. Planets characteristics

- Gravitational constant

Denoted G, this is a physical constant involved in the calculation of the gravitational attraction between two objects with mass. It appears in Newton's law of universal gravitation, which is $F = G \frac{m_1 m_2}{r^2}$.

It says that the attractive force F between two bodies is proportional to the product of their mass and inversely proportional to the square of the distance between them.

$G = 6{,}674.10^{-11}\ m^3\ kg^{-1}\ s^{-2}$

- Standard gravitational parameter

The standard gravitational parameter μ of a celestial body is the product of the gravitational constant G and the mass M.

$$\boxed{\mu = G\,M} \quad in\ m^3\ s^{-2}$$

We use this parameter to get the calculations easier.

- Local gravitational field

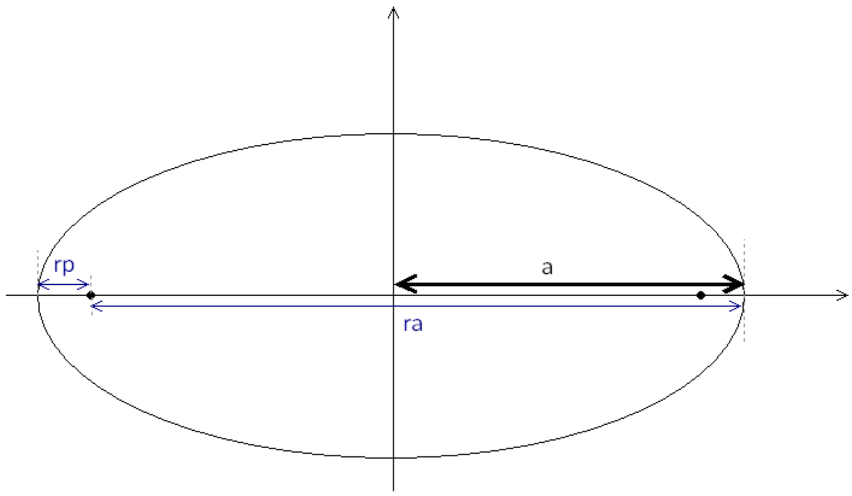
Denoted g, it is the local acceleration due to gravity. This value depends on the celestial body. For example, on the Earth, this "standard gravity" depends on one's position on Earth. However, for our project, we will consider this acceleration as a constant.

- Planet characteristics

	Earth	Moon
Radius (m)	6378135	3474600
Mass (kg)	5,97E+24	7,34E+22
µ (m3/s2)	3,987E+14	4,90E+12
g (m/s2)	9,801	1,623
rl (m)	[redacted]	57595462,5

I.1.2.Elliptical orbit

Given an ellipse, with two centers
It has five characteristics as follows: apogee, perigee, eccentricity, semi major axis and parameter of the ellipse.



The apogee, denoted r_a, is the furthest point from the main centre. On the contrary, the perigee (r_p) is the closest point.

The eccentricity, denoted e, of an ellipse is always inferior to one. Here is the relation between the eccentricity and r_a and r_p:

$$e = \frac{r_a - r_p}{r_a + r_p}$$

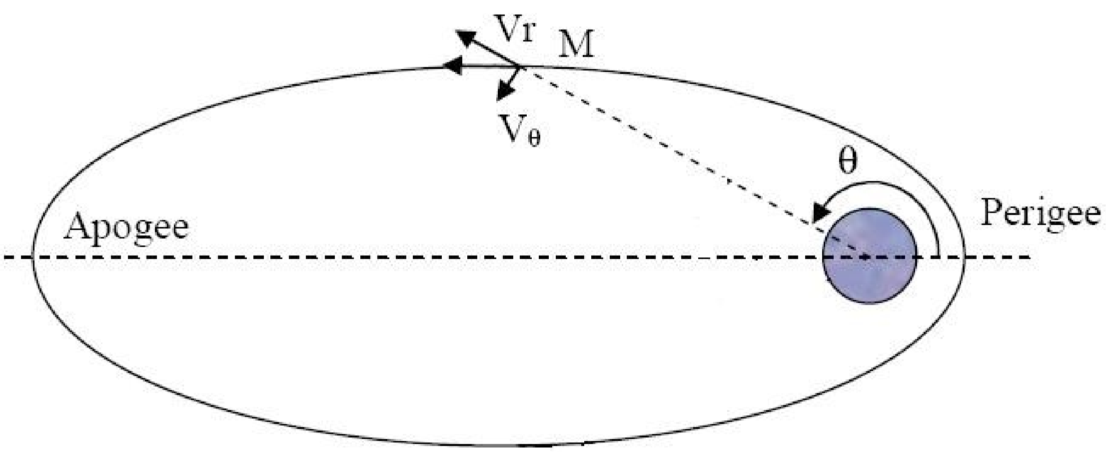
We also define a parameter written as follows: $p = \frac{2\, r_a r_p}{r_a + r_p}$

The characteristic equation of an ellipsis is: $r = \frac{p}{1 + e\, \cos\theta}$

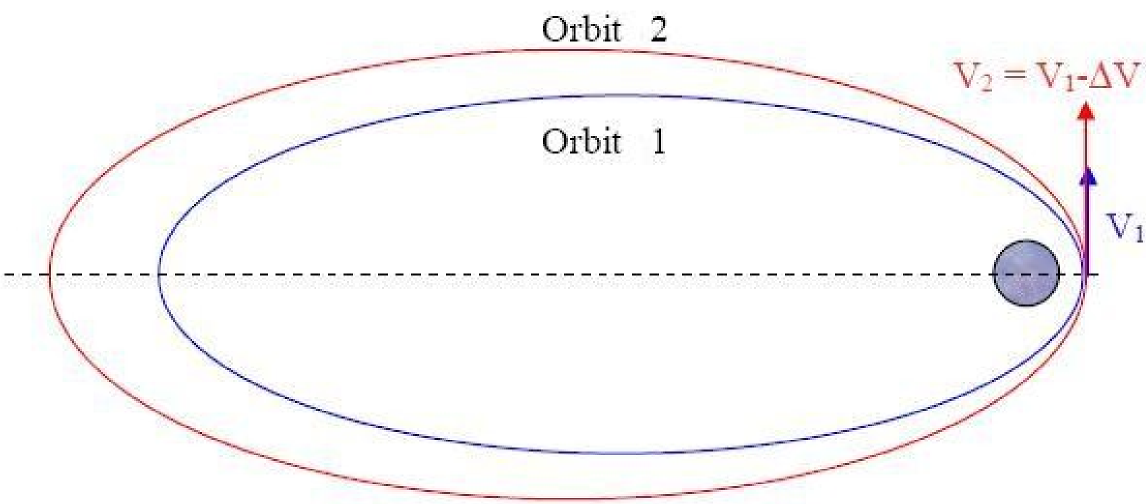
If we project on the horizontal and vertical axis, we find: $\begin{cases} r_x = r\ \cos\theta \\ r_y = r\ \sin\theta \end{cases}$

We can find the speed of each point of an ellipse thanks to the following formula: $\boxed{V = \sqrt{\mu(\frac{2}{r} - \frac{1}{a})}}$

Projecting on the two axes, we obtain: $\begin{cases} V_r = -\sqrt{p\ \mu}\ (e\ \sin\theta) \\ V_\theta = \sqrt{p\ \mu}\ (1 - e\ \cos\theta) \end{cases}$



To pass from an orbit 1 to an orbit 2, without changing the perigee, we have to provide a change in velocity.

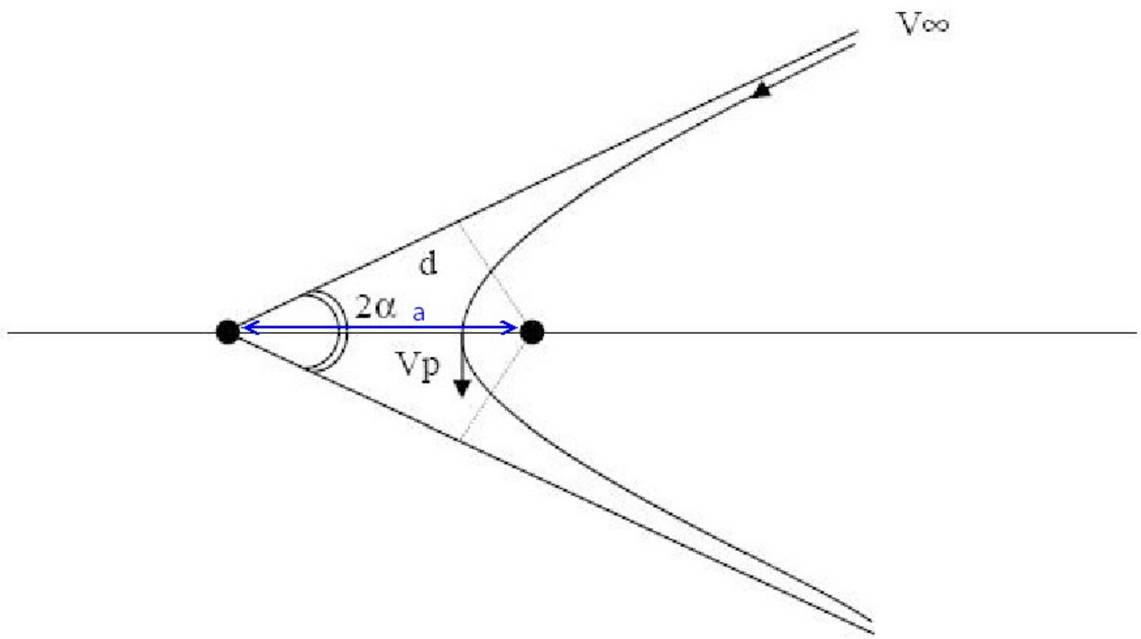


Orbit 1: $V = \sqrt{\mu(\frac{2}{r_1} - \frac{1}{a_1})}$

Orbit 2: $V = \sqrt{\mu(\frac{2}{r_2} - \frac{1}{a_2})}$

And $\Delta V = V_2 - V_1$

I.1.3. Hyperbolic orbit



The equation is the same as the ellipsis equation. However, the eccentricity is superior to one.

$$r = \frac{p}{1+e\cos\theta} \quad and \quad e = \sqrt{\frac{d^2 V_\infty^{\ 4}}{\mu^2} + 1}$$

The speed on the orbit is: $V = \sqrt{\mu(\frac{2}{r} + \frac{1}{a})}$

The speed at the perigee is as follows: $V_p = \sqrt{\frac{\mu}{a}(\frac{e+1}{e-1})}$

The distance between the center of the hyperbole and the intersection of the asymptotes is denoted a. $a = \frac{\mu}{V_\infty^{\ 2}}$

Finally, we have the parameter that expresses as follows: $p = a(e^2 - 1)$

I.2. Trajectory

I.2.1. Approach

We decided that the SRV would use four different orbits to go back to Earth. First, there is the lift-off phase. This one exclusively depends on the propellants and the engine we chose. Therefore, we will develop this part once we would have chosen the adequate propellants. (See II. Propulsion system).

At the end of the lift-off, we choose to go on a circular orbit for several reasons. First, as we do not know the injection point and as all the points have the same characteristics, we will reach the same altitude, wherever the lift-off zone is. Then, if there is a hitch on the mission, we can make multiple rounds to wait for the injection to the other orbit. Orbital maneuvers are also possible like orbit inclination which is not possible with an elliptical orbit.

From the circular orbit, we want to leave the gravitation field of the moon. We have two possibilities to do so. We can opt for either an elliptical orbit or a hyperbolic one. We first tried our calculations with an ellipsis. However, the Microsoft Excel solver was converging on a hyperbole. Therefore, we chose a hyperbolic trajectory.

Then, we would like the SRV to do the main course trajectory thanks to an ellipsis that will lead us to the Earth's atmosphere. Finally, the SRV will turn around the Earth on another circular orbit to wait until the time of the landing.

I.2.2. Calculations

- Circular orbit

For the moment, we will admit that the orbit altitude is at 2363,7 km high.
The speed of the module on a circular orbit is constant and the formula is as follows:

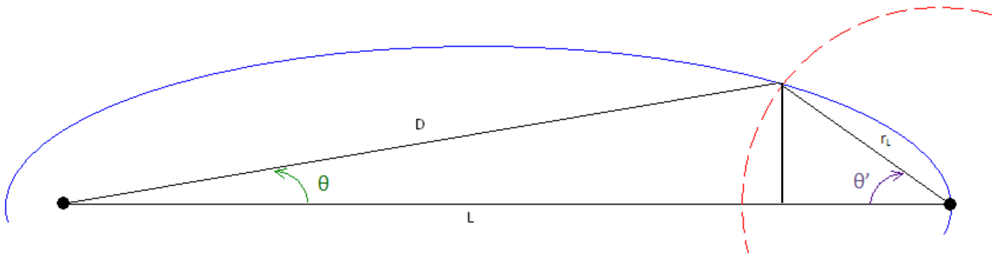
$$V = \sqrt{\frac{\mu}{R}}$$

So we find **V_{circle}= 1439,6 m/sec**

As we have a speed of 2650 m/sec at the end of the lift-off, we have to provide a change in velocity of – 1210,37 m/sec. → **ΔV=1439-2650=-1210,37 m/sec**

- Hyperbolic orbit

Now that we have determined the circular orbit, we are going to find the hyperbole to be tangential to the circle. Besides, we want to pass from the hyperbolic trajectory to an elliptical one. Therefore, to avoid a change in velocity to go on the ellipsis, we are going to find the appropriate hyperbole, which will lead us to find the ellipsis characteristics.



As we can see on the scheme above, there is an intersection point between the ellipsis and the moon's sphere of influence (colored in red on the picture). The hyperbole will also pass by this point. Therefore, to find this hyperbole, we need to find these first two angles θ and θ' and the distance D. The distance r_l corresponds to the radius of the moon's sphere of influence.

Thanks to the Al-Kashi theorem, we find the following relation:

$$D^2 = L^2 + {r_l}^2 - 2\,L\,r_l\,\cos\theta'$$

We also know that, on an ellipse, $D = \frac{p}{1-e\,\cos\theta}$ so it comes $\boxed{\theta = \cos^{-1}(\frac{1}{e}(1-\frac{p}{D}))}$

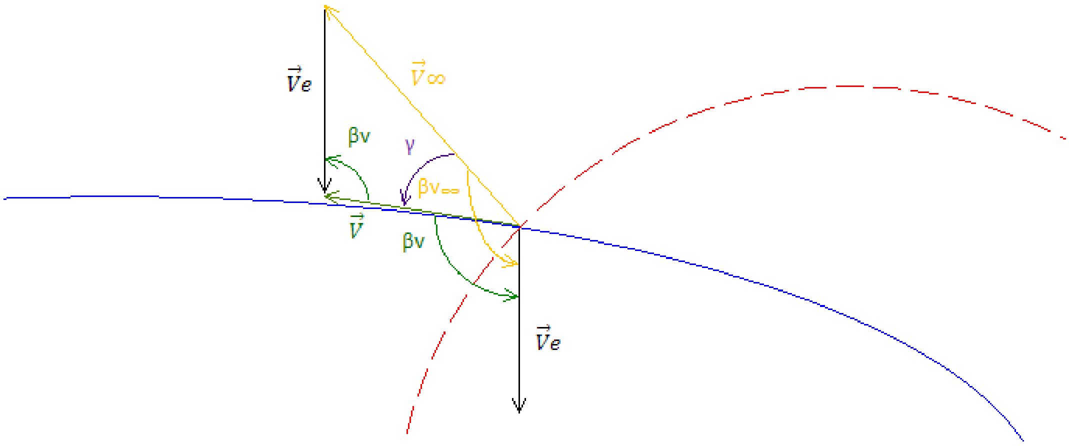
Thanks to the trigonometric relations, we find the expression of the angle θ':

$$\boxed{\theta' = \sin^{-1}(\frac{D}{r_l}\sin\theta)}$$

Therefore, we find a first equation, which is:

$$\boxed{D^2 = L^2 + {r_l}^2 - 2\,L\,r_l\,\cos\left[\sin^{-1}(\frac{D}{r_l}\sin\left[\cos^{-1}(\frac{1}{e}(1-\frac{p}{D}))\right])\right]} \quad (1)$$

Thanks to the scheme below, we can find the different angles.



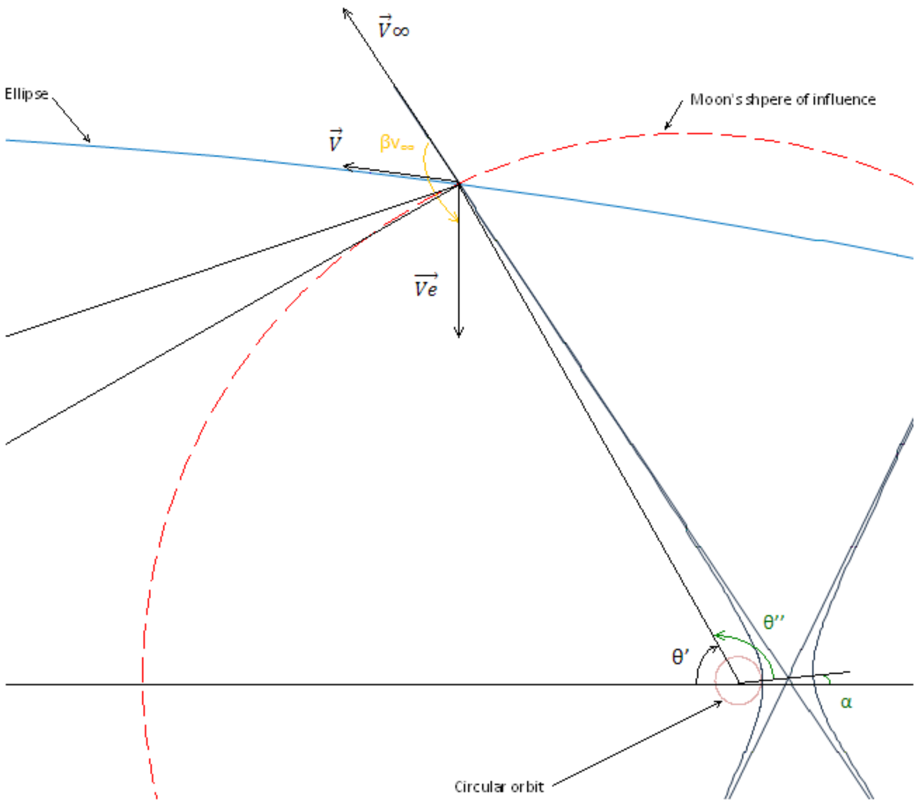
Using the Al-Kashi theorem, we obtain the relations:

$$\begin{cases} V_\infty{}^2 = V_e{}^2 + V^2 - 2\, V_e\, V \cos\beta_v \\ V_e{}^2 = V_\infty{}^2 + V^2 - 2\, V_\infty\, V \cos\gamma \end{cases}$$

We can deduce γ: $\gamma = \cos^{-1}\left(\frac{V_e{}^2 - V_\infty{}^2 - V^2}{-2\, V\, V_\infty}\right)$

We also have:

$\beta_v = \pi - \theta - \cos^{-1}\left(\frac{V_\theta}{V}\right)$ and $\beta_{v\,\infty} = \beta_v + \gamma$



The hyperbole has as equation: $r_l = \frac{p}{1+e\cos\theta"}$

Thanks to the picture above, we can find a relation between α, θ' and θ", which represents the second equation we have to solve.

$$\boxed{\pi - \left(\beta_{v\infty} + \left(\frac{\pi}{2} - \theta'\right)\right) - (\pi - \alpha - \theta") = 0} \quad (2)$$

As we said earlier, we want the hyperbole to be tangential to the circular orbit, so it leads us to the third equation, which is:

$$\boxed{r_{p\,hyperbole} = r_{circle}} \quad (3)$$

The eccentricity (e) and the parameter (p) of the ellipse depend on the ellipsis apogee (r_a).
The angle $\beta_{v\infty}$ also depends on the apogee.
The angles α, θ" and the radius of the hyperbole perigee depend on d, which is a parameter of the hyperbole (cf.I.1.3.hyperbolic orbit)

Thanks to the Microsoft Excel solver, we can solve this system of three equations with three unknown variables, D, d and r_a. We obtain the following results:

Hyperbole	
rp (m)	2363737
βV∞ (°)	145,2
d (m)	4477985
a (m)	2673535
e	1,95
p (m)	7500313
α (°)	59,16
V∞ (m/sec)	1353,65
Vp (m/sec)	2384,74

Calculations	
D (m)	358677854
θ (°)	7,96
θ' (°)	59,56
θ" (°)	116,48
βv (°)	96,88
α (°)	3,96
V (m/sec)	778,11
Vr (m/sec)	-752,17
Vθ (m/sec)	199,21

Now that we have the correct speed on the hyperbole, we can calculate the change in velocity the SRV has to provide to pass from the circular orbit to the hyperbolic one.

$$\Delta V = V_p - V_{circle} = 945{,}1\ m/sec$$

- Ellipsis

As we have resolved the three equations with the apogee of the ellipsis as a variable, we have all its characteristics.

Ellipsis	
ra (m)	4,86E+08
rp (m)	6488135
a (m)	2,46E+08
e	0,97
p (m)	12805468
Va (m/sec)	146,89
Vp (m/sec)	11012,81
Vr (m/sec)	-752,17
Vθ (m/sec)	199,21

However, the perigee is not perfectly defined because it depends on the atmospheric re-entry. Therefore, for the moment, we choose the perigee as the altitude of the Earth's atmosphere.

- Deobitation

Not finished yet

I.3. Atmospheric re-entry

Nowadays, there are two ways to slow down a vehicle when it comes back to Earth: powered braking and atmospheric braking.

I.3.1. Powered braking

This technique consists in turning on retro-rockets engine to create a reverse thrust for the vehicle to brake. Even if this is simple to set up, powered braking is a really arduous and expensive technique because it requires a huge load of liquid propellant and then increases the launching mass.

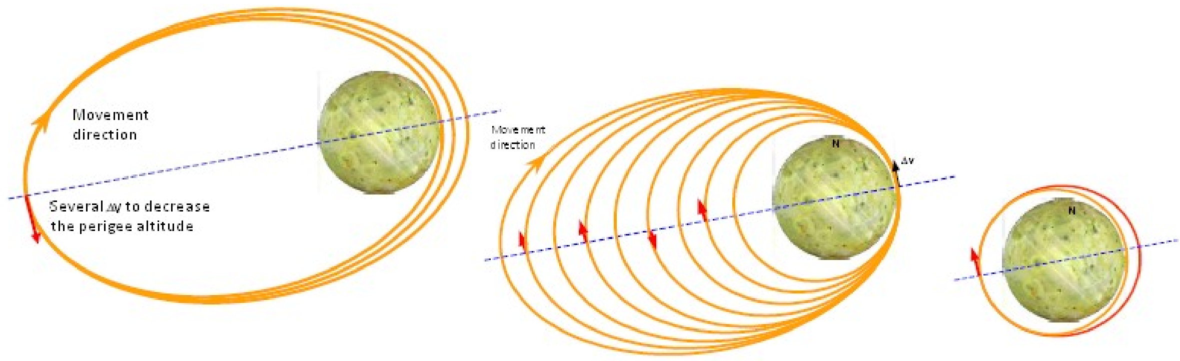
I.3.2. Atmospheric braking

This is an economic technique since it uses the atmosphere to create friction that will lead the vehicle to slow down. We make out two sort of atmospheric braking: aerobraking and aerocapture.

- *Aerobraking*

To begin the aerobraking, the craft has to meet the atmosphere at the perigee (for an elliptical orbit, it is the shortest distance to the planet). Thanks to brief impulsion at the apogee to decrease the craft velocity, aerospace flight mechanics says that the altitude at the perigee, during the next pass, will be lower. Thanks to this maneuver, the orbit passes through the top layer of the atmosphere that creates a friction. This friction enables the vehicle to slow down and to lower the apogee altitude. Repeated drag passes change the initial elliptical orbit into a circular orbit.

Even if this technique seems to be simple, aerobreaking is extremely challenging. At each drag pass, the calculator has to give the engineers the atmosphere's characteristics (height and density) for them to determine the atmosphere's effects on the craft structure. If the vehicle plunges too deeply into the atmosphere, the friction could become really high and damage or destroy the vehicle. On the other hand, if the craft just skim the atmosphere, the braking will not be sufficient to modify the obit parameters.



- *Aerocapture*

Not finished yet

I.4. Landing

- Where should the SRV land on Earth?

Two sites seem to be good for this.

The first one is the Pacific Ocean, the biggest ocean on Earth. The rescue will be expensive because helicopters, aircrafts and/or boats are needed, but the precision can be "bad". Of course, the SRV has to be waterproof. It less needs to slow down than landing in the desert.

The second one is the Nevada desert. In the United States, near the Nellis Air-Force Base so the SRV can be recovered easily. The rescue is not expensive, cars or trucks are sufficient. The precision can be 100 miles. The module needs to resist the aridity and the dust.

The two sites are barely equivalents except for the rescue price so we choose the Nevada desert; we just need a truck with a crane; and to go back to the laboratory to open safely the sample compartment. Of course, if necessary, the module can splash down in the ocean.

- How should it land?

To land in the desert, the SRV can take off at any moment and stay in orbit around the moon or the Earth (after the aerocapture) to wait for the appropriate moment. The SRV will enter the atmosphere by aerocapture and be naturally slowed down once. In the final phase, about some hundred meters from the ground, it would spread a parachute to land softly. By softly we want to say the module can be damaged, but the sample compartment and the commands to access it need to be safe. For this, the headgear's top will be separated from the module by pyrotechnics and pull the parachute out of the module to blow it up.

II. Propulsion system

II.1. Approach

Now that we have determined the trajectory to get back to Earth, we have to design the appropriate propulsion system. As a propulsion system is specific to a rocket or a vehicle, we are going to conceive it entirely. The first step is to choose the oxidizer/fuel combination. Then, thanks to their performance and characteristics, we will find the diameter of the different tanks and the nozzle dimensions in the structure part.

From a list of several oxidizer/fuel combinations, we have to pick one. The aim is to find the oxidizer/fuel combination that optimizes the SRV weight. Therefore, we based our choice on multiple criteria as follows:

Hypergolic: oxidizer and fuel ignite spontaneously on contact with each other and do not need ignition source.
Remaining mass: considering the fuel mass, a minimum of 40 kilograms out of 200 kilograms are necessary to integrate the communication systems, the sample compartment, the SRV structure etc….
Propellant preservation: As the SRV will stay a year on the Moon, the propellant must be easy to preserve to avoid a pressurization system and a too heavy thermal protection on the tanks.

The remaining oxidizer/fuel combinations will be eliminated with other reasons developed in the next parts.

Thanks to an example, we are going to develop all the calculations. All the formulas used afterwards are valid for all the different oxidizer/fuel combinations.
We choose nitrogen tetroxide/MMH combination as an example.

II.2. First selection

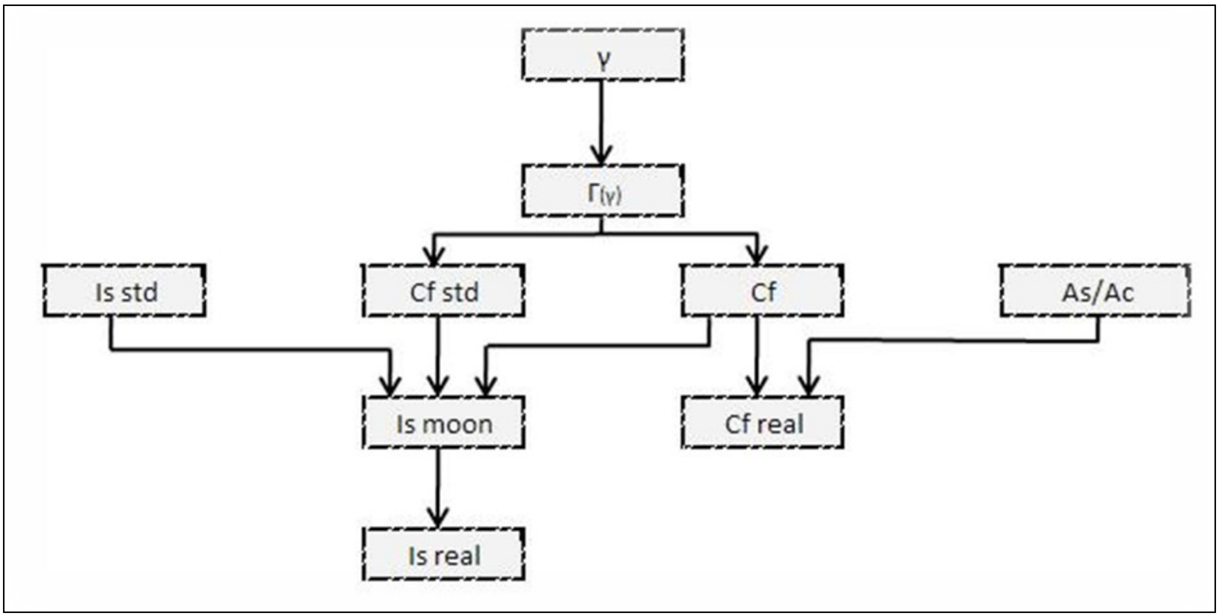
To avoid too many calculations we are going to eliminate the propellants that are not hypergolic. Here is a chart of all the possible propellants and another one after this first selection.

Oxidizer	Fuel	Hypergolic
Liquid Oxygen	Liquid Hydrogen	NO
	Liquid Methane	NO
	Ethanol+25% water	NO
	Kerosene	NO
	Hydrasine	NO
	MMH	NO
	UDMH	NO
	50-50	NO
Liquid Fluorine	Liquid Hydrogen	YES
	Hydrasine	YES
FLOX-70	Kerosene	YES
Nitrogen Tetroxide	Kerosene	NO
	Hydrasine	YES
	MMH	YES
	UDMH	YES
	50-50	YES
Red-Fuming Nitric Acid	Kerosene	NO
	Hydrasine	YES
	MMH	YES
	UDMH	YES
	50-50	YES
Hydrogen Peroxide	Kerosene	NO
	Hydrasine	YES
Nitrous Oxide	HTPB (solid)	NO
Chlorine	Hydrasine	YES
Ammonium Perchlorate (solid)	Aluminum +HTPB	NO
	Aluminum +PBAN	NO

Oxidizer	Fuel	Hypergolic
Liquid Fluorine	Liquid Hydrogen	YES
	Hydrasine	YES
FLOX-70	Kerosene	YES
Nitrogen Tetroxide	Hydrasine	YES
	MMH	YES
	UDMH	YES
	50-50	YES
Red-Fuming Nitric Acid	Hydrasine	YES
	MMH	YES
	UDMH	YES
	50-50	YES
Hydrogen Peroxide	Hydrasine	YES
Chlorine Pentafluoride	Hydrasine	YES

II.3. Propellants performance

We are now going to calculate the remaining propellants performance, which will be useful afterwards. The graphic below explains how we calculated them.



Reminder: Calculations made for Nitrogen tetroxide/MMH combination.

As we don't have the exact heat capacity ratio we approximate the value to 1,22.

So, γ= 1,22

$$\Gamma(\gamma) = \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}}\sqrt{\gamma}$$ **Γ(γ) = 0,65**

With this function we can calculate the thrust coefficient in standard conditions, in other words the Cf when the nozzle exit pressure is $10^5\ Pa$ and the combustion chamber is $7 \cdot 10^6\ Pa$.

$$Cf\ std = \Gamma(\gamma)\sqrt{\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{P_s}{P_0}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$ **Cf std= 1,59**

For the nozzle to work adequately into space, we command $\frac{A_s}{A_c} = 50$

We also know the $\frac{A_s}{A_c}$ formula, which is: $\frac{A_s}{A_c} = \frac{\Gamma(\gamma)}{\left(\frac{P_s}{P_0}\right)^{1/\gamma}\sqrt{\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{P_s}{P_0}\right)^{\frac{\gamma-1}{\gamma}}\right]}}$

P_s=nozzle exit pressure
P_0= combustion chamber pressure

We approximate the combustion chamber pressure to $10^6\ Pa$ which is a good value for this type of nozzle. Since we have P_0 and using Microsoft excel solver we find the ratio $\frac{P_s}{P_0}$ that leads us to Ps value.

Finally, $\frac{P_s}{P_0} = \frac{1}{690}$ and $P_s = \frac{P_s}{P_0} * P_0 = 1448{,}5\ Pa$

Let's determine the real thrust coefficient Cf_{real} and the real specific impulsion Is_{real}.

$$Cf\ real = 0{,}98 * \Gamma(\gamma)\sqrt{\frac{2\gamma}{\gamma-1}\left[1-\left(\frac{P_s}{P_0}\right)^{\frac{\gamma-1}{\gamma}}\right]} + \frac{P_s}{P_0} * \frac{A_s}{A_0}$$

0,98 represents the expansion quality coefficient.

For Nitrogen Tetroxide and MMH combination the thrust coefficient is worth Cf_{real}=1,84.

With this coefficient, we can now calculate the specific impulsion on the moon.
Thanks to the following relation, $I_{s\ moon} = I_{s\ std} * \frac{C_f}{C_{f\ std}}$, we find Is_{moon} and Is_{real}.

Is_{moon}=331,2 sec
Is_{real}=0,96*Is_{moon} =318 sec
0,96 represents the global quality coefficient (combustion + expansion).

Oxidizer	Fuel	Mixture ratio	IS std	γ	Γ(γ)	Is moon (sec)	Is real (sec)	Cf std	Cf	Cf real
Liquid fluorine	Liquid hydrogen	0,17	400	1,33	0,67	461,7	443,2	1,54	1,78	1,74
	hydrasine	0,55	338	1,33	0,67	390,1	374,5	1,54	1,78	1,74
FLOX-70	Kerosene	0,26	320	1,22	0,65	378,6	363,4	1,59	1,88	1,84
Nitrogen Tetroxide	hydrasine	0,93	286	1,26	0,66	335,2	321,8	1,57	1,84	1,80
	UDMH	0,48	277	1,22	0,65	327,7	314,6	1,59	1,88	1,84
	MMH	0,58	280	1,22	0,65	331,2	318,0	1,59	1,88	1,84
	50-50	0,63	280	1,24	0,66	329,7	316,5	1,58	1,86	1,82
Red-Fuming Nitric Acid	Hydrasine	0,78	276	1,22	0,65	326,5	313,4	1,59	1,88	1,84
	MMH	0,47	269	1,22	0,65	318,2	305,5	1,59	1,88	1,84
	UDMH	0,38	266	1,22	0,65	314,7	302,1	1,59	1,88	1,84
	50-50	0,52	270	1,22	0,65	319,4	306,6	1,59	1,88	1,84
Hydrogen Peroxide	Hydrasine	0,47	269	1,22	0,65	318,2	305,5	1,59	1,88	1,84
Chlorine Pentafluorine	Hydrasine	0,47	297	1,22	0,65	351,3	337,3	1,59	1,88	1,84

II.4. Propellant mass calculation

II.4.1. Thrust

Thanks to Newton's second law, we can determine the thrust needed to take off.

$$\sum \vec{F} = M\,\vec{\gamma}$$

The forces, applied on the SRV, we take into account are:

- the SRV weight, P
- the reaction to this force, T

$\sum \vec{F} = \vec{P} + \vec{T} = M\,\vec{\gamma}$

Projecting the equation on the vertical axis, we obtain:

$$T - M g_{moon} = M\gamma \qquad \text{or} \qquad T = M(\gamma + g_{moon})$$

M: SRV mass
γ: acceleration undergone by the vehicle= g_{moon}

With an initial mass of 200kg, we find the following thrust: **T=649,25 N**

II.4.2. Mass calculations

According to the initial evaluation of mass budget, done by UAH students, we based our calculations on a 250kg SRV. However, weeks later, they revised the mass budget, and we approximated our SRV mass to 200kg.

In order to go from one orbit to another, the vehicle has to change his velocity that will modify the orbit parameters. This change in velocity (ΔV) is made thanks to short engine's impulsions, which obviously consume fuel. The more important is the change in velocity the heavier is the fuel consumption.

As the trajectory is divided into four main parts, from the lift-off to the arrival around the Earth, the propellant loads calculations has four steps.

- Lift-off

As written in the preliminary requirements, we have to provide to the SRV a change in velocity of 2650 m/s for the initial lift-off. Thanks to the following formula, we can calculate the fuel consumption during this phase and then deduce the SRV mass at the end of the lift-off phase.

$$\Delta M = M_i \left(1 - e^{\left(\frac{-\Delta V}{I_{s\,real} \cdot g_0}\right)}\right)$$

ΔM: consumed fuel mass

M_i=200 kg
ΔV=2650 m/s
$I_{s\ real}$= 318 sec
g0=constant=9,801 m/s^2

ΔM= 114,63 kg

$$M_f = M_i - \Delta M$$

M_f = 85,37 kg

- End of lift-off to circular orbit

The final mass for the lift-off corresponds now to the initial mass of this phase. Applying the same formula as above, we can calculate the fuel consumption.

Mi=85,37 kg
ΔV=-1209,8 m/s

Mf= 65,34 kg

- Circular orbit to hyperbolic orbit

Thanks to the trajectory calculations, the SRV need a change in velocity of 834,86 m/s^2 to go to the hyperbolic orbit.

Mi=65,34 kg
ΔV=834,86 m/s

Mf= 54,33 kg

- Aerocapture

Not finished yet

- Deorbitation

From the circular orbit, we would need a ΔV to deorbit and enter the atmosphere. However, we cannot calculate it because this is after the aerocapture and, as we said before, this technique is being developed. We are not able to determine the circular orbit altitude so we cannot calculate the ΔV the SRV has to provide.

- Trajectory course maneuver system

The preliminary requirements impose a change in velocity of 150 m/s for midcourse trajectory correction. After calculations, we found a 1,77 kg fuel consumption.
Besides, we will probably need to correct the trajectory on the circular orbit or on another one. Therefore, we have to take an extra-load propellant. In general, this extra-load is situated between 5 and 8 per cent of the fuel mass. We opt for an 8% extra-load fuel to take into account the aerocapture and the Deorbitation. However, we do not know if this reserve will be sufficient.

- Total mass

The total mass of fuel is the sum of the fuel consumption of each phase to pass from an orbit to another plus the extra-load: we obtain **M_{fuel}= 159,24 kg**.

Finally, with an initial mass of 200kg, it remains 40,76 kg for the structure, the equipments and the samples compartments.

Thanks to the chart below, we can see the different fuel consumption according to the propellant.

	Oxidizer	Liquid fluorine		FLOX-70	Nitrogen Tetroxide				Red-Fuming Nitric Acid				Hydrogen Peroxide	Chlorine Pentafluorine
	Fuel	Liquid hydrogen	Hydrasine	Kerosene	Hydrasine	UDMH	MMH	50-50	Hydrasine	MMH	UDMH	50-50	Hydrasine	Hydrasine
Take-off	*ΔV (m/sec)*	2650	2650	2650	2650	2650	2650	2650	2650	2650	2650	2650	2650	2650
	Mi (kg)	200	200	200	200	200	200	200	200	200	200	200	200	200
	Mfuel(kg)	91,34	102,85	104,96	113,69	115,33	114,55	114,89	115,59	117,47	118,29	117,19	117,47	110,28
	Mf (kg)	108,66	97,15	95,04	86,31	84,67	85,45	85,11	84,41	82,53	81,71	82,81	82,53	89,72
Take-off =>> Circular orbit	*ΔV (m/sec)*	1209,80	1209,80	1209,80	1209,80	1209,80	1209,80	1209,80	1209,80	1209,80	1209,80	1209,80	1209,80	1209,80
	Mi (kg)	108,66	97,15	95,04	86,31	84,67	85,45	85,11	84,41	82,53	81,71	82,81	82,53	89,72
	Mfuel(kg)	25,49404	22,79	22,30	20,25	19,87	20,05	19,97	19,80	19,36	19,17	19,43	19,36	21,05
	Mf (kg)	83,17	74,36	72,74	66,06	64,81	65,40	65,14	64,60	63,17	62,54	63,38	63,17	68,67
Circular orbit =>> Hyperbolic orbit	*ΔV (m/sec)*	834,86	834,86	834,86	834,86	834,86	834,86	834,86	834,86	834,86	834,86	834,86	834,86	834,86
	Mi (kg)	83,17	74,36	72,74	66,06	64,81	65,40	65,14	64,60	63,17	62,54	63,38	63,17	68,67
	Mfuel(kg)	14,01	12,53	12,26	11,13	10,92	11,02	10,98	10,89	10,64	10,54	10,68	10,64	11,57
	Mf (kg)	69,15	61,83	60,43	54,90	53,83	54,33	54,12	53,66	52,47	51,95	52,64	52,47	57,04
Midcourse correction	*ΔV (m/sec)*	150	150	150	150	150	150	150	150	150	150	150	150	150
	Mfuel (kg)	2,25	2,02	1,97	1,79	1,76	1,77	1,77	1,75	1,71	1,70	1,72	1,71	1,86
	M sub-total fuel (kg)	133,10	140,19	141,49	146,86	147,87	147,39	147,60	148,03	149,19	149,69	149,02	149,19	144,76
	Extra-load	8%												
	M total fuel (kg)	143,75	151,40	152,81	158,61	159,70	159,18	159,41	159,87	161,12	161,67	160,94	161,12	156,34
	M dry (kg)	56,25	48,60	47,19	41,39	40,30	40,82	40,59	40,13	38,88	38,33	39,06	38,88	43,66

II.5. Selection of the appropriate combination

According to the criteria we settled, we have to eliminate the oxidizer/fuel combinations that allocate less than 40kg for dry mass. As we can see on the chart above, four out of thirteen combinations do not respect this condition. Therefore, we rule three out of four propellants with Red-Fuming Nitric Acid as oxidizer and hydrogen peroxide/Hydrasine out.

Among the nine remaining propellants, we leave the four combinations with Hydrasine as fuel aside. Hydrasine is really toxic and unstable when used alone. Besides, we eliminate FLOX-70/Kerosene because this combination is generally used for rockets not for modules. The liquid fluorine/Liquid hydrogen is also ruled out because we need a system to keep the hydrogen at the liquid phase. As we do not have a very important dry mass, we cannot afford to integrate this kind of system that will weigh down our SRV.

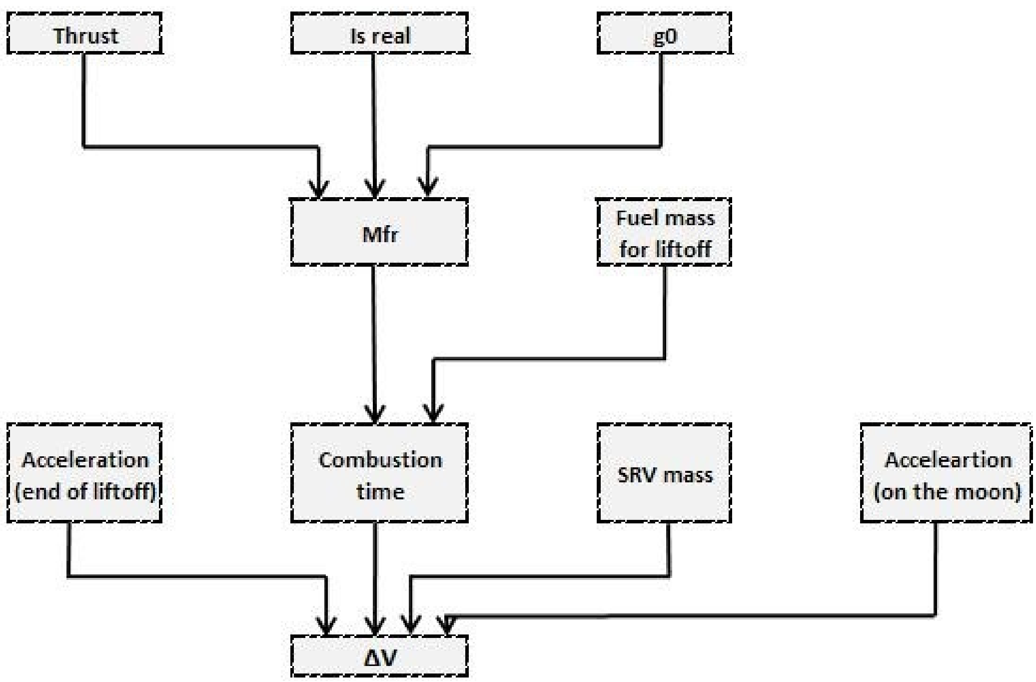
Finally, it remains three possible combinations with Nitrogen tetroxide as oxidizer. As it is the one with the most remaining dry mass and as we use it for modules, we chose **Nitrogen tetroxide/MMH combination**.

II.6. Lift-off validation

We are now going to check if the oxidizer/fuel combination we chose is able to provide a change in velocity of 2650m/sec for the initial lift-off.

- Approach

First, we are going to calculate the mass flow rate of fuel, related to the thrust and the specific impulsion. Therefore, for each combination we will obtain a different mass flow rate. Then, thanks to this mass flow rate, we will determine the time for the propulsion system to consume the fuel.
Finally, we will find the change in velocity for each combination.



- Calculations

Let's calculate the mass flow rate of fuel. We can relate the thrust to the mass flow rate with the following formula:

$$\boxed{T = M_{fr}\, I_{s\,real}\, M}$$

So, $\boxed{M_{fr} = \frac{T}{I_{s\,real}\, M}}$ We find **M_{fr}= 0,21 kg/sec**

T= 649,25 N
$I_{s\,real}$= 317,6 sec
M=200 kg

Thanks to the mass of fuel needed for lift-off, we can determine the time to consume it.

$\boxed{t = M_f / M_{fr}}$ **t= 549,82 sec**

To know if the propellant we chose can deliver a change in velocity of 2650 m/sec we are going to make the average of the accelerations at the beginning and end of lift-off. Multiplying by the time to consume the fuel, we will obtain the change in velocity.

$$\boxed{\Delta V = \left(\frac{T}{M_{f\ liftoff}} + \frac{T}{M_{SRV}}\right) * 0{,}5 * t}$$

We find **ΔV= 2981,10 m/sec**

As the change in velocity we found is higher than the wanted one, the propellant is adapted for our SRV.

II.7. Oxidizer and fuel mass

To determine the size of the tanks we need to know the mass of both, oxidizer and fuel.
Obviously, the total fuel mass is the sum of the oxidizer and the fuel masses. We also know that the mixture ratio is the ratio between the masses of the oxidizer and the fuel. Therefore, to find the quantity of oxidizer and fuel, we need to solve a 2-equation system with two unknown quantities.

$$\begin{cases} M_{Total} = M_{oxidizer} + M_{fuel} \\ \varphi = \frac{M_{oxidizer}}{M_{fuel}} \end{cases}$$

We find, $\boxed{M_{fuel} = \frac{M_{Total}}{\varphi + 1}}$ and $\boxed{M_{oxidizer} = \varphi\, M_{fuel}}$

For our propellant (nitrogen tetroxide/MMH), the mixture ratio is $\varphi = 0{,}58$ and $M_{Total} = 159{,}18\ kg$.

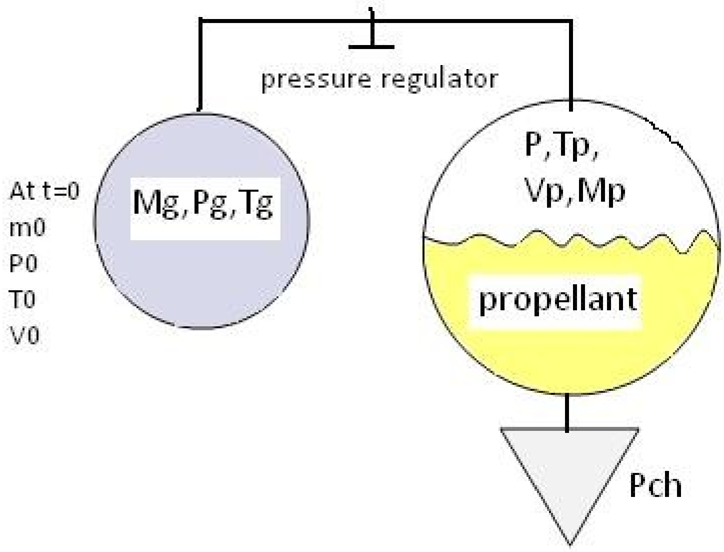
M $_{fuel}$= 100,8 kg **M $_{oxidizer}$= 58,3 kg**

II.8. Pressurization system

For the propellants to go into the combustion chamber we have the choice between two systems. We use either a turbo pump system or a pressurized system. As we have a low dry mass, we can't use a turbo pump, which is very bulky and heavy. Therefore, we opt for a pressurization system with helium. Thanks to the pressure into the helium tanks, the propellants will be pushed into the combustion chamber. Besides, it is an inert gas, so it will not react with the propellants combination.

As we have four tanks of propellants and for stability issue, we chose to adopt two tanks of helium that will feed two tanks of propellants each. According to the propellants volume the helium has to move, we are able to determine the needed helium quantity. For the tank not to be full, we add an extra volume of 10% of the propellants volume.

$V_h = (V_{oxidizer} + V_{fuel}) * 1{,}1$ → V_h=0,15 m^3



To determine the helium mass, we will admit the following relation:

$$m_0 = \frac{P\,V_p}{r\,T_0} * \frac{\gamma}{1-P_g/P_0}$$

For propellants we can stock, we can package the helium at a temperature of 280 K=T_0. For this kind of propulsion system, the pressure in the combustion chamber is about 1.10^6 Pa. Therefore, the pressure in the propellant tank must be higher; considering the pressure losses in the system, the pressure in the combustion chamber and in the injectors, we take 18.10^5 Pa in the propellant tank. Besides, the helium tank is under 400.10^5 Pa when full and 20.10^5 Pa when almost empty.

<u>Characteristics of the system</u>
$P=18.10^5$ Pa
$Pg=20.10^5$ Pa
$P_0=400.10^5$ Pa
$T_0=280$ K
$M_{mass}=4.10^{-3}$ kg/mol
$\gamma=1,66$
$r=R/M_{mass}$
$R=8,314$

m_0= 0,811 kg

Now, we can calculate the volume of helium with the ideal gas law.

$$V_0 = \frac{m_0\, r\, T_0}{P_0} \quad \longrightarrow \quad \mathbf{V_0=1,18.10^{-2}\ m^3}$$

As we have two tanks, we calculate the tank diameter with half of the volume found above.

$$D_{helium} = \sqrt[3]{\frac{3\,V_0}{\pi}} = 0,11\ m$$

II.9. Circular orbit altitude

Now that we have defined all the nozzle characteristics, we are able to find the SRV altitude at the end of the liftoff. As the vehicle gains height it gets lighter thanks to the fuel consumption. Therefore, its acceleration changes throughout the liftoff. This acceleration is defined as follows: $\boxed{\Gamma(t) = \frac{T}{M_{SRV} - M_{fr}\, t}}$

T: nozzle thrust
M_{fr}*t: fuel consumption
t: time at the end of the liftoff

To find the SRV altitude we have to integrate this equation twice. We will integrate it in two steps: first, we will find the vehicle velocity and then its altitude as a function of time.

$$V(t) = \int \Gamma(t)\, dt = \int \frac{T}{M_{SRV} - M_{fr}\, t}\, dt$$

$$V(t) = \frac{T}{M_{fr}} \ln\left(\frac{M_{SRV} - M_{fr}\, t}{M_{SRV}}\right) \quad \longrightarrow \quad \boxed{V(t) = \frac{T}{M_{fr}} \ln\left(1 - \frac{M_{fr}\, t}{M_{SRV}}\right)}$$

The altitude reached for a change in velocity of 2650 m/sec is as follows:

$$Z(t) = \int V(t) = \int \frac{T}{M_{fr}} \ln(1 - \frac{M_{fr}\ t}{M_{SRV}})\ dt = \frac{T}{M_{fr}} \int \ln(1 - \frac{M_{fr}\ t}{M_{SRV}})\ dt$$

For the calculation to be simpler, we make a variable change to make the function $\ln(1+X)$ appear. We can easily integrate it.

We put down: $X = -\frac{M_{fr}\ t}{M_{SRV}}$ It follows $dX = \frac{M_{fr}}{M_{SRV}}\ dt$

$$\boxed{Z(t) = \frac{-T\ M_{SRV}}{M_{fr}{}^2} \int \ln(1+X)\ dX}$$

We are going to replace this logarithm function with its Taylor series for the integration to be easier. However, we first have to find the Taylor series degree. To do so, we try to integrate several Taylor series with a different degree. We remark that the limit of the Taylor polynomials, which correspond exactly to the logarithm function, is at the 10th order. Indeed, when we integrate this Taylor series and the logarithm, we notice that their curves are merged.

<u>Taylor series</u>

$$\ln(1+X) = \sum_{i=0}^{n} (-1)^n \frac{X^{n+1}}{n+1}$$

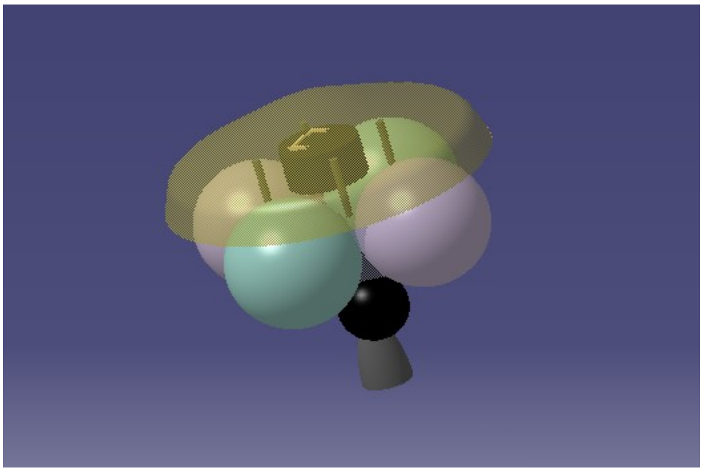
$$\int \ln(1+X)dX = \int \sum_{i=0}^{n} (-1)^n \frac{X^{n+1}}{n+1} \quad \Longrightarrow \quad \boxed{\int \ln(1+X)dX = \sum_{i=0}^{n} (-1)^n \frac{X^{n+2}}{(n+2)\,(n+1)}}$$

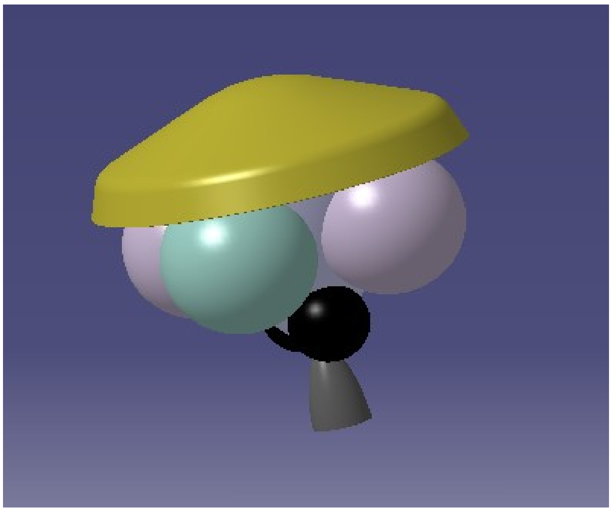
After calculation we find Z=623,4 km

Considering the radius of the moon the altitude at the end of the liftoff is **Z=2363,7 km**.

Interfaces

I. Design





We chose this design for several reasons. First, we tried to optimize the body to minimize the mass structure. For that we needed the tanks to be as much as possible outside the body, the sphere being the best shape (cf. 3-view plan p.41).
There are two tanks for each fuel components to equilibrate the module.
To get into the Earth atmosphere we need some protection. A shield is perfect for that because it deflects the high temperature flow. It has a double angle of attack to protect the tanks a little bit more from the flow and reflow.

II. Structure

II.1. Tanks diameter

We chose to adapt spherical tanks on our SRV. For a stability issue, we take two tanks for the nitrogen tetroxide and two others for the MMH.
In order to determine the tanks diameter we first have to calculate the volume of the tanks. To do that, we need the density of both propellants.

- Nitrogen Tetroxide

We know that $M_{oxidizer} = \rho V$
Nitrogen tetroxide has a density of 1447 kg/m^3

So, $V = \frac{M_{oxidizer}}{\rho}$ ➡ **V_{total}= 0,07 m^3**

As we take 2 tanks, we divide this volume by 2, to determine the diameter of one tank.

The volume of a sphere is $\frac{V_{total}}{2} = \frac{4}{3}\pi(\frac{D}{2})^3$ ➡ $\boldsymbol{D_{oxidizer} = \sqrt[3]{\frac{3\,V_{total}}{\pi}} = 0,42\ m}$

- MMH

We use the same approach as above to determine the MMH tank diameter.

We find, for one tank $\boldsymbol{V=0,033\ m^3}$ and $\boldsymbol{D_{fuel} = 0,41\ m}$

II.2. Tanks weight

To calculate the tanks weight we first need their thickness. To do so, we have a formula:

$$e = \frac{P * r}{\sigma}$$

With P the pressure into the tank, r the tank radius and σ the material admissible constraint

Then we need the material volume and its density to have the mass.

$V = \frac{4}{3}\pi\,(r^3 - (r-e)^3$ Then $m = V * \rho$

We compared the 3 most common metals: Aluminum, titanium and steel. Here is one chart for each kind of tank.

N2O4	R (m)	P(Pa)	e(m)	σ(Pa)	Vol (m^3)	ρ (kg/m^3)	Mass (kg)
Titanium	0,209	1,80E+06	4,18E-04	9,00E+08	2,30E-04	4500	**1,034**
Aluminium	0,209	1,80E+06	1,26E-03	3,00E+08	6,86E-04	2700	**1,853**
Steel	0,209	1,80E+06	3,14E-04	1,20E+09	1,72E-04	7900	**1,361**

MMH	R (m)	P(Pa)	e(m)	σ(Pa)	Vol (m^3)	ρ (kg/m^3)	Mass (kg)
Titanium	0,206	1,80E+06	4,12E-04	9,00E+08	2,20E-04	4500	**0,989**
Aluminium	0,206	1,80E+06	1,24E-03	3,00E+08	6,57E-04	2700	**1,773**
Steel	0,206	1,80E+06	3,09E-04	1,20E+09	1,65E-04	7900	**1,303**

Helium	R (m)	P(Pa)	e(m)	σ(Pa)	Vol (m^3)	ρ (kg/m^3)	Mass (kg)
Titanium	0,112	4,00E+07	4,98E-03	9,00E+08	7,52E-04	4500	**3,386**
Aluminium	0,112	4,00E+07	1,49E-02	3,00E+08	2,06E-03	2700	**5,562**
Steel	0,112	4,00E+07	3,74E-03	1,20E+09	5,71E-04	7900	**4,509**

We can see that Titanium is the best metal for all the tanks because the tanks mass is minimal.

	N2O4	MMH	Helium
e(m)	4,18E-04	4,12E-04	4,98E-03
Mass (kg)	1,034	0,989	3,386

II.3. Nozzle dimensions

II.3.1.Exit and neck diameter of the nozzle

As written in the *propulsion system* part (II.3), we take $\frac{A_s}{A_c} = 50$, $P_0=10^6$ Pa.

We can link the nozzle's neck section to its thrust with the following relation: $A_c = \frac{T}{C_{f\ real}\, P_0}$

With T= 649,25 N, $C_{f\ real}$= 1,84 , we find Ac= $3{,}52 \cdot 10^{-4}$ m²

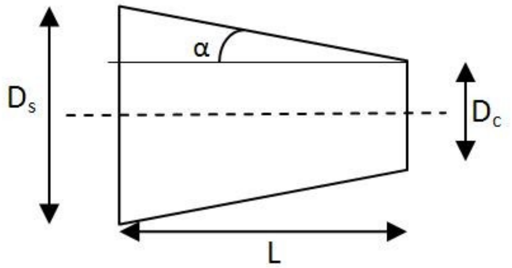
Therefore, we find the exit section of the nozzle: $A_s = 50 * A_c$ It means A_s= 0,0176 m²

We know that $A_s = \frac{\pi\, D_s^2}{4}$ so, we can find the exit diameter and the neck diameter of the nozzle.

D_s= 0,15 m and **D_c= 0,021 m**

II.3.2. Nozzle's length

To simplify the calculation, we consider the nozzle as a trapeze. The optimum angle for a nozzle is α=15°.



$$L = \frac{D_s - D_c}{2 \tan 15}$$

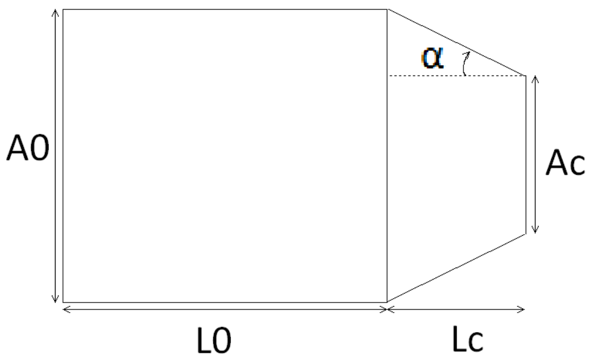
And finally, L= **0,24 m**

II.3.3. Nozzle's material

To construct the nozzle we use niobium, one of the most resistant metals in temperature. The temperature at Ac is the same as in the chamber: 3473K, but the fusion temperature of the niobium is 2750K so we need a cooling system. We use the gas before entering the chamber. It passes through pipes all around the nozzle top part. To estimate the mass we suppose the nozzle is a trapeze with an average thickness of 5mm (top thicker than the bottom).

Ac (m^2)	3,53E-04
As (m^2)	1,71E-02
Niobium ρ (kg/m^3)	8570
length (m)	0,236
e(m)	0,005
Area (m^2)	2,58E-02
volume (m^3)	1,29E-04
mass (kg)	**1,104**

II.4. Combustion chamber dimensions



To conceive the combustion chamber we need four value. The two length L0 and Lc and the two radius r0 and rc.
To calculate all the dimensions we have several formulas:

Areas ratio: $\sigma 0 = \frac{A0}{Ac}$

Cone length: $Lc = 2\sqrt{\frac{Ac}{\pi}} * \frac{\sqrt{\sigma 0}-1}{2\tan\alpha}$

Sound velocity: $a = \sqrt{\gamma * \frac{R}{Mmass} * T0}$

Mach number: $M0 = \frac{Vgaz}{a}$

We also know that:

$$A0 = Ac * \left(\frac{\Gamma}{\sqrt{\gamma}}\right) * \frac{1}{M0} * \left(1 + \frac{\gamma-1}{2} * M0^2\right)^{\frac{\gamma+1}{2(\gamma-1)}}$$

We put down:
Cylinder length: L0=r0*2; Cone angle: α=45°

And we already have:

- Γ=0,676
- The combustion temperature: T0=3200°C
- MMH molecular mass: Mmass=0,046 kg/mol
- R=8,314
- γ=1,35
- Gas velocity: V_{gaz}=100m/s

Datas		Results	
T0 (K)	3473	M0	0,109
γ	1,35	a (m/s)	920,55
Vgaz (m/s)	100	Ac (m^2)	3,53E-04
σ0	5,39	A0 (m^2)	1,90E-03
α (°)	45	rc m)	**0,0106**
Mmass (kg/mol)	0,046	L0 (m)	**0,0492**
Γ (γ)	0,676	r0 (m)	**0,0246**
R	8,314	Lc (m)	**0,0087**

To construct the chamber we use niobium like for the nozzle. The temperature into the chamber is 3473K so we need a material that can resist. Although the niobium fusion temperature is 2750K, it is one of the most resistant metals. We will use a cooling system to decrease the walls temperature. This cooling system consists in tubes all around the chamber (and a part of the nozzle) that conduct fresh gas (MMH or N2O4) before entering the chamber.

We estimate the total mass with a thickness of 1cm:

Niobium ρ (kg/m^3)		8570
Chamber area (m²)		1,13E-02
	A0 (m²)	1,90E-03
	Cylinder (m²)	7,61E-03
	Cone (m²)	1,82E-03
e (m)		0,01
volume (m^3)		1,13E-04
mass (kg)		**0,972**

Environments

I. Thermal protection

The SRV needs to be protected from very high temperatures (+1650°C) during the atmosphere entrance and from high and low temperatures (+200°C/-150°C) in space.

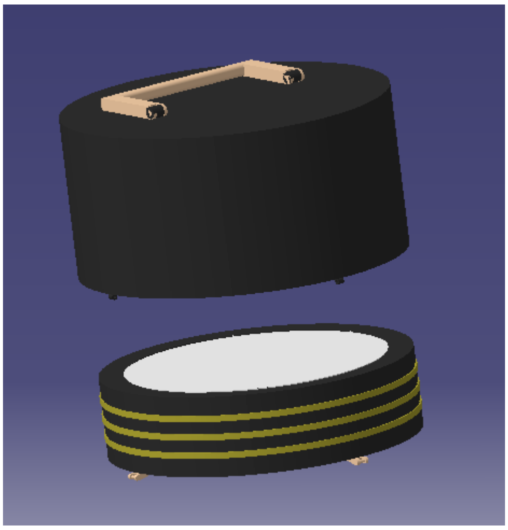
We can just give some ideas because we could not determinate the fluxes. Besides, because of the aerocapture, we do not know the entry angle into the atmosphere.
In space, only radiations can provide temperature. We need colors that reflect a lot the solar radiations. That is why every satellites and modules have gold Mylar sheets. We choose the same protection: simple, thin and light.

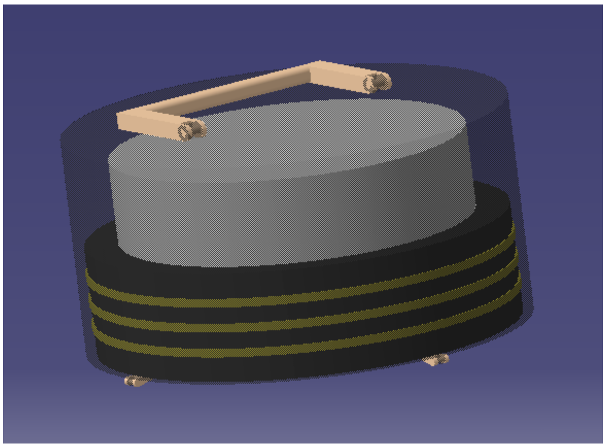
The embedded systems do not like high and low temperature difference. Maybe we can some black plates to keep heat and white plates to provide cold to minimize the range.

In the atmosphere, we need better protection, especially on the headgear. At 1650°C almost all metals melt. Materials like the RCC, reinforced carbon-carbon are better. We can use RCC linked with inconel (metal) and quartz sheets, like on the nose of the Americans orbiters.

II. Samples compartment

The most important part of the SRV is the sample compartment. The samples have to stay viable after the landing. It has to be portable because it cannot be opened in the desert, the samples would be infected after that. The box is a 25cm-diameter circle and about 13cm high. It can be opened in two equivalent parts.
In the inside, there is a material like EVA mousse, which resists lunar temperatures, to maintain the samples softly.
All around the under part there is a series of joints (like the joints on a piston of a car engine) to hermetically close the box. In case of the fault of one joint, there are two more.
On the top part, a ring goes on the joints.





At the end, there are mobile parts that can bend to maintain the compartment closed. A flexible handle can be added on the top.
There is a sensor inside and when it detects the sample weight, then the contact of the two parts make the mobile parts bend to maintain the box closed even if the SRV open again. As the box is hermetic, the pressure inside is the same as on the moon. The regolith temperature can vary between -150°C and 100°C so we have to adapt. Inside the box, there are heaters or coolers, to stay at the moon temperature.
The heater system is an electric circuit with resistances supplied by a lithium-ion battery, the best ratio longevity/power in batteries.

The compartment is in the center of the SRV so if the thermal protections are efficient we can use any metals like aluminum.

III. Systems

We are not specialized in embedded systems. We favored the mechanical parts but we had not enough time to develop this part more than we did.

The SRV needs some systems to operate correctly.

- A calculator

The SRV trajectories have to be calculated and updated at any moment to prevent drifts. The calculator needs to be coupled with an ignition system to command the impulsions.

- An ignition system

It is coupled with the calculator that tells when to operate. The ignition system open or close the fuel injectors.

- Sample compartment unit manager

The compartment can be opened with the SRV headgear, as often as wanted when it is empty. However, when the samples are inside it needs to stay closed even if the headgear is opened.
There are sensors to detect the samples in the box and other ones to detect the contact between the box two parts. If these conditions are realized, the unit closes the compartment with the top mobile parts and leaves it like this. The compartment cannot be opened anymore.
There are also temperature sensors on the SRV to detect the Moon temperature and consequently we start the heater or the cooler system in the box to keep the samples at the same temperature.

- Communication system

A communication is useful to give and receive instructions and information from the Earth.

- Unit to open the headgear

The headgear can be lift by jacks. An electronic unit has to command it. This unit needs also a manual command outside the body, well protected by a removal panel, to allow scientists to lift the headgear and take the samples compartment. This unit can also command the parachute exit. An altimeter is needed to determine when the parachute is supposed to blow up.

- Beacon

The SRV needs to be detected when it has landed to be found easily. We use a beacon for that.

All the systems can be situated under and around the sample compartment.

Nomenclature

I_s: specific impulsion
Cf: thrust coefficient
C*: characteristic exhaust velocity
ρ: density
γ : heat capacity ratio
Mi: initial mass
Mf: final mass
Mfr: mass flow rate
A_s: nozzle's exit section
A_c: nozzle's neck diameter
A_0: combustion chamber section
M_{mass}: molecular mass
ra: apogee
rp: perigee

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Book

Cours de technologie spatiale. Techniques et technologies des véhicules spatiaux. *Volume 3 Plateformes* Edition du CNES

Appendix

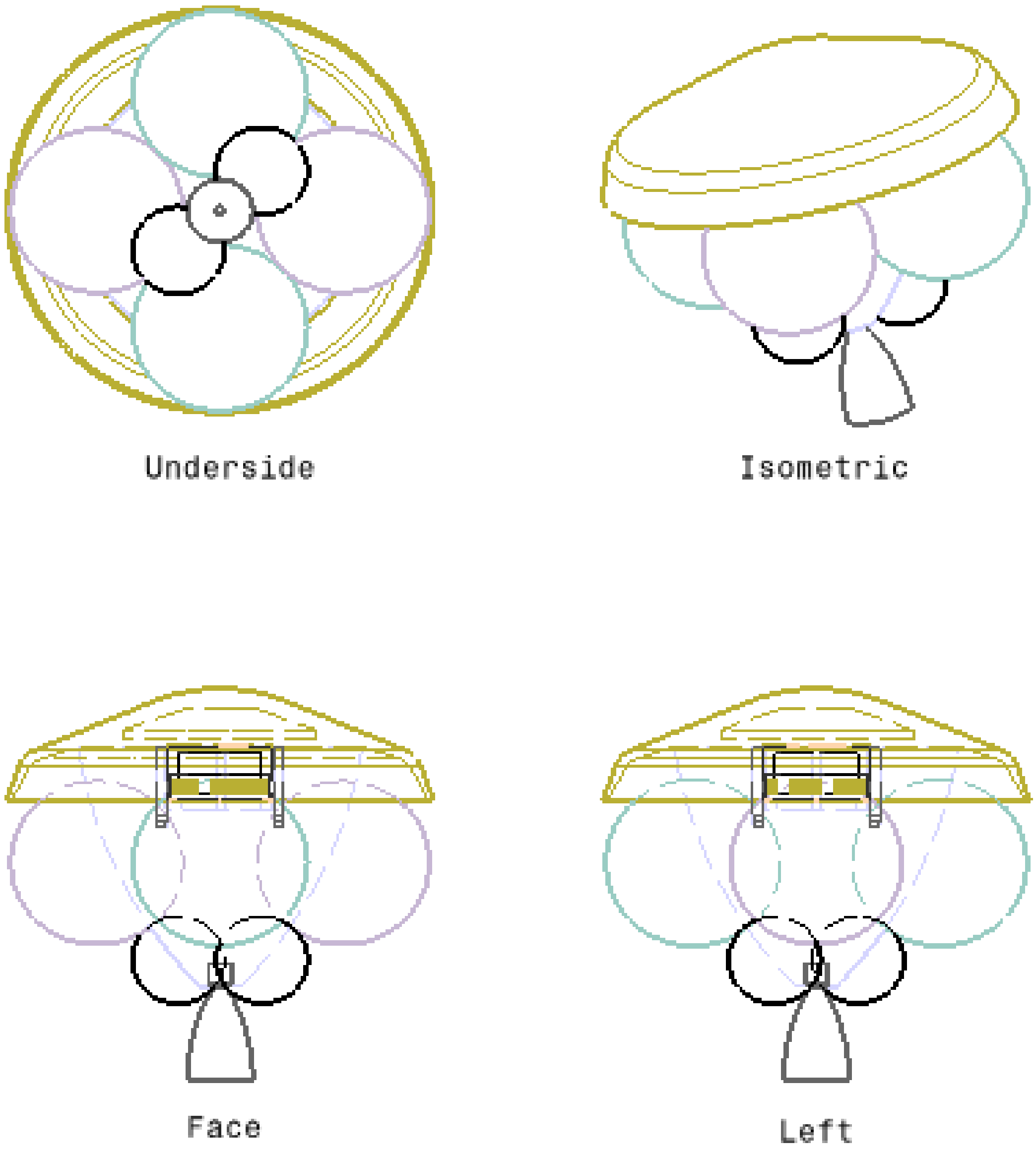
Sample Return Vehicle
Preliminary Requirements

1. Overview. This document outlines the preliminary requirements for the Sample Return Vehicle (SRV) in support of the Lunar Exploration Transportation Systems (LETS). The SRV's function will be to return a sample of lunar regolith safely to Earth. As the design of the LETS progresses throughout the semester, additional team-based requirements could be added to the SRV design.

2. Performance. The SRV shall be designed for the following performance requirements.
   2.1. Payload. The SRV shall deliver a minimum of one (1) kilogram of lunar regolith back to Earth. Additional delivery capability will be considered positive.
   2.2. Trajectory Course Maneuver System. The SRV shall provide a trajectory course maneuvering system that provides a change in velocity (ΔV) of 150 m/sec for midcourse trajectory correction from the moon to Earth.
   2.3. Primary Propulsion System. The SRV shall provide a primary propulsion system that provides a change in velocity (ΔV) of 2650 m/sec for initial liftoff from the lunar surface.

3. Environments. The SRV shall be designed to withstand the environments for launch, both cruise phases (trans-lunar injection and trans-earth injection), landing, lunar surface, and atmospheric reentry.
   3.1. Launch. The SRV shall be designed to withstand the Atlas V-401 launch environment per the Atlas Launch System Mission Planner's Guide, Rev 10a, January 2007, CLSB-0409-1109.
   3.2. Cruise phase. The SRV shall be designed to withstand the environment (e.g., radiation, thermal, and micrometeoroids) during the cruise phase for 28 days TLI and 28 days TEI.
   3.3. Landing. The SRV shall be designed to withstand the environment for landing.
   3.4. Lunar surface. The SRV shall be designed to withstand the lunar surface environment (e.g., temperature, radiation, micrometeoroids, dust) for the duration of the operational concept of each individual team.
   3.5. Atmospheric reentry. The SRV shall be designed to survive atmospheric reentry, estimated to be between 1-2 km/sec.

4. Interfaces. The SRV shall be designed to interface with the LETS team designs.
   4.1. The SRV shall be designed to interface with the individual team lander and mobility concepts, noting the volumetric shroud constraints of the Atlas V-401.



DESIGNED BY: Sebastien Bouvet
DATE: 04/11/2008
CHECKED BY: Julie Monszajn
DATE: 04/11/2008
SIZE: A4

Sample Return Vehicle

ESTACA IPT Team D

SCALE	WEIGHT (kg)	DRAWING NUMBER	SHEET
1:15	200	01	1/1