POLITECNICO DI TORINO

Department of Mechanical and Aerospace Engineering

Master’s degree in Aerospace Engineering

Master’s degree Thesis

Lunar Nano Drone for a mission of exploration of lava tubes on the Moon: Propulsion System

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December 2020

ACKNOWLEDGMENTS

Vittori, Bellomo, Maggiore e Messidoro, Pina, Gi, Ste Mauri, famiglia e amici, colleghi.

Thanks to General Astronaut Roberto Vittori, who conceived and initiated this project. Meeting him personally and sharing a half day with him has been an exciting experienced, hoping it won't be the last this way.
Thanks to Engineer Bellomo, competent and expert despite his young age, funny and passionate, who has demonstrated an absolute availability and helped me a lot during the development of this thesis.
Thanks to Engineer Messidoro, who has made his great experience available e gave me the possibility to meet several people, all accomplished in space sector.
Thanks to Professor Maggiore for sharing with me the possibility to carry this thesis out and for the lectures he gave. He is, without any doubt, one of the best teachers I have ever met in my whole career as a student at any level (well, maybe he is the best, actually).

Thanks to my two roommates, with whom I have shared my whole academic path, despite not without difficulties.
Thanks to all the friends of mine, who have always supported me, especially Simo, Matte, Cami, Hiere, the Pore, Chè, the Fedes, coach Cappitti and many others. I would need an entire page to name them all.
A particular and unusual thank to Stefano Mauri, with whom I periodically had an exchange of stupid puns: he deserves this.

Thanks to my physics high school teacher, who have enhanced and cultivated my passion for science. She has always given me precious advice.
Thanks to my neighbours Mimmo and Laura, who have always treated me like a grandson and let me relax spending time playing their beautiful piano.
Thanks to Gianni, the landlord, who has always been generous, honest, cordial, kind and all the best things one can imagine.

Thanks to whom of my family has always supported me.
Thanks to my additional aunt, Sardy and to my additional brother, Gi.
Thanks to my girl, who, despite we have known each other for a short while, cares about me and my degree more than I do.

Thank you all

ABSTRACT

In the last years, the idea of a return on the Moon has become more and more popular all over the world, in a special way for NASA's dedication, which led to the definition of the Artemis program. The Moon can potentially offer, looking at the long term, various resources for future missions or even for the Earth, as well as a unique environment to conduct scientific experiments.

ESA and NASA are currently maintaining a dialogue, which emphasises the Gateway station, while on the 29th October 2020 ASI and NASA signed an agreement which sets the bases for a long-life bilateral cooperation in the above-mentioned Artemis program, particularly for activities on the lunar surface. The LuNaDrone mission would perfectly fit for this scenario, to contextualise combined initiatives like shelters or foundation habitats. Furthermore, it would represent an incentive for Italian SME and start-up with a view on the New Space Economy, as well as for the academia.

This thesis focuses on a preliminary study of a mission that aims to the exploration of lunar caves by means of a drone, able to fly autonomously. These could be noticeably of interest, because they could potentially represent shelters for human life on our satellite, guaranteeing protection from radiations.

The beginning of the thesis introduces an overview about the past and actual lunar exploration, in which LuNaDrone mission would fit, followed by another overview about the mission itself and the spacecraft to be designed. These two sections are in common with two colleagues, who then treat other subjects separately.

The remaining parts are the presentation of the study about the propulsion system hypothesised for the spacecraft, as the specific subject of this thesis. The aim is to motivate the choices and to understand which parameters have the major impacts on the whole system in terms of mass and encumbrances. The methods utilised are the research of the state of the art (available items on the market or in development, favouring COTS items) and preliminary calculations for a rough sizing of the components. Taking into account the requirements of the mission, the final step is composed of a confrontation and an iteration among the colleagues in order to obtain information about the actual feasibility of the mission.

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Nomenclature

A_e	Exit area
A_t	Throat area
I_{sp}	Specific impulse
T_c	Combustion chamber temperature
c^*	Characteristic velocity
g_0	Standard sea level acceleration of gravity
$\dot{m}$	Mass flow rate
p_c	Combustion chamber pressure
r_e	Exit radius (nozzle)
r_t	Throat radius (nozzle)
θ_e	Final parabola angle (nozzle)
θ_i	Initial parabola angle (nozzle)
$\mathcal{M}$	Molar mass
$\mathcal{R}$	Universal gas constant
Γ	Correct flow rate when Mach number equals one
ΔU	Finite variation of internal energy [J]
F	Thrust
GOX	throughput, mass flow per unit area (catalyst)
L	Work [J]
Q	Heat [J]
R	Specific gas constant
c	Effective exhaust velocity
p	Pressure (general)
r	Radius (general)
α	Half angle (nozzle)
γ	Specific heats ratio
ε	Expansion ratio
λ	Corrective factor (nozzle)
σ	Stress

ACS	Attitude Control System
AND	Ammonium Dinitramide
ASI	Agenzia Spaziale Italiana (Italian Space Agency)
CLPS	Commercial Lunar Payload Services
COTS	Commercial off-the-shelf
ECSS	European Cooperation for Space Standardization
ESA	European Space Agency
GN2	Gaseous Nitrogen
GPIM	Green Propellant Infusion Mission
GRAIL	Gravity Recovery and Interior Laboratory
HAN	Hydroxyl amine nitrate
JAXA	Japan Aerospace Exploration Agency
LOS	Line of Sight
LRO	Lunar Reconnaissance Orbiter
LuNaDrone	Lunar Nano Drone

MIL-STD	Military Standard
NASA	National Aeronautics and Space Administration
RP-1	Rocket Propellant-1 / Refined Petroleum-1
SELENE	Selenological and Engineering Explorer
SLIM	Smart Lander for Investigating Moon
SME	Small and Medium Enterprises
SoA	State of the Art
TBC	To be Confirmed
TBD	To be Defined
VIPER	Volatiles Investigating Polar Exploration Rover

1 Context and objectives of the study

1.1 Mission statement and objectives

Mission statement:

"To design a flying drone able to autonomously hover inside a lava tube entering a skylight, taking photos and mapping the internal surface. The concept of the drone is based on the strategic idea already experienced with Cubesats: small spacecraft, low mass and cost, standardised, affordable by Academies and SME's and easy to deliver in Space"

The primary and secondary objectives of the mission have been deduced from the mission statement and are:

Primary objectives:

- To design an autonomously flying drone
- To explore and photograph lava tubes

Secondary objectives:

- To map the interior of lava tubes
- To develop a low cost and low mass drone concept to be standardised

1.2 Observational evidence and characteristics of lunar lava tubes

The chance for the actual existence of lunar lava tubes was postulated during the early 1960's. In 1971, a study concerning the possible presence of lava tubes in the Marius Hills region was published, with the evidence of rilles on the surface [1].

In 2009, the 10 m–resolution images taken by the Terrain Camera (TC) aboard SELENE (nicknamed Kaguya) showed three huge vertical holes in the lunar Marius Hills, Mare Tranquillitatis and Mare Ingenii. The holes have aperture diameters and depths of several tens of metres to one hundred metres [2]. Haruyama et al. (2009) [3] hypothesised that they are possible “skylights” opened on subsurface caverns such as lava tubes.

In 2010, NASA’s Lunar Reconaissance Orbiter (LRO) photographed the skylight in the Marius Hills in more detail, showing both the 65-metre-wide pit and the floor of the pit about 36 metres below [4]. The LRO has also imaged over 200 pits that show the signature of being skylights into subsurface voids or caverns, ranging in diameter from about 5 m (16 ft) to more than 900 m (3,000 ft), although some of these are likely to be *post-flow features* rather than volcanic skylights [5].Figure 2 shows some examples.

In 2011, NASA launched GRAIL, whose purpose was to evaluate the gravity field of the Moon in order to detect its internal structure. It also made it possible to confirm the presence of lava tubes underneath the surface. To this purpose, a skylight of 65 metres in diameter and 80-85 metres deep, with a roof thickness of 20-25 metres has been proven to exist. It is located in the Marius Hills region and occurs in a shallow rille-like trough about 400 metres wide and 300-400 metres deep. It is expected to hide a large cavern beneath the visible surface that extends about 60 km to the west of the skylight, where the cavern itself is approximately 30 km in length [6].

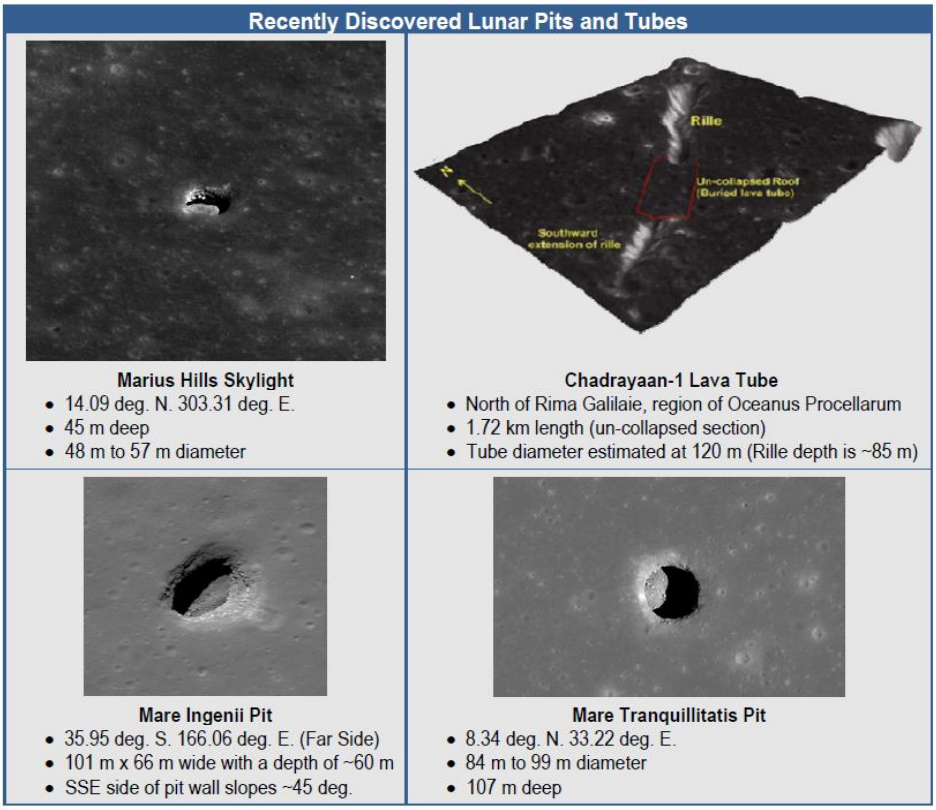


Table 1 High resolution imaging confirms existence of cavernous lunar sub-surfaces [8]

Again in 2011, a study conducted on Chandrayaan-1's observations, an Indian lunar spacecraft, showed a buried, un-collapsed and near horizontal lava tube in the vicinity of Rima Galilaei [7].
A remnant of the volcanic tube, whose roof has capsized and created a valley is named a "rille". It may happen that the roofs of such tubes do not collapse and remain intact, with a hollow interior in most cases. A skylight is a lava tube ceiling collapse potentially providing a means of entrance into the tunnel. Figure 1 illustrates typical entrance possibilities which may be encountered for the pits and tubes identified in Table 1. More details about Mare Tranquillitatis Pit are reported in Figure 3.

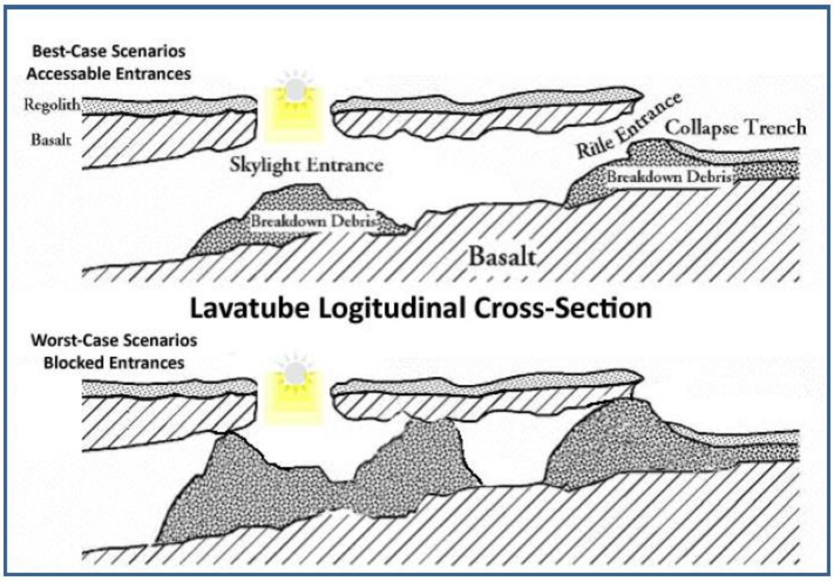


Figure 1 Potential entrance outcomes from observed pits and rilles

Scientifically speaking, to map the distribution and age of bedrock at the surface, investigations for understanding the geological processes associated with ancient lunar basaltic lava flows are needed.

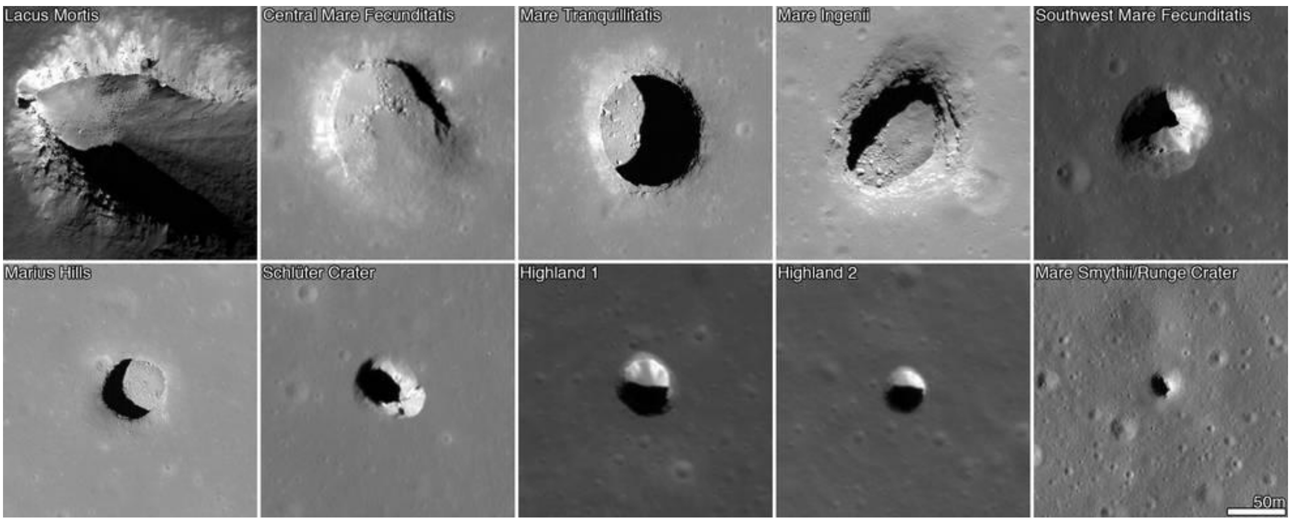


Figure 2 Images from NASA's LRO spacecraft

Figure 2 shows all the known mare pits and highland pits. Each image is 222 metres wide [8].

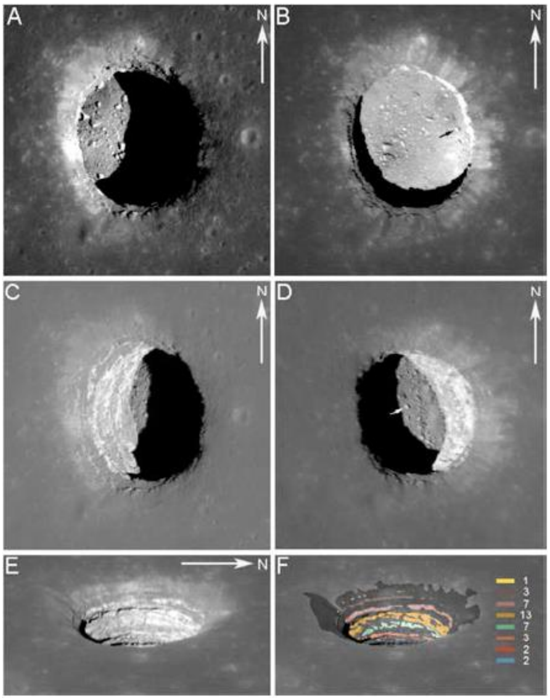


Figure 3 Mare Tranquillitatis pit [8]

Figure 3 shows various views of Mare Tranquillitatis Pit[1].
Lava tubes tend to have smooth floors, with possible "soda straws" stalactites formed by lava dipping from the ceiling. Due to the lesser gravity, lava tubes on the Moon may be much larger in diameter than those found on Earth [9]. These caverns would be suitable for human habitation, because they could provide safety from hazardous radiations, micro-meteoritic impacts, extreme temperatures and dust. However, polar regions are interesting as well, because they appear to have abundant ice water. According to a new discovery presented at NASA's Lunar Science for Landed Missions Workshop, it appears that there is a location on the Moon that merges both aspects: a possible lava tube that is located in the northern polar region (Philolaus Crater) [10].

For what concerns radiation protection, the thickness of the roof of the caves is expected to be tens of metres [7], that would certainly be advantageous, but, on the other hand, it would represent an obstacle for the design of a communication system between the inside of these caves and the surface.

On the surface of the Moon the fluctuations of the temperature are extremely wide, whereas the interior of the caves is expected to maintain an almost constant temperature around -20°C. As a result, this aspect would greatly ease the design of the thermal control system of all those devices that would operate in this environment.

[1] (A, B) show two near nadir images with opposite Sun azimuth angles. Both images are approximately 175 m wide. Oblique views: (C) layering in west wall and a portion of pit floor beneath overhanging mare (29° ema); (D) A significant portion of the illuminated area is beneath the eastern over hanging mare in this image (26° ema), white arrow indicates same boulder marked with black arrow in B. Detailed layering is revealed in (E) and (F). Outcropping bedrock layer thickness estimates are presented in (F) in metres, ±1m.

1.3 State-of-the-art of mission concepts

Mission architecture usually includes the number of robotic entities and their roles (i.e. probes, landers…), their approximate mass (which has implications on the traditional space mission architecture components of launch vehicle and trajectory), the methods of communication, the power strategies employed, and the concept of operations. Multi-mission architectures are also possibilities for skylight and cave exploration. One such multi-mission architecture would be broken into three phases, the first phase being the flyover and surface investigation of a skylight and deployment of a sensor package to a skylight entrance. The second phase could send mobile robots in to explore lava tubes or cave network. The third phase could include deliver of habitats, robots and personnel with specialised scientific instruments. A reference set of mission goals can be defined in order to compare mission architectures. Those goals are to enter a lava tube cave via skylight, to explore it and to send data [9].

1.4 Challenges

From several tests on Earth, it is known that ground penetrating radar often fails to detect lava tubes especially if the lava was deposited in multiple flows. This is because of the partial reflection of the radar at interfaces between layers of material, caused by repeated lava flows.

From a scientific perspective, in many cases it may be sufficient to get beyond the "twilight zone" (the transition between areas illuminated for some period during the day and areas of constant darkness) to define the distance to be travelled inside a cave. This region is likely to be indicative of the variation of different significative parameters, such as potential to support life, volatile contents and geological features, impacted by sunlight, temperature variations or rock fall during skylight formation.

Moreover, some scientists believe that using propulsive vehicles may lead to possible problems, such as the contamination of volatiles trapped at the bottom of a skylight or even the death of living organisms inside a cave. Additionally, there may be the possibility of contaminating scientifically important sites with that strategy.

The main issues to cope for planetary cave exploration are:

- access to the cave
- in-cave mobility
- data collection and processing
- power sources
- communication

Spacecraft configuration has a large impact on how these issues have to be managed. As an example, the lack of solar power underground may put large limitations on how the spacecraft could move. Energetically, it does not make sense to carry the propulsion system required for landing along for further cave explorations activities. Tethered solutions may also be considered.
Note that modelling in lava tubes requires active sensing and due to the expected larger size of lava tubes on the Moon, sensors in this environment must have long range, which requires increased power. To this purpose, technologies like active sensing could provide a physical barrier to miniaturization [9].

1.5 Summary

Hybrid propulsive configurations may be considered. External tethered enlightenment or power source systems may be considered. Wireless power and data transmission within LOS of the tethered communication node would eliminate the need for exploration robot to physically reach it, which is critical in unpredictable environments. Combination of active sensing (good for shadowed regions but lower resolution and range limited by power) and cameras (higher resolution but unable to determine 3D scale) required to build sufficiently detailed models for science and robot operations. Commercial magneto-inductive communications system indicates an achievable data rate of 2412bps through rock. Magneto-inductive comm requires a large and heavy antenna. While it is a great technology for later use in cave operations, it may not be feasible for the first, lightweight robotic explorers [9].

For what concerns power and communication, extended periods without access to solar power, limited accessibility to communication and operating exclusively in a dark environment have to be taken into account. High energy density batteries would enable longer cave excursions with low battery masses [9].
Limited data link through rock can be achieved with very low-frequency radio or magneto-inductive comm. These technologies are under development terrestrially for cave and mining communication and rescue and have undergone significant advances in mass and power requirements over the past few years [9].

1.6 Possible supporting missions

Moon exploration will gain more and more interest in the next few years. To this purpose, many different companies are developing new landers and rovers to be launched. The study in this document is based on the idea that LuNaDrone will have to be carried to the proximity of an above-mentioned skylight by one of them. For instance, JAXA's lander named SLIM (Smart Lander for Investigating Moon), whose departure is planned for January 2022, will land in the vicinity of Marius Hills Hole, with an accuracy of about 100 metres, next to a lava tube [11].

For instance, it would be reasonable to take into consideration NASA's Commercial Lunar Payload Services (CLPS), an initiative which allows rapid acquisition of lunar delivery services from American companies for payloads that advance capabilities for science, exploration or commercial development of the Moon. Investigations and demonstrations launched on commercial Moon flights will help the agency study Earth's nearest neighbour under the Artemis program. Moreover, NASA has identified agencies and external science payloads that will fly on future CLPS missions, including the Volatiles Investigating Polar Exploration Rover (VIPER). Future payloads could include other rovers, power sources, and science experiments, including the technology demonstrations to be infused into the Artemis program [8]. NASA has chosen Astrobotic, Intuitive Machine and Masten to take part to CLPS programme.

Hakuto-R is the program name for iSpace's first two lunar missions, a commercial initiative with the purpose to demonstrate the capability to softly land and release a rover. It will lead to various subsequent high-frequency, cost-effective missions to establish a payload delivery system to the Moon [12].

2 LuNaDrone mission design

2.1 Mission overview

The mission concept is based on the assumption that a lander and/or a rover would deploy LuNaDrone in the proximity of a selected skylight, at a maximum distance of TBD metres from the waypoint from which the vertical descent phase will start. That distance shall be lately in subsequent iterations decided according to the physical dimensions of the crater to be explored and the performances/architecture of the spacecraft.

After the deployment, LuNaDrone shall be able to conduct at least one autonomous flight. It consists of different phases: take off, climb, hover, horizontally translate to reach the skylight, descend into the lunar pit and come back to the initial point following the same trajectory.

Because landing hazard avoidance was not prioritised, the flight will start and end within an area to be pre-inspected and determined to be safe in terms of obstacles and ground slope.

While hovering, LuNaDrone has to be able to deal with disturbance of the flight and maintain its stability. In addition, a plan about when and how to implement the acquisition of the images has to be carefully developed. During the acquisition, LuNaDrone would need to enlighten the subject.

LuNaDrone has to be able to either store and/or forward the images to the rover/lander.

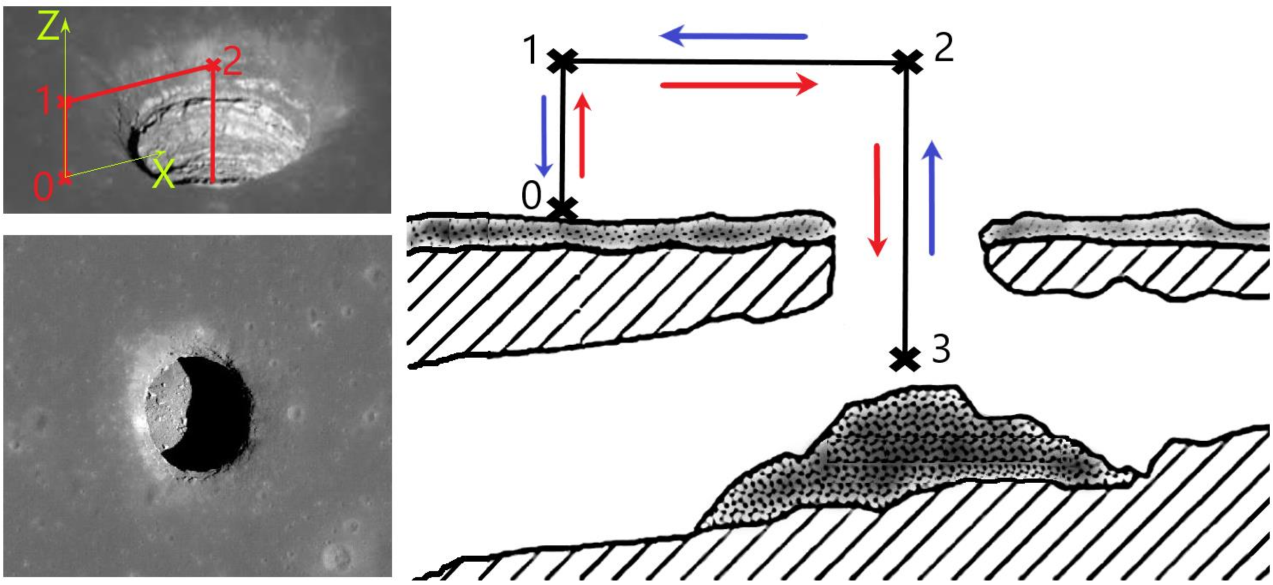


Figure 4 Illustration of the flight segments

A possible flight of the LuNaDrone might be the one shown in Figure 4. The first step, identified by the numbers 0-1, refers to a vertical ascent manoeuvre in which the spacecraft rises from the lunar surface and reaches a predetermined altitude. It will then follow the horizontal translation manoeuvre, where the spacecraft will cover a certain distance along the X-axis without changing its altitude. The last manoeuvre, identified by the numbers 2-3, refers to the descent segment where the spacecraft will decrease its altitude until it stops at point 3. It is assumed that the LuNaDrone will reach points 1, 2 and 3 with zero residual velocity and, if required, it will have to hover at these points for a predetermined time frame before moving on to the next flight segment. After acquiring the photos of the inside of the lunar pit, LuNaDrone will have to come back to the initial point, following the same trajectory.

Code	Functional and Performance Requirements
R1	The LuNaDrone shall be able to autonomously depart from the surface of a rover, hover, enter in a target Lunar Cave and exit at the end of the mission returning to the rover (TBC)
R2	The LuNaDrone shall be equipped with a propulsion system able to support the entire mission
R3	The LuNaDrone shall be able to withstand a travel time of at least TBC min
R4	The LuNaDrone shall be able to withstand a travel distance of at least TBC Km
R5	The LuNaDrone shall be able to take images of the Cave, store them on-board and transmit them to the Rover and/or Lander at the end of the mission or as soon as possible (TBC)
R6	The LuNaDrone wet mass shall be less than 15 Kg (TBC)
R7	The volume of the LuNaDrone shall be less than 30 X 20 X 20 (TBC) cm while in stowage in the ROVER
	Interface Requirements
R8	The LuNaDrone vehicle shall be able to autonomously depart from the surface of a rover, hover and return to its base on top of the rover.
	Environmental Requirements
R9	The LuNaDrone shall be able to withstand environment (day/night) of the Moon site and Moon Cave (TBC)
R10	The LuNaDrone shall be able to withstand the launch and transport to the Moon environment in stowed conditions (TBC)
	Operational Requirements
R11	The LuNaDrone shall be able to take pictures in visual wavelengths (TBC).
R12	The LuNaDrone shall be able to fly autonomously by means of a pre-programmed flight sequence
	Implementation requirements
R13	The LuNaDrone shall make use of non-toxic propellants that are safe to handle on ground
R14	The LuNaDrone functional simulator shall be able to show the vehicle functional architecture and simulate the mission
R15	The LuNaDrone 3D-printed scaled model shall be able to show the vehicle physical architecture and the technology critical components

Table 2 Mission requirements

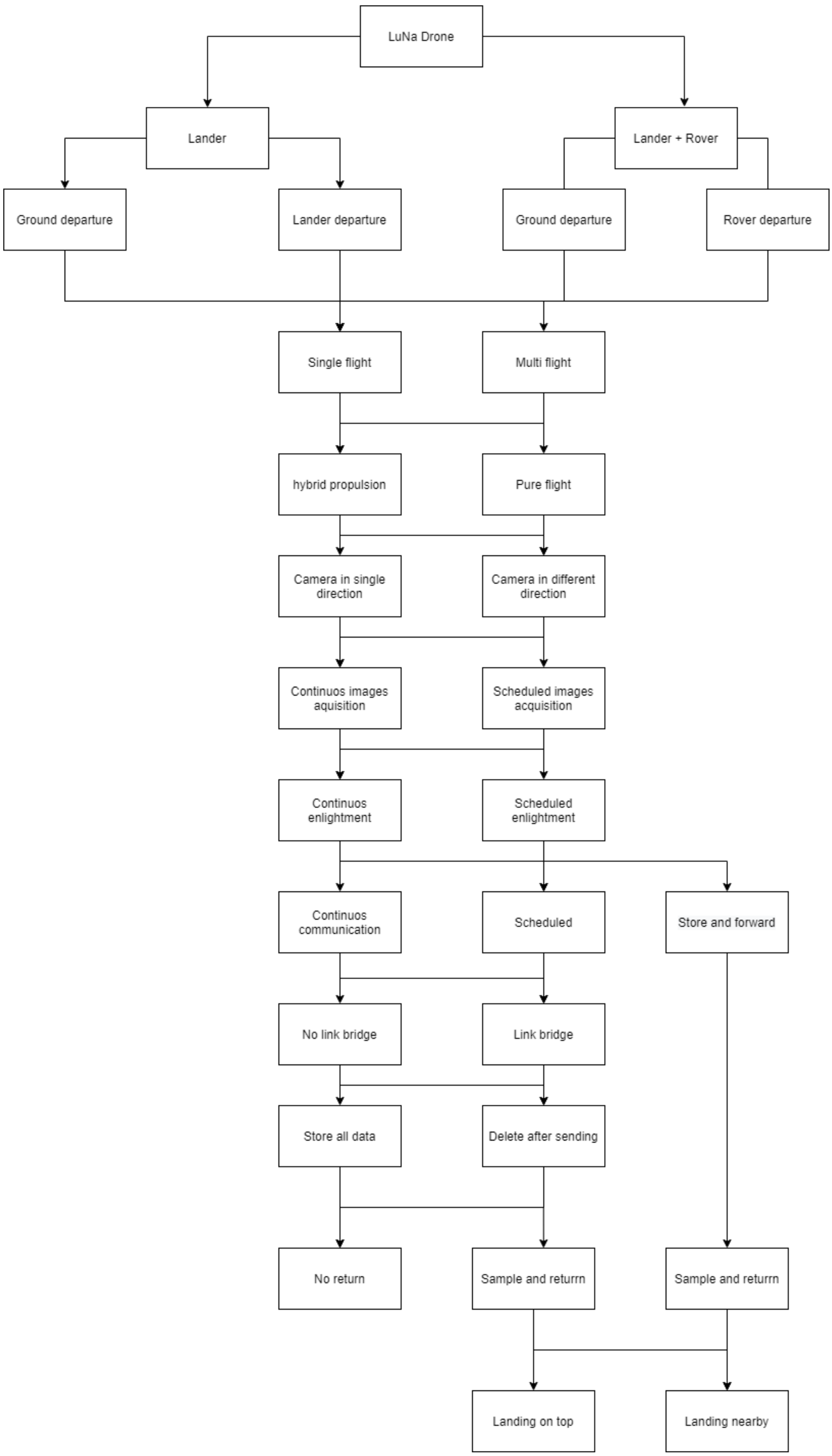


Figure 5 Brainstorming on the possible scenarios of the mission

ID	Subject	Mission contingencies	Comment
MC-001	Lander	The lander must land no more than TBD meters from the skylight.	The maximum distance from the skylight is determined by the flight autonomy of the LuNaDrone.
	Lander + rover	The rover must be able to approach at least TBD meters from the skylight.	In this case the minimum distance from the skylight will be reasonably shorter than in the "Lander" case.
MC-002	Ground departure	The LuNaDrone will first be released to the lunar surface and then take off.	
	Rover/Lander departure	The LuNaDrone will take off directly from the lander/rover	
MC-003	Continuous communication	The LuNaDrone must be able to communicate continuously.	The LuNaDrone will communicate either with the lander/rover, or with a lunar satellite or lunar gateway or directly with the ground station.
	Scheduled communication	The LuNaDrone has to communicate only when necessary	
	Store and forward	The LuNaDrone will communicate mission data once it emerges from the lava tube.	
MC-004	Sample and return	The LuNaDrone will be able to return to the lander/rover.	
	No return	The LuNaDrone will not necessarily be able to return to the lander/rover.	In this case, continuous communication must be guaranteed.
MC-005	Flight	The LuNaDrone must be able to explore the hole by flying.	
	Hybrid propulsion	In addition to flying, the LuNaDrone must be able to move on the surface with a more efficient propulsion.	In this case, if the terrain conditions allow it, it would be easier to meet the travel distance requirements.
MC-006	Landing on top of the lander/rover	The LuNaDrone must be able to land on top of the lander/rover.	
	Landing nearby	The LuNaDrone must be able to land at a maximum distance of TBD meters from the lander/rover.	
MC-007	NO link bridge	The LuNaDrone shall be able to communicate without LOS through the rocks	
	Link bridge TBD	The LuNaDrone shall be able to communicate in LOS with a TBD link bridge	In this case, bridge drones/bridge antennas shall be placed in strategic locations
MC-008	Camera/cameras in single direction	The LuNaDrone shall be able to take images only towards its movement	Consider that camera/cameras shall be placed far from the propulsion system
	Camera/cameras in different directions	The LuNaDrone shall be able to take images in all/different directions	
MC-009	Single flight approach	The LuNaDrone shall depart only once, do its operations and land only once.	Consider what is best for calibration of instruments, reference for navigation and propulsion/EPS performance
	Multi-phase flight approach	The LuNaDrone shall depart and land TBD times, with scheduled and programmed acquisition plan.	
MC-010	Store all data	The LuNaDrone shall be equipped with sufficient memory to save all the mission data	
	Delete stored data after sending	The LuNaDrone shall not preserve data after sending	
MC-011	Continuous images aquisition	The LuNaDrone shall continuously use cameras	TBD in accomplishment with communication system and system performances
	Scheduled images acquisition	The LuNaDrone shall use cameras following a predicted plan	
MC-012	Continuous enlightment	The LuNaDrone shall continously enlight its way for the cameras	See MC-011
	Scheduled enlightment	The LuNaDrone shall enlight following cameras needs	

Table 3 Mission contingencies

2.2 Spacecraft overview

The main objective of the mission is to acquire images of the inside of lava tubes on the Moon. At this purpose, the system of cameras which will be designed and utilised is of fundamental importance. In particular, the number of cameras, their resolution and their positioning will have to be discussed in order to find the best solution in terms of mass, volume and compatibility with the other subsystems. The propulsion system includes one hydrogen peroxide monopropellant rocket as main engine and at least other eight for the ACS. The spacecraft will obtain its necessary electrical power from lithium primary batteries. The above-mentioned subsystems are strongly linked to the flight profile development, which can state how efficient a manoeuvre is and the angle of inclination of the spacecraft for its movements, which in turn gives information again to the navigation and propulsion systems design.

Subsystem/object	Type
Navigation	IMU+Visual Navigation, IMU+LiDARs
Image acquisition	One 12 Mpx camera, 120° FOV, 15 fps
Propulsion	Hydrogen peroxide 92% wt monopropellant rockets
Electrical power source	Lithium primary batteries
CommSys	X-band 8 GHz, 40-50 Mbit/s

Table 4 Main characteristics of the actual configuration of the spacecraft

The drafting of Chapter 1 and Chapter 2 was carried out in collaboration with two colleagues from Politecnico di Torino: Stefano Pescaglia [13] and Gael Latiro [14], who were respectively responsible for developing the flight profile (as well as identifying possible electrical power sources), and studying the navigation system of spacecraft.

Here below some images of the current configuration are presented. They do not represent a complete spacecraft and no detailed analysis for the compatibility of the subsystems have been carried out. The purpose of these images is to give a rough idea of how the room inside the spacecraft may be utilised, considering the requirements of the different components (e.g. the necessity of the LiDARs to have nothing to hinder their view).

Number	Component
1	LiDAR – 360° horizontal plane
2	Propellant tank
3	IMU
4	Lithium primary D-cells
5	OBC
6	LiDAR – vertical plane
7	LED + optics
8	Camera + optics
9	ACS thrusters
10	Pressurant tank
11	Engine

Table 5 Part numbers

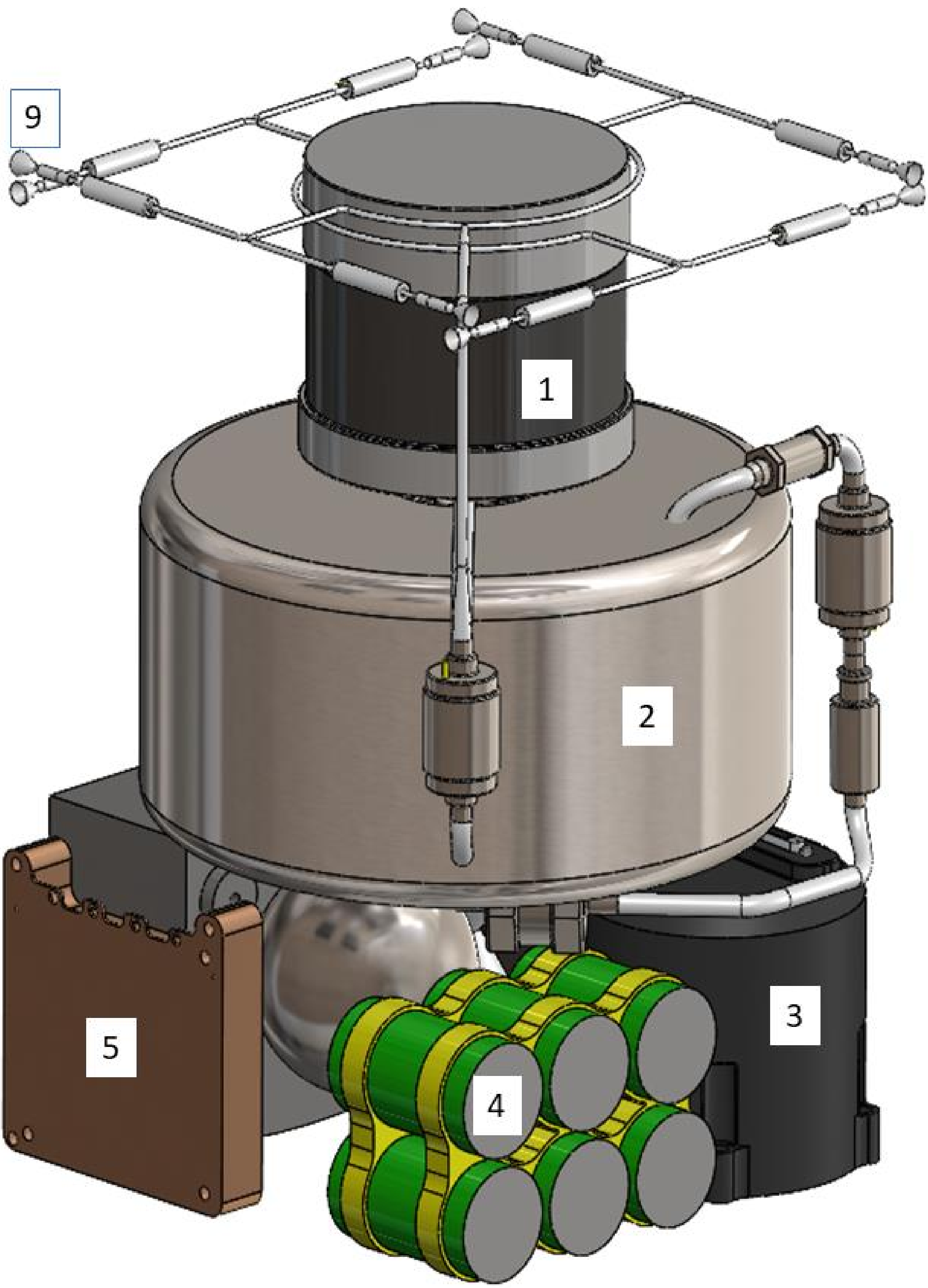


Figure 6 Side view #1 of the current configuration of LuNaDrone

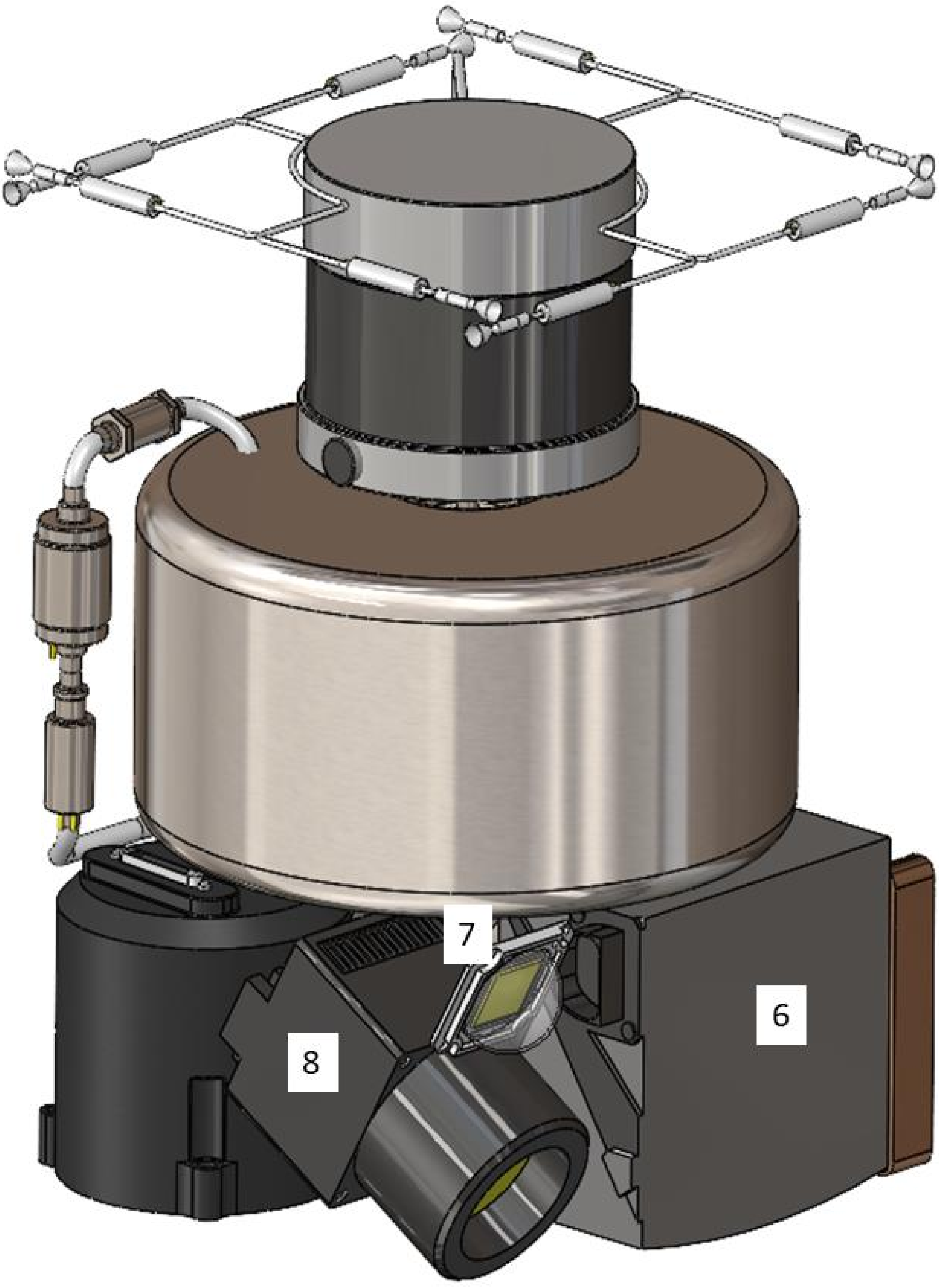


Figure 7 Side view #2 of the current configuration of LuNaDrone

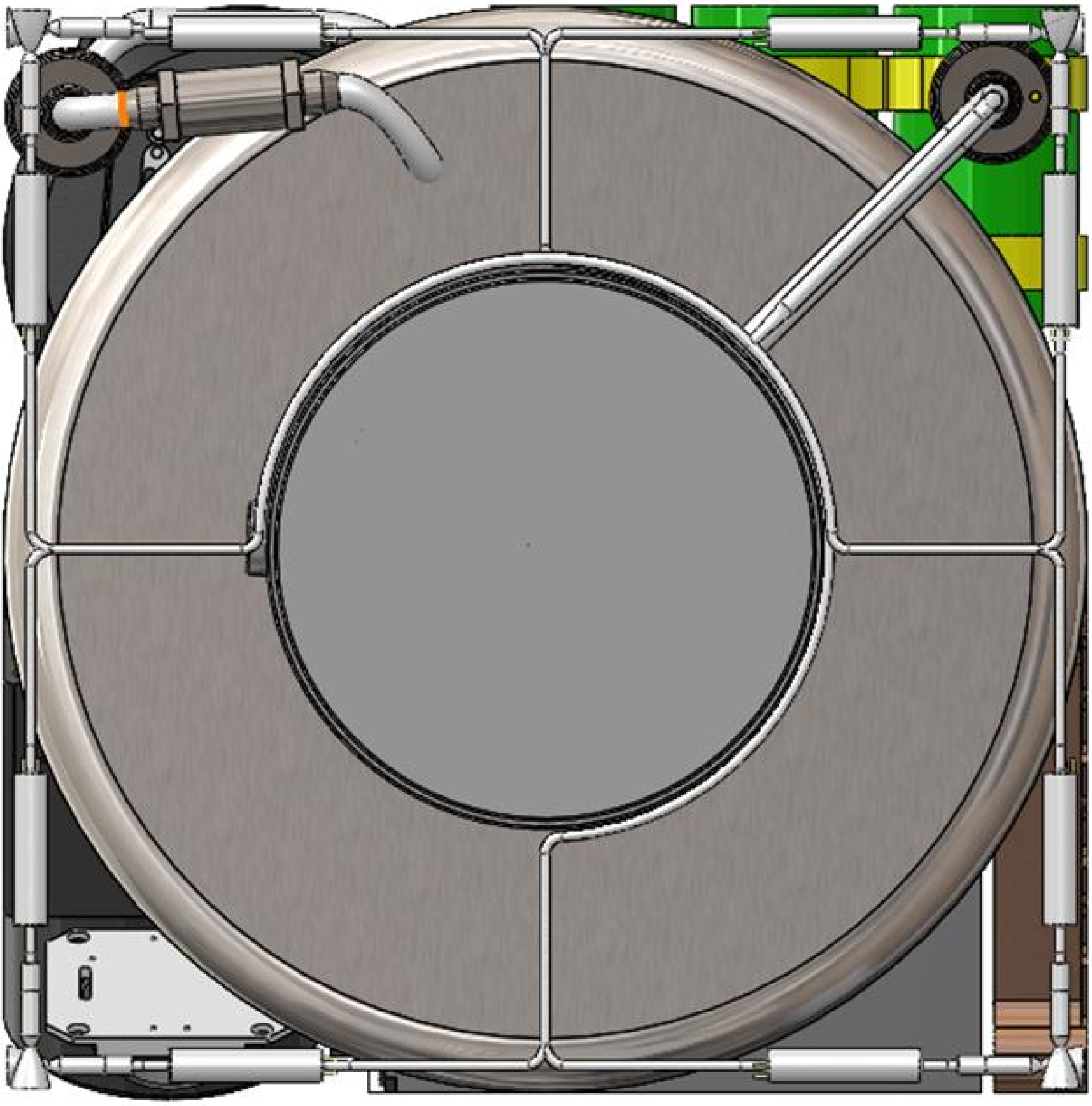


Figure 8 Top view of the current configuration of LuNaDrone

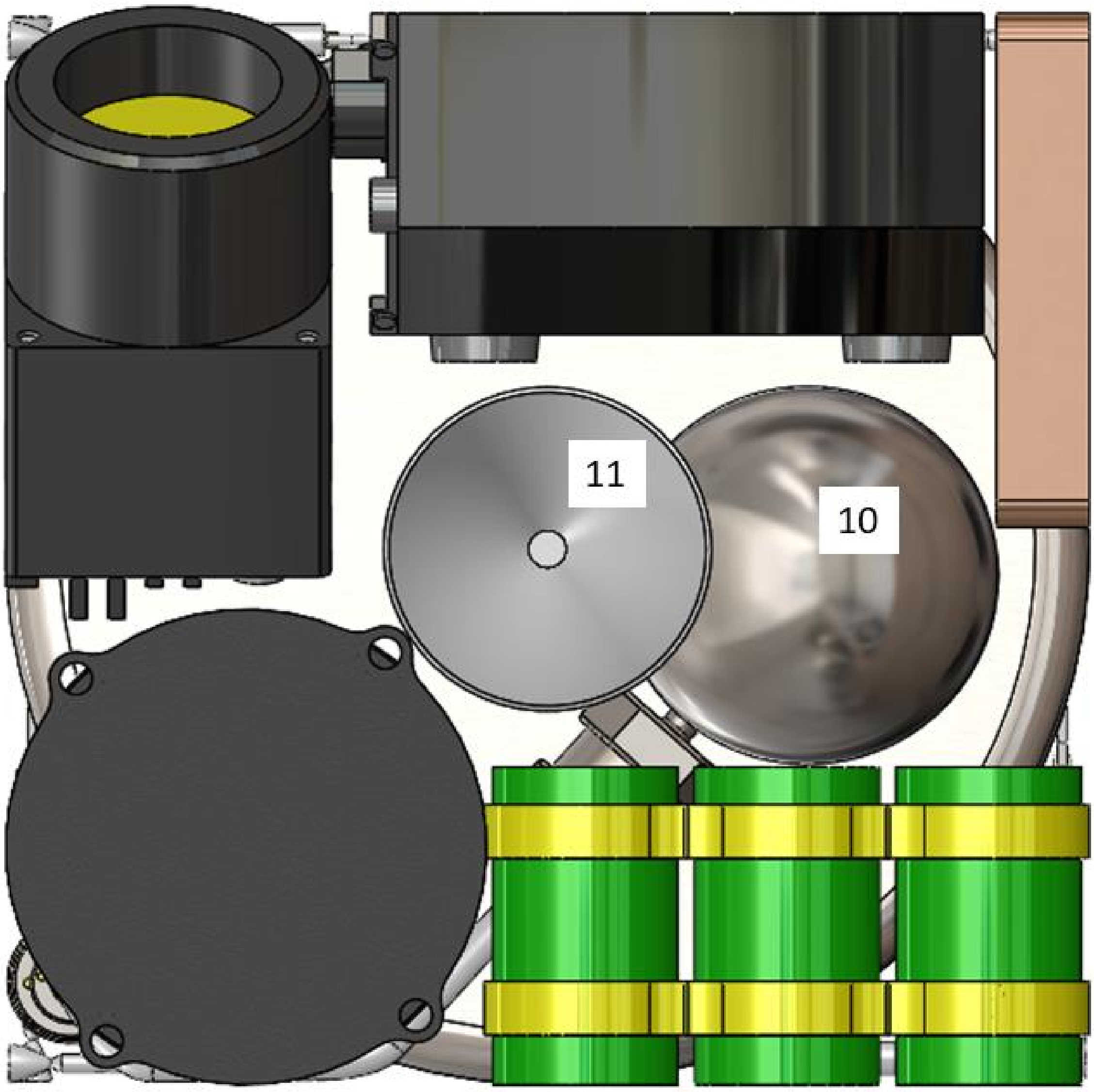


Figure 9 Bottom view of the current configuration of LuNaDrone

3 Propulsion system - Literature

3.1 Introduction

Depending on what types of propulsion one chooses to utilise, there are distinctive features of each of them, but also common aspects. In other terms, it is trivial to say that propellant has to be stored somewhere and somehow, independently of what it is. Likewise, a feed system composed by valves, tubes and other devices has necessarily to be implemented.
According to the requirements of the mission, the aim is to find an optimal solution capable of fulfilling what mission needs with the minimal mass and volume, but also aspects like cost and reliability have to be taken into account. Minimal mass and volume would mean miniaturised components, which are usually costly and require many hours of testing. So, talking about optimal solution is not only to be intended in terms of encumbrances, but of cost and availability as well. In fact, this is a sort of feasibility study, in which the present technology will be analysed as a first choice, where possible.

3.2 State of the art of Rocket Propulsion

In the last years, small spacecrafts have been more and more utilised both as a support to great missions and as main actors for complete missions. This has implied the progress of the technology over the years for all the subsystems, in particular miniaturised technology.

Product	Thrust	Specific Impulse (s)	TRL Status
Hydrazine	0.5 – 30.7 N	200-235	9
Cold Gas	10 mN – 10 N	40-70	GN2/Butane/R236fa 9
Alternative (Green) Propulsion	0.1 – 27 N	190-250	HAN 6, AND 9
Pulsed Plasma and Vacuum Arc Thrusters	1 – 1300 µN	500-3000	Teflon 7, Titanium 7
Electrospray Propulsion	10 – 120 µN	500-5000	7
Hall Effect Thrusters	10 – 50 mN	1000-2000	Xenon 7, Iodine 3
Ion Engines	1 – 10 mN	1000-3500	Xenon 7, Iodine 4
Solar Sails	0.25 – 0.6 mN	N/A	6 (85 m^2), 7 (35 m^2)

Table 6 Propulsion systems types for small spacecrafts [15]

As shown in Table 6,which is extracted from the NASA's state of the art for small spacecrafts [15], different types of propulsion systems are available. In this context, it is important to remember that LuNaDrone's mass is supposed to be about 15-20 kg, with a consequent 24-33 N of required thrust just to hover.
Although electric propulsion can provide extremely high values of specific impulse, it only reaches very low grades of thrust. In fact, it is commonly utilised for slow manoeuvres, like orbit transfers and corrections and station-keeping.
Cold gas propulsion could offer acceptable ranges of thrust, but also unacceptable values of specific impulse, which would mean the necessity to carry huge amount of pressurised gas, therefore increasing mass and encumbrances.
Solar sails show limited thrust and enormous encumbrances, once deployed.
Chemical propulsion is the unique possible choice, with relatively low values of provided specific impulse but with no issues related to the obtainable levels of thrust.

The new NASA's "State of the art of Small Spacecrafts Technology" should have been published in fall 2020, as written on their official site, but it had not been available yet when this thesis had been developed. However, no significant variations in Table 6 were expected to be seen. Surely, chemical propulsion would any way have been the only possible solution, due to the significantly higher thrust to power ratio.

- **Hydrazine rockets**

One of the most utilised solutions in compact propulsion systems is hydrazine-based monopropellant rockets. There are a significant number of mature systems implemented on large spacecrafts, because they are generally reliable and convenient in terms of mass and volume. Typically, such solutions incorporate double stage flow control valve to regulate propellant supply and a catalyst bed heater with thermal insulation. Since hydrazine rockets are widely used for large satellites, a robust ecosystem of components exist, so custom-designed systems for specific applications may be constructed with available components [15].

Product	**Manufacturer**	**Thrust (N)**	**Isp (s)**	**Status**
MR-103D	Aerojet Rocketdyne	0.28 – 1.02	209 - 224	TRL 7
MR-111C	Aerojet Rocketdyne	1.3 – 5.3	215 - 229	TRL 7
MR-106E	Aerojet Rocketdyne	11.6 – 30.7	229 – 235	TRL 7
1N Ariane	Ariane Group	0.32 – 1.1	200 - 223	TRL 7
20N Ariane	Ariane Group	7.9 – 24.6	222 - 230	TRL 7

Table 7 Hydrazine propulsion systems [16]

Table 7 shows five different examples of high TRL hydrazine rockets, with thrust from less than 1 N to more than 30 N. The specific impulse obtainable with hydrazine is usually between 150s and 250s, thanks to the great capacity to decompose by hydrazine, reaching high temperatures in the combustion chamber. Here it is clear that, as said before, hydrazine rockets have already been widely utilised and developed by many important manufacturers in the space sector.

Even MOOG has extensive experience in the design and testing of propulsion systems and components for large spacecrafts.

Engine	MONARC-1	MONARC-5	MONARC-22-6	MONARC-22-12	MONARC-90LT	MONARC-90HT	MONARC-445
Steady State Thrust	0.22 lbf (1N) @275 psia	1.0 lbf (4.5 N) @325 psia	5 lbf (22N) @275 psia	5 lbf (22N) @190 psia	20 lbf (90 N) @ 235 psia	26 lbf (116 N) @ 235 psia	100 lbf (445N) @ 275 psia
Feed Pressure	70 – 400 psia (4.8 – 27.6 bar)	80 – 420 psia (5.5 – 29.0 bar)	70 – 400 psia (4.8 – 27.6 bar)	70 – 400 psia (4.8 – 27.6 bar)	80 – 400 psia (5.5 – 27.6 bar)	80 – 370 psia (5.5 -25.5 bar)	70 – 400 psia (4.8-27.6 bar)
Nozzle Expansion	57:1	135:1	60:1	40:1	40:1	50:1	50:1
Valve Power	18 watts	18 watts	30 watts	30 watts	72 watts	72 watts	58 watts
Mass	0.83 lbm (0.38 kg)	1.08 lbm (0.49 kg)	1.58 lbm (0.72 kg)	1.51 lbm (0.69 kg)	2.47 lbm (1.12 kg)	2.47 lbm (1.12 kg)	3.5 lbm (1.6 kg)
Engine Length/Exit Diam	5.2 in (13.3 cm) / .2 in (0.5 cm)	9.4 in (41.8 cm) /1 in (2.5 cm)	8 in (20.3 cm) / 1.5 in (3.8 cm)	9 in (22.9 cm) / 1.2 in (5.3 cm)	12 in (30 cm) / 3.3 in (8.4 cm)	12 in (30 cm) / 3.3 in (8.4 cm)	16 in (41 cm) / 5.8 in (14.8 cm)
Specific Impulse	227.5 sec	226.1 secs	229.5 secs	228.1 secs	232.1 secs	234.0 secs	234.0 secs
Minimum Impulse Bit	0.0006 lbf-sec (2.6 mN-sec)	0.0007 lbf-sec (3.1 mN-sec)	0.07 lbf-sec (312m N-sec)	0.12 lbf-sec (526m N-sec)	0.04 lbf-sec (1.8 N-sec)	0.26 lbf-sec (1.16 N-sec)	2.59 lbf-sec (11.52 N-sec)
Total Impulse	25,000 lbf-sec (111,250 N-sec)	138,000 lbf-sec (613,852 N-sec)	120,000 lbf-sec (533,784 N-sec)	263,720 lbf-sec (1,173,085 N-sec)	786,000 (3,500,000 N-sec)	459,100 lbf-sec (2,042,178 N-sec)	1,250,000 lbf-sec (5,600,000 N-sec)
Pulses	375,000	205,000	230,000	160,000	50,000	70,000	12,000

Table 8 Performance characteristics of MOOG's MONARC rockets series

Table 8, which is taken by one of MOOG's datasheets available on their sites [17], show MONARC rocket series, which has already a long successful heritage, with a range of applications that includes Earth observation and communication, space exploration and missile defence. As it can be seen, the level of available thrust is wide, from less than 1 N to more than 400 N. Pressure feed is usually under 30 bars up to 5 as lower limit, when the engine can no longer work properly. Specific impulses are again around 230s. Encumbrances are around one kilogram on mass and 30 cm of length.

Product	Manufacturer	Thrust (N)	Isp (s)	Status
MT-9	IHI AeroSpace	0.29 – 1.13	208 - 215	Flight proven
MT-8A	IHI AeroSpace	1.80 – 5.01	212 - 225	Flight proven
MT-2	IHI AeroSpace	6.9 – 19.6	210 – 226	Flight proven
MT-6	IHI AeroSpace	23 - 50	215 - 225	Flight proven

Table 9 IHI AeroSpace's hydrazine monopropellant engines

Table 9 shows, likewise above, hydrazine monopropellant engines by IHI AeroSpace [18]. Feed pressure range is about 5 – 28 bar and specific impulse is around 220s, with different levels of thrust. This is to bring another example on how many solutions the market can offer lately.

- **Alternative (Green) Propellants rockets**

Since hydrazine has been proven to be highly toxic, new propellants have been developed in the last years. The latter result to be less flammable, with convenient practical implications, like fewer safety requirements for handling and so on, which reduce operational oversight by safety and emergency personnel.
As an example, external hydrazine leakage is considered “catastrophic”, while the same situation with green propellants could be classified as “critical” or even “marginal” per MIL-STD-882E. This is really important because a classification of “critical” or less only requires two-seals to inhibit external leakage, which means that no additional latch valves or other isolation devices are required. The overall reduced toxicity of these green propellants is due to released gases when combusted, like water vapor, hydrogen and carbon dioxide. Moreover, fueling spacecraft with these propellants may be quicker due to a smaller exclusionary zone, making launch operations accelerated. Usually, they are less likely to exothermically decompose at room temperature due to higher ignition thresholds. So, they require fewer inhibit requirements, fewer valve seats for power, less stringent temperature requirements, and lower power requirements for system heaters. Alternative propellants also provide higher performance than the current state-of-the-art fuel and have higher density-specific impulse achieving improved mass fractions [15].

Product	**AND or HAN based Propellant**	**Manufacturer**	**Thrust (N)**	**Isp (s)**	**TRL Status**
GR-1	HAN	Aerojet Rocketdyne	0.26 – 1.42	231	6
GR-22	HAN	Aerojet Rocketdyne	5.7 – 26.9	248	5
1 N HPGP	ADN	Bradford Engineering	0.25 – 1.00	204 – 235	9
HYDROS-C	Other	Tethers Unlimited, Inc.	1.2	310	6
AMAC	Other	Busek	0.425	225	5
Lunar Flashlight MiPS	ADN	VACCO	0.4	190	6
Integrated Propulsion System	ADN	VACCO	4.0	220	6
ArgoMoon Hybrid MiPS	ADN	VACCO	0.1	190	6
BGT-X5	HAN	Busek	0.5	220	5
EPSS C1K	ADN	NanoAvionics	0.3	252	7
Green Hybrid	Other	Utah State	8	215	6

Table 10 Green propulsion systems [15]

Table 10 show, once again, how many solutions there are available on the market. Many of the above have low thrust, if utilised singularly, except for one, GR-22 by Aerojet Rocketdyne. It was used, together with the 1 N version, for NASA’s GPIM AF-M315E Propulsion System [19].

- **Hydrogen peroxide rockets**

Hydrogen peroxide is another green propellant which can be utilised as monopropellant as well. Compared to hydrazine and green propellants, it reaches lower decomposition temperatures and thus lower specific impulse, but higher density. It is non-toxic and therefore can reduce costs and difficulty in handling. Contrarily to hydrazine and green propellants rockets, hydrogen peroxide rockets are less common in the market and are not presented in NASA's SoA 2018, which was the only available at the time this thesis has been written. Despite that, some technical reviews and articles can be found on the internet, showing the design of various prototypes of hydrogen peroxide monopropellant rockets.
As an example, Alta S.p.A (Italy) and DELTACAT Ltd. (United Kingdom) have conducted a study, funded by ESA, on 5N and 25N engines, in which even different catalyst bed configurations have been tested [20].
Another example is a study on a 2N microthruster which utilises 90% concentrated hydrogen peroxide as monopropellant [21].
Again, from Universiti Teknologi Malaysia, two hydrogen peroxide monopropellant engines of 50 N [22] and of 100 N [23] have been developed and tested.

- **Cold gas rockets**

Cold gas systems are relatively simple and mature, by now. An inert gas, stored in high pressure gas or saturated liquid forms, is expelled from a nozzle, producing thrust. The values of specific impulse are definitely low (under 100s usually), but it can be improved using warm gases. These engines are inexpensive and robust, thanks to their very low grade of complexity.

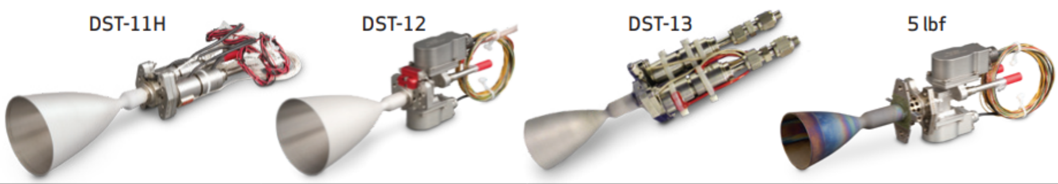
Product	**Manufacturer**	**Thrust**	**Specific Impulse(s)**	**Propellant**	**TRL Status**
MicroThruster	Marotta	0.05 – 2.36 N	65	Nitrogen	9
Butane Propulsion System	SSTL	0.5 N	80	Butane	9
Nanoprop 3U	GomSpace/NanoSpace	0.01 – 1 mN	60 – 110	Butane	9
Nanoprop 6U	GomSpace/NanoSpace	4 – 40 mN	60 – 110	Butane	9
MiPS Cold Gas	VACCO	53 mN	40	Butane	7
MarCO-A and B MiPS	VACCO	25 mN	40	R236FA	9
CPOD	VACCO	10 mN	40	R236FA	7
POPSAT-HIP1	Micro Space	0.083 – 1.1 mN	32 – 43	Argon	8
CNAPS	UTIAS/SFL	12.5 – 40 mN	40	Sulfur Hexafluoride	9
CPOD	VACCO	25 mN	40	R134A/R236FA	6

Table 11 Cold gas thrusters [15]

As it can be seen from Table 11 Cold gas thrusters Table 11, the technological readiness level is very high, but these thrusters can only provide low specific impulse. Levels of thrust presented are low as well, but one could think to enhance them by increasing the dimensions of the engine and the feed pressure, with increase in mass obviously.

- **Bipropellant rockets**

Bipropellant engines can provide higher specific impulses (over 300 s), depending on the propellants, and arbitrarily high levels of thrust, depending on the dimension of the engine. They are a mature technology by now, but not really common in small spacecrafts because they need at least two separate tanks (one for the fuel and one for the oxidiser), excluding the one for the pressurant, if needed, a combustion chamber subjected to elevated temperature and a system to control the mixture ratio.



Design	DST-11H	DST-12	DST-13	5 lbf
Propellant	Hydrazine/MON	MMH/MON	MMH/MON	MMH/MON
Nominal Steady State Thrust	5 lbf (22N)	5 lbf (22N)	5 lbf (22N)	5 lbf (22N)
Feed Pressure	80 – 400 psia (5.5 – 27.6 bar)	60 – 400 psia (4.1 - 27.6 bar)	80 – 400 psia (5.5 - 27.6 bar)	39 - 320 psia (2.8 - 22.1 bar)
Nozzle Expansion	300:1	300:1	300:1	150:1/300:1
Nominal Mixture Ratio	0.85	1.61	1.65	1.61/1.65
Valve	Solenoid	Latching Torque Motor	Solenoid	Latching Torque Motor or Solenoid
Valve Power	41 watts max (2 coils wired in series)	6 watts max (Latch) 7 watts max (primary) 9 watts max (secondary	41 watts max (2 coils wired in series)	6 watts max (Latch) 7 watts max (primary) 9 watts max (secondary) (Torque Motor) 15.6 watts max (solenoid)
Mass	1.7 lbm (0.77 kg)	1.4 lbm (0.64 kg)	1.5 lbm (0.68 kg)	1.4 – 2.0 lbm (0.64 – 0.91 kg)
Length	10.3 in (262 mm)	9.6 in (244 mm)	10.4 in (264 mm)	9.7-13.5 in (248 - 343 mm)
Chamber Material	Platinum/Rhodium Alloy	Platinum/Rhodium Alloy	Platinum/Rhodium Alloy	C-103
Performance	**DST-11H**	**DST-12**	**DST-13**	**5 lbf**
Specific Impulse	310 secs	302 secs	298 secs	288 secs/292 secs
Throughput	907 kg (2000 lbm)	1073 kg (2365 lbm)	637 kg (1404 lbm)	484 kg (1068 lbm)
Programs	Intelsat, BepiColombo, Wild Geese, Tenacious, GOES-R	AsiaSat 5, Telstar, Himawari, Turksat	NASA SDO	ETS-8, QZSS, Superbird-7, ST-2, WGS, Intelsat
Highlights	DST-11H provides highest performance available in a hydrazine/ MON ACS Thruster	DST-12/13 Provides highest performance available in MMH/MON ACS Thruster		Engine has been in production for more than 30 years, with > 2000 delivered and flown

Table 12 MOOG's attitude control bipropellant thrusters

Table 12 shows some ready solutions by MOOG. Hydrazine is used, so the problems mentioned in the previous sections are present again.

- **Solid-propellant rockets**

Solid-propellant rockets are usually used for impulsive manoeuvres, achieving moderate specific impulse and high trust magnitudes. Thanks to the solid propellant, they can be compact and suitable for small buses as well. There are some examples, like CAPS-3 by DSSP and even MAP by PacSci EMC, which is even customizable [15]. Usually, solid propellant rockets are more difficult to test, because the grain must be installed inside the engine and no prior to flight check is possible. They may present DDT issues (Detonation-Deflagration-

Transition). They are also difficult to restart and to throttle and can be even toxic and hazardous due to the propellants.

- **Hybrid rockets**

Hybrid rockets are usually low cost and safer than solid rockets, have medium performances (between solid and liquid rockets ones) and can be restarted, tested and throttled. Throttling is less efficient if compared to liquid rockets. On the other hand, they could be cumbersome because they need a properly long combustion chamber (depending on the level of thrust to be exerted) for the fuel grain and a tank for the oxidiser.

3.3 *“SphereX” robots example*

A high-level study on a mission about exploring lava tubes with spherical robots has already be carried out [24]. It presents a team of 30 cm diameter spherical robots, as they are not prone to single-point failure and they are able to operate in bucket brigades, which is an advantageous aspect because of the lack of line-of-sight communication due to the thick rocks. These microbots are capable of hopping, flying and rolling through caves, lava-tubes and skylights.

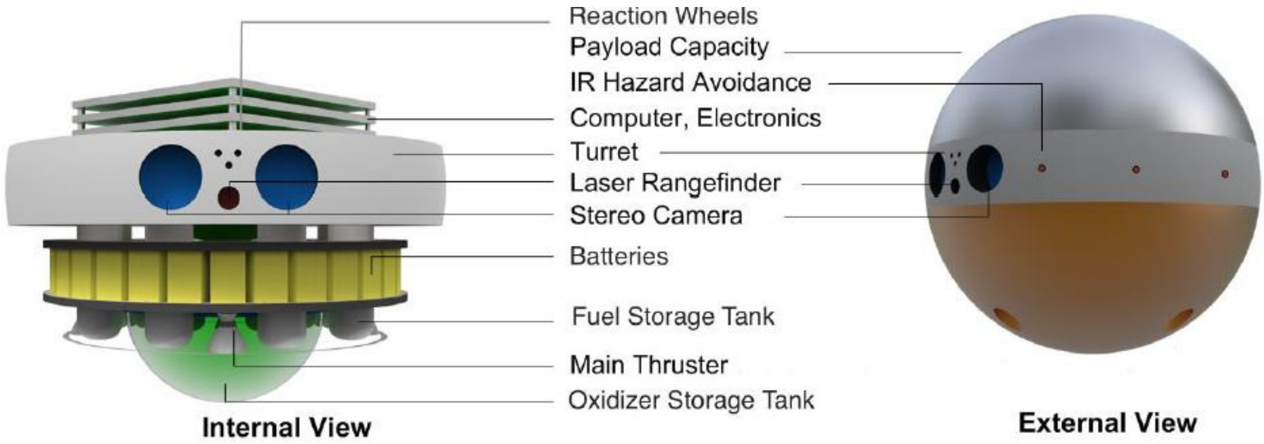


Figure 10 Internal and external views of SphereX

Figure 10 shows the complete design of SphereX. Focusing on the propulsion system, a bipropellant rocket engine is utilised as main engine, while eight little “warm gas” thrusters are dedicated to the attitude control system, as shown in Figure 12.

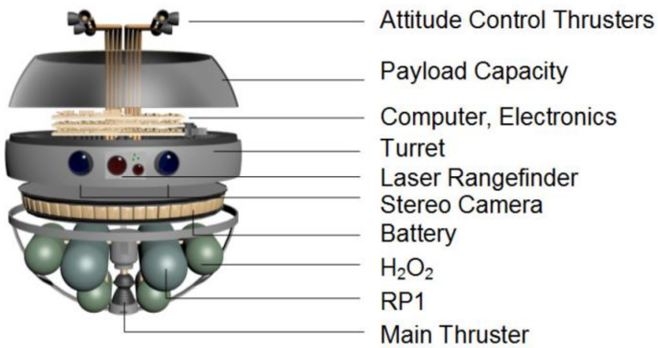


Figure 11 Other internal view of SphereX

Figure 11 shows an internal view of SphereX, with a more detailed focus on the propulsion system. Hydrogen peroxide plays the role of oxidiser for the main engine and of propellant for the ACS, as it can be seen in Figure 12. RP-1 is the fuel for the main engine. The main engine and the tanks are located at the bottom of the spacecraft, while the ACS thrusters find their position at the top.

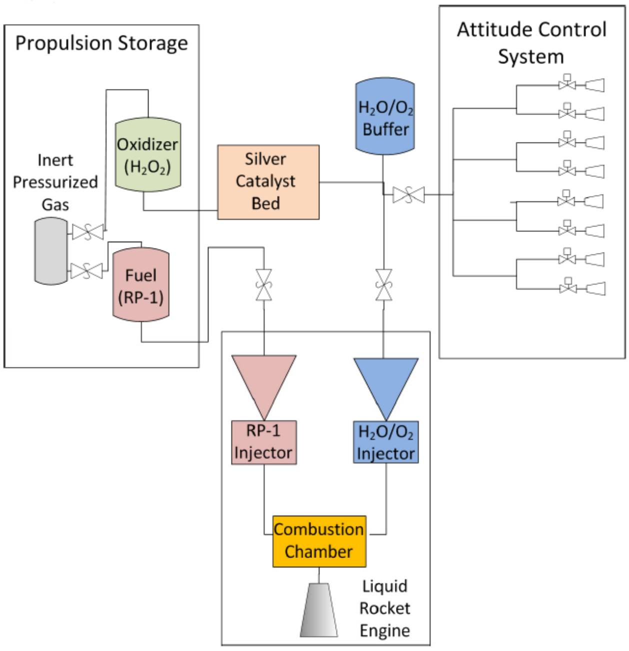


Figure 12 Propulsion system of SphereX

In this case, to avoid the use of pumps and mechanical devices, since the available space is reduced, the design considers pressurised nitrogen to be stored in a reservoir to initiate transport of the reactants into the combustion chamber. A silver catalyst bed makes the hydrogen peroxide decompose into gaseous oxygen and water with a temperature of about 600°C, which then feeds both the ACS thrusters and the combustion chamber. The specific impulse for the ACS thrusters is predicted to be approximately 180 seconds (with no combustion), while 330 seconds with 50% H2O2 concentration for the main engine [25].
The conclusion of this study states that the challenge will be in integration and miniaturization of the system into a 30-cm sphere, although the propulsion technology is mature.

4 Propulsion System – Design

4.1 Introduction

First of all, the propulsion system represents one of the most important subsystems in LuNaDrone, because it defines how the spacecraft is able to move in space and for how long, both in term of distance and time. Since the environment does not include the presence of an atmosphere, no air-breathing engines can be taken into consideration. A rover is supposed to carry the spacecraft in immediate proximity to a lunar skylight, so the point where the rover would stop would be considered as the starting point for the mission. The spacecraft would have to be able to autonomously move from that point to others, towards the inside of the skylight. This action could be performed with different strategies, but the goal of this thesis is to develop a 12U spacecraft, so a system of wheels would not be suitable in terms of encumbrances. This is also due to the impossibility to know the physical configuration of the lunar soil perfectly, which would mean unpredictable obstacles for a too small system of wheels. Moreover, the objective of the mission is to explore the inside of the hole beneath the skylight, whose walls have been demonstrated to be extremely steep. For these reasons, rocket propulsion is the chosen strategy for LuNaDrone.

So, among the various types of rocket propulsion, which one to choose?

Using a cold gas thruster as main engine would be not convenient because of the very low specific impulse, despite its simplicity.
A bipropellant rocket would not be suitable despite its high specific impulse and ease to throttle, because of its complexity in managing two different propellants and the feed system.
A solid rocket would not be suitable, because it is not easily restartable, it usually utilises toxic propellants and thus considered dangerous.
A hybrid rocket would be not easy to develop, because it includes many technological difficulties, like the construction of a suitable grain and the management of high temperatures in the combustion chamber. It is not impossible, but, in this case, it would require too much effort in its development.

As seen in section 3.2, in general hydrazine and green propellants have greater specific impulses with respect to hydrogen peroxide. Despite that, hydrazine is considered to be highly toxic, while green propellants not, but both of them reach elevated temperatures in the combustion chamber. Hydrogen peroxide is not toxic and reaches lower temperatures, consequently leading to a loss in specific impulse but also to a simpler heat management.
Although the propulsive performances of hydrogen peroxide are about 20% lower than hydrazine, the volume specific impulse achievable with 90% H_20_2 is higher than most of other propellants [20], due to its high density (about $1400\ kg/m^3$).

Another really important feature to be considered is the availability of components and facilities where to test the engine. This aspect fits in the scenario described in the abstract, where it has been said that involving Italian SME and universities would be useful and advisable. In fact, the objective of this thesis is to produce a study in which the LuNaDrone mission could be feasible (in technological terms) in a few years, so the main drivers of the choice are the ones described in this paragraph.

To conclude, for what been said before, the choice fell on a hydrogen peroxide monopropellant rocket, manufactured by T4i, which is a spin-off from the University of Padua. They have developed various engines in the past years and have a solid technological portfolio. All their engines are currently being tested, with a TRL of 5, intended to raise in the next years. Moreover, T4i is fully available to develop customised engines, so, depending on the requirements of the mission, the engine would be tailored as a result of trade-offs.

4.2 Hydrogen peroxide monopropellant propulsion system

Figure 13 An example of hydrogen peroxide rocket, with gentle courtesy by T4i

KEY FEATURES	BENEFITS
Low cost	The system is specifically conceived to represent a trade-off between costs and performances, targeting a substantial cost reduction respect to currently available hydrazine mono-propellant units.
Customizable	The system is customizable on thrust, expansion ratio and accommodation to fit specific customer needs.
Green	Hydrogen peroxide is a green non-toxic monopropellant. Furthermore, our system uses highly stabilised peroxide which, respect to MIL-Grade peroxide allows for reduced compatibility issues.
Restartable	The system is restartable many times.
Throttleable	The system allows for a 1:5 throttleability.
Performance	Current delivered specific impulse of 155 s.

Table 13 Key feature of monopropellant hydrogen peroxide rocket, by T4i [26]

Table 13 Key feature of monopropellant hydrogen peroxide rocket, by T4i Table 13 is directly taken from T4i website and presents, substantially, the reasons why the choice fell on their engine. Moreover, it has been possible to visit them personally in Padua and seed an example of test.

Another fundamental characteristic of the hydrogen peroxide, in addition to the non-toxicity, is that, after an initial transient, the plume produced is totally transparent to the visible light. This is important because it will not represent an obstacle for the cameras to acquire images.

On the other hand, hydrogen peroxide is fully compatible only with a few materials, sometimes neither able to withstand long-time storage. This is to say that it will be necessary to pay attention to the design of the tanks in which hydrogen peroxide will have to stay from the pre-launch phase to the end of the mission.

4.3 Propellant handling and storage

As just above-mentioned, hydrogen peroxide shall be stored in the tank from before the launch to the end of the mission, so the magnitude of this process is surely days and not only a few hours. Hydrogen peroxide is not easy to store because of its great tendency to react with many materials. In fact, there are four classes of material compatibility. Only class-1 materials (e.g. pure aluminium and PTFE), after additional surface treatment, may be used for long-term storage. It is not recommended to use class-1 liners covering class-2 (or higher) structures, because, in case of a failure in the liner, the whole system fails. It is also obvious that, since high pressures are needed in the propellant tank, the chosen material shall be able to bear those pressures.

The following information is not compulsory for the aim of this study, but it is really important during the phases of test of the spacecraft and in the phases of its operational life.

		Plastics								Elastomers				Metals							
H_2O_2	Material	Polypropylene (GF)	PVDF (carbon filled)	Nylon 11	PVC	PPS (GF)	PTFE	ABS (GF)	Acetal (Delrin)	Viton	EPDM	Nitrile	Silicone	St Steel 303/304	St Steel 316	Aluminium	Brass	Hastelloy C	Titanium	Alumina Ceramic	Ceramic Magnet
30%		L	A	X	A	A	A		X	A	L	X	L	L	L	L		A	L	A	A
87%		L	A	X	A	L	A	A	X	A	X	X	L	L	A	A	X	A	L	A	A

Table 14 Compatibility with some materials[2]

Table 14 [27] shows the compatibility of two differently concentrated solutions of hydrogen peroxide with various materials. Viton and Stainless Steel 316 will be taken into consideration.

Iron, steel, copper, brass, nickel and chromium, many common materials of construction, are not suitable for handling solutions of hydrogen peroxide, and recommended materials must be used.

Personnel working with high concentration hydrogen peroxide have also to have a personal protective equipment made of chemically resistant material, including gloves, boots, rain suits, eye protection, face shields.
Groups of two or more should always do HTP operations. HTP is never handled alone.
Cleanliness of operations is one of the best means to mitigate and reduce the chance of contamination. HTP systems normally remain quite safe as long as proper cleanliness is maintained. Easy procedural steps which will make overall safety very high are keeping systems clean and free from catalytic agents or other contaminants that can compromise the system compatibility.

[2] A = suitable, L = limited, X = unsuitable, Blank = insufficient data

If a filter is used, it should be considered as a high-risk component not a passive part of transfer system. Additional precautions should be used to monitor the behaviour of the filter such as periodic inspections, temperature measurements, flow rate or pressure drop measurements, fluid temperature changes in the filter and others.
A good practice is not to pressurise the run tank by opening a vent valve, because this will ingest the local air, bringing contamination into the system. Instead the system should be pressurised with a slight positive pressure with a source of clean filtered gas and then vented [28].

Strength of hydrogen peroxide solutions expressed in	% W/W	H_2O_2 content	g/100 g	3.0	10.0	20.0	27.5	30.0	35.0	50.0	60.0	70.0
	% W/V	H_2O_2 content at 20°C	g/100 ml	3.0	10.4	21.4	30.3	33.3	39.6	59.8	74.5	90.2
	Volume	Volume of gaseous oxygen (litre) given off per litre of solution at 20°C (0°C and 760 mm Hg or 101,325 kPa)	l/l	10	34	71	100	110	130	197	246	298
		Active oxygen content	g/kg	14.1	47.0	94.1	129.3	141.1	164.6	235.2	282.2	329.2
		Freezing point	°C	-1.6	-6.4	-14.6	-22.6	-25.7	-33.0	-52.2	-55.5	-40.3
		Boiling point at 101.325 kPa (760 mm Hg at 0°C)	°C	100.4	101.5	103.6	105.5	106.3	107.4	113.9	119.0	125.5
		Density at 0°C	kg/dm^3	1.012	1.039	1.080	1.112	1.123	1.144	1.211	1.258	1.307
		25°C	kg/dm^3	1.007	1.032	1.069	1.098	1.108	1.128	1.191	1.236	1.284
		50°C	kg/dm^3	0.997	1.020	1.055	1.082	1.091	1.110	1.171	1.214	1.260

Table 15 Example of physical properties of hydrogen peroxide solutions

Despite there is reference to a 90% wt hydrogen peroxide solution, which will be used in this case, Table 15 [29] shows some important physical characteristics. Among them, the freezing point is really important, because solidification of propellant during the operating life of the spacecraft must be avoided, both during commissioning and during the flight of the spacecraft, when the discharge of the tank could leak to a sudden decreas in the temperature.

Another aspect to be taken into account is the effect of heat. In fact, apart from self-heating as a result of decomposition, consideration must be given to the effect of temperature rises caused by outside sources of heat. For purely physico-chemical reasons, the rate of the decomposition reaction in solution (homogeneous) will increase 2 to 3 times for every 10°C increase in temperature, and the rate of the surface decomposition (heterogeneous) will increase 1 to 2 times per 10°C.

Ruthenium oxide	RuO_4	Platinum	Pt
Manganese oxides	Mn_2O_3, MnO_2	Osmium	Os
Iron oxides	FeO, Fe_2O_3	Iridium	Ir
Cobalt oxide	CoO	Palladium	Pd
Nickel oxides	NiO, Ni_2O_3	Rhodium	Rh
Lead oxide and hydroxide	PbO, $Pb(OH)_2$	Silver	Ag
Mercuric oxide	HgO	Gold	Au

Table 16 Most active catalysts for hydrogen peroxide

The heterogenous decomposition is what is needed for the functioning of the engine and fast decomposition may also occur if the hydrogen peroxide is brought into contact with insoluble solids. This is known as heterogeneous decomposition. Hydrogen peroxide would decompose to some extent on any surface even at ambient temperature, although the rate varies enormously with the nature and state of the surface. Thus, the rate of decomposition on silver is 107 times faster than that, for example, on polyethylene, which is one of the common handling materials. Some of the solid compounds which catalyse the decompostion of hydrogen peroxide are the hydroxides and oxides of the heavy metals, as well as the noble metals themselves. Table 16 shows a list of the most active catalysts.

Even considering low dilutions, hydrogen peroxide will continuously decompose into water and oxygen. This rate can be maintained very low storing hydrogen peroxide in approved materials and keeping it free from contaminants. However, if oxygen pressure is not relieved, then high gas pressure may build up.
If the heat of decomposition is not removed with the rate at which it is developing (by heat loss to the surroundings or cooling), the temperature will rise and the rate of decomposition will increase. This may lead to a self-accelerating decomposition which, for badly contaminated solutions, may end with an extremely rapid decomposition or “boil off”.

Another hazard is the explosion. The most important factors which could cause an explosion are:

a) the concentration of hydrogen peroxide, water and organic material;
b) the nature of the organic material;
c) the presence of an initiation source;
d) the temperature of the mixture.

Another issue to be considered is the interface between hydrogen peroxide and valves. Hydrogen peroxide solutions, if confined between closed valves, can lead to a pressure burst even when uncontaminated. The problem can be overcome by several methods, like vents, pressure relief valves, elimination of valves where possible, and locking open certain valves during normal running, where appropriate. Even non-contaminated hydrogen peroxide solutions, in ball valves or diaphragm valves, may lead to pressure bursts if the ball or the bonnet are not vented.
If hydrogen peroxide is pumped against a dead end (e.g. closed valve) the heat generated can lead to rapid decomposition with gas evolution, and a pressure burst can subsequently follow. Steps must be taken during the design (e.g. no-flow trip, kickback line or pressure relief) to avoid this happening [29].

Many other information may be found on the internet, about storage and handling, tanks, valves and fittings [30] [31].

4.4 Engine

Unlike bipropellant engines, which need a combustion chamber, monopropellant engines only need a "decomposition chamber", where the propellant decompose into different hot gaseous products.
In this case, hydrogen peroxide decomposes into water vapour and oxygen, following a chemical reaction in which both oxidation and reduction occur simultaneously

$$2H_2O_2 \rightarrow 2H_2O\ (g) + \ O_2(g) + heat$$

This is very energetic, producing up to 98.1 kJ for every mole of peroxide that is decomposed (2.89 MJ/kg). What is important for the performances of the engine is the decomposition temperature, which vary with the mass fraction of the hydrogen peroxide, because it defines the characteristic velocity

$$c^* = c^*(\Gamma, R, T_c) = c^*(\gamma, \mathcal{M}, T_c)$$

which can be computed using various software available on the internet.

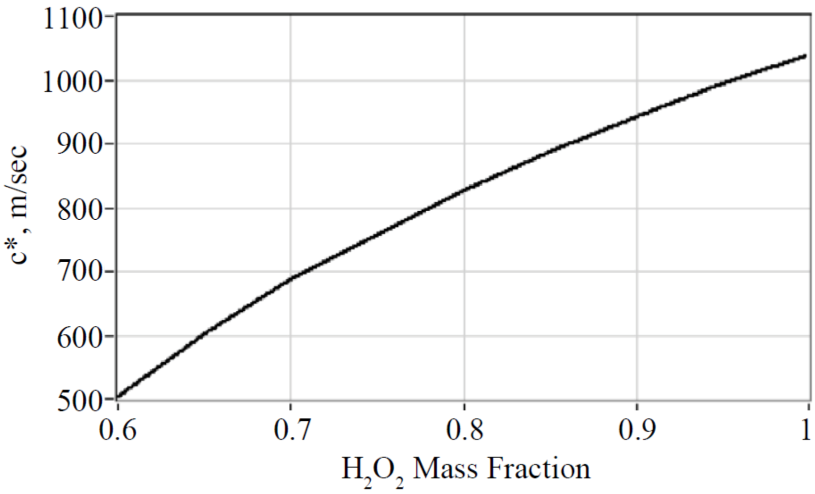


Figure 14 Characteristic velocity as a function of mass concentration [19]

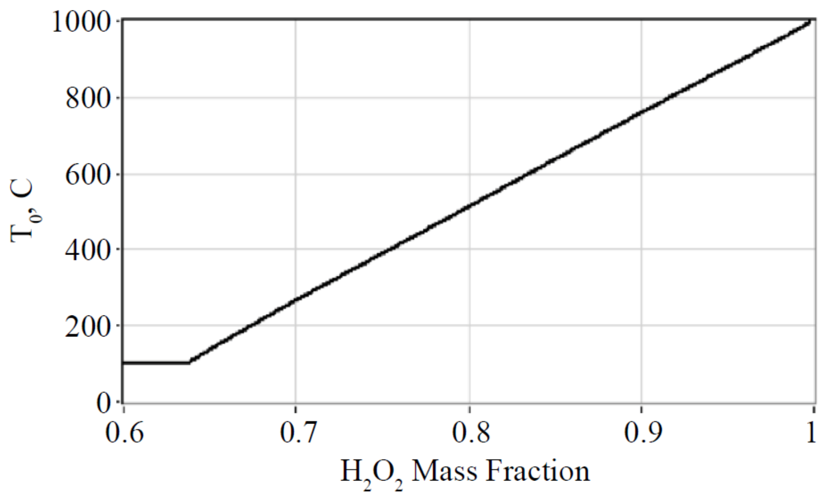


Figure 15 Decomposition temperature as a function of mass concentration [19]

Figure 14 and Figure 15 [32] show how hydrogen peroxide mass fraction affects the decomposition temperature and thus the characteristic velocity. Usually, 3-10 % hydrogen peroxide is utilised for medical applications, 30-50 % for industrial and agricultural applications and much higher compositions for propulsion applications. Hydrogen peroxide solutions with mass concentration greater than 95% are often called HTP (high test peroxide), but in this study the mass fraction is supposed to be around 90-92%, which leads to a decomposition temperature of about 750 °C and a characteristic velocity of about 950 m/s.

The characteristic velocity, together with the pressure in the "combustion chamber", are the inputs for the following computations. From a private call, it is known that the engine can work properly in certain range of combustion chamber pressures, approximately from 7 bar to 15 bar: the lower the pressure is, the greater the engine turns out, at a given thrust. Low pressures are not suitable for the correct functioning of the catalyst, while high pressures lead to higher pressure in the propellant tank, which means an increase in mass.

Obviously, the amount of thrust is required as well and it is an output from mission analysis, even though it is difficult to be defined and it is a result of a trade-off, in which one tries to optimise fuel consumption and encumbrances of the propulsion system.

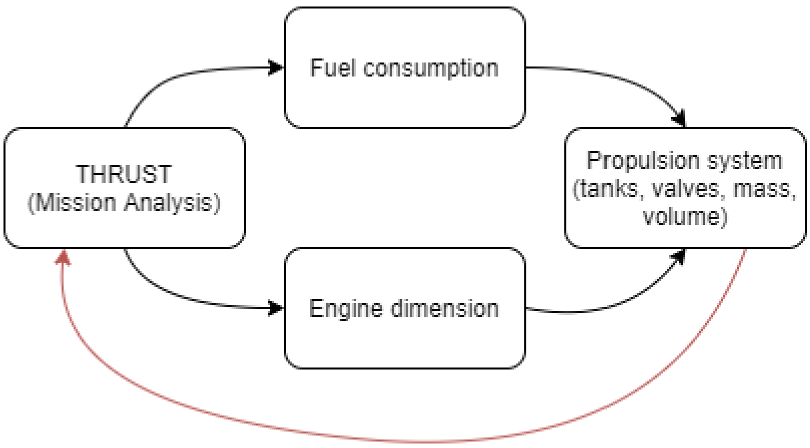


Figure 16 Conceptual propulsion system design

To sum up, three different inputs are needed:

INPUT	Description
Thrust	To be defined with Mission Analysis
c^*	Defined by mass fraction of hydrogen peroxide
Chamber pressure	To be defined

Table 17 Inputs for the fundamental relations

4.4.1 Fundamental relations

The next step is to apply the relations of an ideal rocket [33]. The definition of characteristic velocity is

$$c^* = \frac{p_c A_t}{\dot{m}}$$

from which one can derive the expression for the throat area

$$A_t = \frac{c^* \dot{m}}{p_c} \quad (*)$$

Now, the mass flow is not known yet, but there is another definition which states

$$c = \frac{F}{\dot{m}} \quad (**)$$

where F is the thrust and c is the effective exhaust velocity. Again, c is not known yet and the definition of specific impulse can be manipulated to obtain

$$I_{sp} = \frac{c}{g_0} \quad (***)$$

The specific impulse is a measure of how efficiently a rocket utilises the propellant and it can be found on literature or evaluated, see Annex I - Chemistry for further details, in which a value of 168 s is calculated by using CproPep. In this case, a value of 155 s is a good approximation, because it is currently the value written on T4i's website [26]. Since it is proportional to the effective exhaust velocity (***), it is easy to obtain the latter, with which one can evaluate the mass flow (**) and finally the throat area (*). Then, hypothesising an expansion ratio, all the other quantities can be evaluated.

The entire engine may be built by using additive manufacturing. Inconel 718 is widely common among monopropellant rockets for this scope, while the catalyst bed has obviously to be tailored and made of another material suitable for the hydrogen peroxide decomposition.

4.4.2 Nozzle

Now, it is important to say that, since the spacecraft is supposed to operate in vacuum, high values of expansion ratios are needed in order to let the nozzle be adapted. This means that the expansion of the gases occurring from the throat to the exit of the nozzle is such that the pressure of the gases at the exit section equals the external pressure, which is practically zero. This would be the ideal condition, with neither over-expansion nor under-expansion, but it would require an infinite expansion ratio ε, theoretically. It is clear that this condition is not applicable because of dimension constraints of the mission. A trade-off has to be found, trying to minimise losses in the nozzle with acceptable encumbrances.

The aim of the nozzle is to direct exhaust gases from the throat to the environment, trying to exploit them the most possible, in terms of thrust, and thus to make losses small. A good nozzle configuration let one obtain the highest practical specific impulse, minimise inert nozzle mass and conserve length [33], which is compulsory in this case to manage the 12U available volume.
The ideal thrust would be given by

$$F = \dot{m} v_e + A_e (p_e - p_0)$$

but usually the gases do not exit with axial velocity.

The reference nozzle is a 15° half angle cone, which gives a correction factor of

$$\lambda = \frac{1}{2}(1 + cos\alpha) = 0{,}9830$$

This represents the ratio between the momentum of the gases in a nozzle with a finite nozzle angle 2α and the momentum of an ideal nozzle with all gases flowing in axial direction. In this case, with a divergence cone angle of $2\alpha = 30°$ the exit momentum, and therefore the exhaust velocity, will be 98,3% of the ideal velocity as a function of the inlet absolute temperature, the pressure ratio, the ratio of specific heats and the gas constant.
One can immediately say that the smaller the half cone angle is, the better the performances are, which is true, but not suitable in this case because of encumbrances. Conical nozzles are easy to construct, but they do not represent the best solution for a lightweight design. The so called bell-shaped or contour nozzle is the most common solution today, because, even though its design is more complicated than the cone nozzle, it is convenient in terms of mass. Right downstream the throat, a high expansion without separation can occur, thanks to high relative pressures, large pressure gradient and rapid expansion as well. The wall contour is shaped to minimise losses and thus the expansion in the supersonic bell nozzle is more efficient than in the simple straight nozzle.
For sake of simplicity, further details are omitted, but the final step is to find the values of initial and final parabola angle which give the required nozzle length, a compromise between losses and encumbrances.

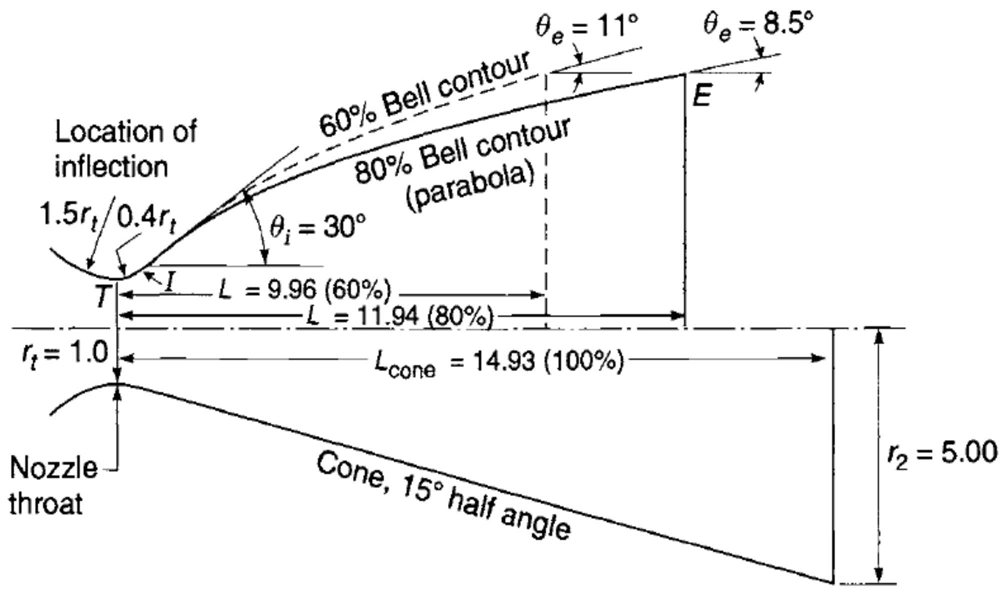


Figure 17 Representation of cone nozzle and bell contour nozzle with the same ε [16]

Given the throat area and the expansion ratio, which should be the highest possible, the length of the reference cone (15° half angle) can be computer ad follows

$$L_{cone} = \frac{r_e - r_t}{\tan\alpha}$$

where r_e and r_t are respectively r_2 and r_1 of Figure 17. Now, the length of a bell nozzle is usually given a fraction of the length of the reference conical nozzle, just computed, obviously at a given expansion ratio.

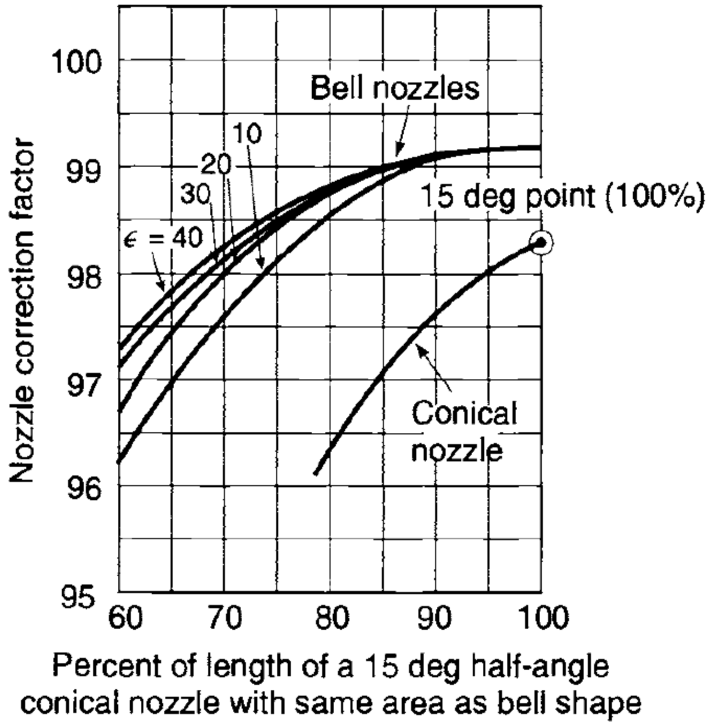


Figure 18 Correction factor and percent of length [16]

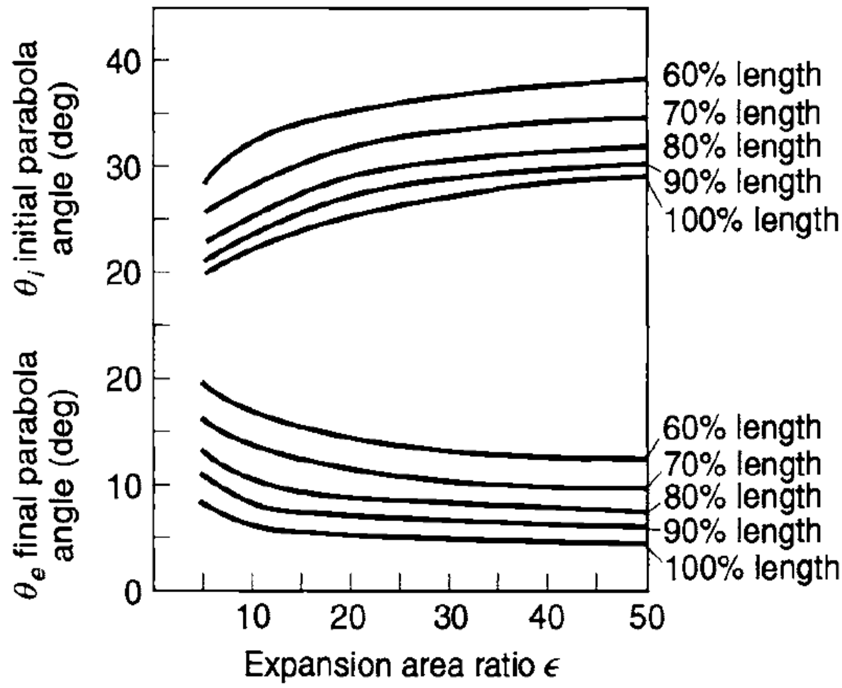


Figure 19 Final and initial parabola angle

Figure 18 shows, as a function of the expansion ratio, that the correction factor decreases if the percent of length of a 15° half angle cone decreases, or, in other terms, if one chooses to reduce the bell shaped nozzle length the losses will increase. A trade-off between acceptable losses and acceptable nozzle length has to be carried out. Figure 19 shows, given the expansion ratio and the percent of length, which angles will the parabola have.

Parameter	**Values**
Correction factor λ	$\approx$ 98%
Expansion ratio ε	70
Percent of 15° cone nozzle	70%
Initial parabola angle θ_i	34°
Final parabola angle θ_f	10,5°

Table 18 Parameters parabola for the bell nozzle

4.4.3 Decomposition chamber

The decomposition chamber in a monopropellant engine is what acts for the combustion chamber in a bipropellant engine. It hosts what is called catalytic bed, whose task is to start and enhance the decomposition reaction. The more efficient the catalyst bed, the better performances. Substantially, a tortuous the flow path is inside the catalyst increases the bed surface area and the residence time of the propellant, favouring the hydrogen peroxide decomposition. Unfortunately, this makes the pressure drop increase and so a trade-off is necessary [34]. There are many techniques to be used to build a catalytic bed. Its performances strongly depend on the type of material and its geometry. Some configurations show a base of manganese and cobalt oxides [21], platinum [35], silver screens [36] and others. Many are the testing plans and the technique of construction, including additive manufacturing. Anyway, the optimization of the catalytic bed in terms of geometry and material is left to the developer of the engine.

What is important in this study is that the temperature of the catalytic bed affects performances and durability.
If the engine is at the right temperature, it needs tenths of a second to work properly. Instead, at the first ignition, the engine requires between about 1-2 seconds to reach its nominal performance, and this is due to the bad response of the catalyst, which is still cold. This could represent an issue for the flight, at least at the first ignition, unless a second ignition after the engine has had time to cool down too much would be necessary. The latter is not the case, because the flight plan is not supposed to leave the engine switched-off for a long time after the first burn. During nominal functioning, the engine would heat by itself, thanks to the decomposition reaction. However, as already said, it would be better to place some heaters in the proximity of the bed, or to heat it up in another way. Not only this would let the engine reach the nominal functioning faster, but also enhance performances and, more importantly, preserve the life of the catalyst [37]. It has been widely demonstrated that strong thermal shocks can strongly and irreversibly ruin the material of the bed. This would not be acceptable in terms of safety and reliability, because if the main engine is lost the mission surely fails.

Another aspect to be carefully designed is the so-called bed loading G, which is substantially how much propellant mass passes through the catalyst per unit of area and time. There are upper and lower limitations for G. If G is too high, it means that the propellant is flowing too quickly or in an excess of mass, and thus the catalyst bed has not enough time to work properly. On the other hand, if G is too low, the propellant is too slow and the catalyst is subject to sedimentations, which lead to a non-uniform utilise, with a decrease in performances. At a given mass flow, usually the longer the catalyst bed is, the better the decomposition occurs, but the worse the pressure drop gets. So, these situations drive the trade-off of the catalyst.

From a private call, it is usually a good compromise to have values of G around $100\frac{kg}{m^2 s}$. Since the mass flow is defined, supposing a cylindrical decomposition chamber, its cross section can be evaluated.

The length of the decomposition chamber plus the convergent section would deserve careful trade-off, impossible to do at this level, because there is no profound knowledge about the catalyst bed. Thus, the length of this section of the engine will be considered to be about the same of the stand off and in turn, the sum of the two is around the length of the nozzle.

For what concerns the pressure drop in the catalyst bed, it is really difficult to effectively estimate, because It strongly depend on the bed load, the type and the condition of the catalyst, the pressure of the propellant and so on.
On the internet it is possible to find various articles in which different formulas are proposed [38] [21]. On a file by NASA one can find some tests with 90% hydrogen peroxide, all showing full decomposition, with clear plumes and decomposition efficiencies above 95%, in which also a 5% loss in c^* is attributed to cold chamber heat sink effects during a short duration firing. Pressure drop is driven by internal catalyst bed geometry, speed of reaction, and downstream chamber pressure. High downstream chamber pressure with a low throughput will result in a low-pressure drop, just as opposite conditions will result in a high-pressure drop. A bed with a very low pressure drop often floods or has chug-like pressure oscillations [36].

Another empiric relation found on a study on the internet [39] is as following

$$\Delta P = 35000 \cdot \frac{G^{1.735}}{p_c^{0.778}}$$

Where ΔP is the pressure drop in psi, G is the mass flow rate per unit of area in lb/in^2/s and p_c is the decomposition chamber pressure in psi.

MASS FLOW	0,049344378	*kg/s*		
	49,3443777	*g/s*		
GOX	100			
chamber diameter	0,025065357	*m^2*	2,506535717	*cm*
chamber section	0,000493444	*m^2*	4,93443777	*cm^2*
rho	87,71108406	*lb/ft^3*	1405	*kg/m^3*
V	840,6405694	*ft/hr*	0,071174377	*m/s*
G	73733,49565	*lb/(hr*ft^2)*	0,142241008	*lb/(s*in^2)*
Pc	1200000	*Pa*	174,0456	*psi*
Δp	43,27542308	*psi*	2,98373656	*bar*

Table 19 Calculation example, with 50N of thrust and 155 of specific impulse

Table 19 shows an example with 50N of thrust and 155s of specific impulse, leading to an almost 3 bar pressure drop, with a mass flow rate per unit of area of 100 kg/m^2/s, as stated before. This is only to see the magnitude of a pressure drop to be expected, but since no precise information about the real G and the composition of the catalyst are provided, no precise evaluations can be done. However, the pressure drop is expected to be around a few bars in the catalyst bed.

Always from the same document, one can summarise the main characteristic quantities of catalysts in heterogenous catalysis.

Parameter	Typical value	Comment
Operating life	3000 s	To be maximised
Number of cycles	10000	Important for repeatability for ACS
T_c functioning temperature	$\sim 1000\ K$ (@90%)	Affected by concentration, pressure and decomposition efficiency
p_c nominal functioning pressure	10-20 bar	Affects the decomposition temperature and pressure drops
G mass flow rate per unit of area	50-400 kg/m^2/s	Affects pressure drops
η_{c^*} characteristic velocity efficiency	95-99%	Diminishes during operating life. Usually, if η_{c^*}=0.95 il EOL
Δp pressure drop	4-20 bar	Depends on compression loading, length of bed, G and p_c. When Δp increases by 10-40% with respect to BOL there is the end of operating life
Pressure oscillations	$\leq 5\%$ from peak to peak	When they increase in intensity with respect to the typical value there is the end of operating life

Table 20 Typical parameters of catalysts

4.4.4 Thermal stand-off

The thermal stand-off is required for one main function, which is a thermal function. It is usually built in such way that heat from the high temperatures in the decomposition chamber has difficulties in going upwards towards the pipes, valves and electronics. At this purpose, it generally presents holes (see Figure 13 or Table 8 for examples) or it is made of a poor heat conductor, to lower the conductive resistance not to let heat go back. Substantially, it has to be resistant enough to bear high thrusts and weak enough not to let the heat go back.

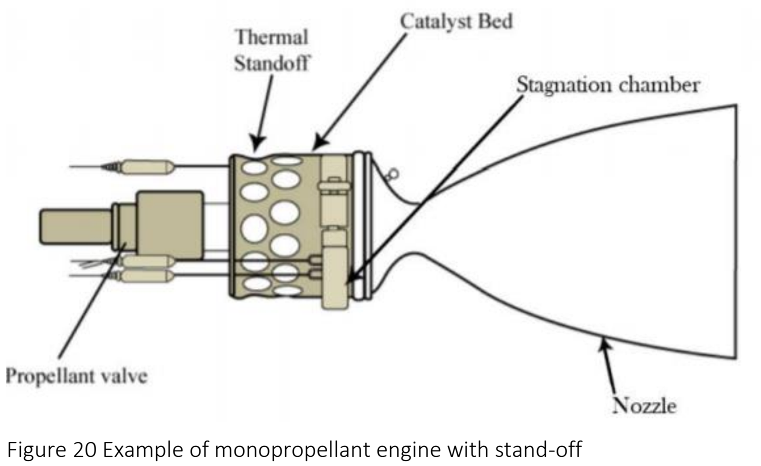


Figure 20 Example of monopropellant engine with stand-off

A good first approximation for the encumbrances of the stand-off is to consider the same length and the same diameter of the decomposition chamber.
There are papers on the internet showing more details on the design with additive manufacturing of the thermal stand-off and the injector [40].

4.4.5 Results

Engine	Thrust [N]	Metal_mass [g]	Catalyst mass[g]	Total_mass [g]	Total Length [cm]	chamber_diam [cm	diam_exit [cm]
1	1	0,75	0,375	1,125	2,3	0,29	0,68
2	5	3,02	1,51	4,53	3,8	0,65	1,51
3	50	88,8	44,4	133,2	12,1	2	4,8
4	75	129,1	64,55	193,65	14,4	2,5	5,85
5	100	170,3	85,15	255,45	16,6	2,9	6,76

Table 21 Encumbrances of engines varying the required thrust

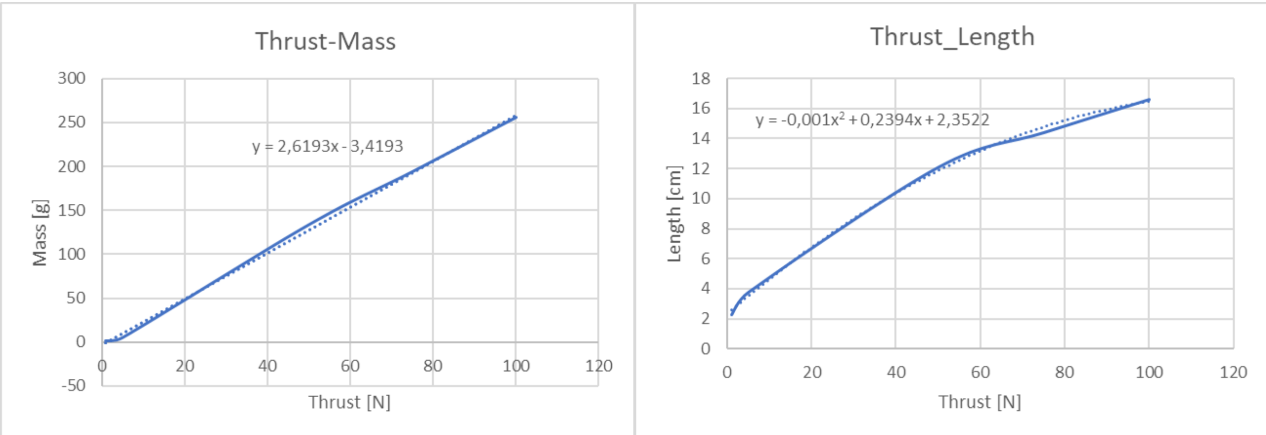


Figure 21 Mass and length of the engine as a function of the thrust

Table 21 and Figure 21 are derived from the drawing of the engines on Solidworks. The lengths are calculated as described in the previous sections, while the masses (Metal mass in the table) are calculated directly by the software, selecting Inconel 718 as material, which is commonly used for these applications. Then, catalysts masses are hypothesised to be 50% of the metal mass and the total mass it their sum.

Inconel 718	
Density	8200 kg/m^3
Yield strength	827 MPa

Table 22 Inconel properties

Note that the 100N engine, as example, has been drawn with a 1mm wall thickness, which is much higher than the required thickness with a safety factor of 3. Each engine has the same geometrical parameters written in Table 18.

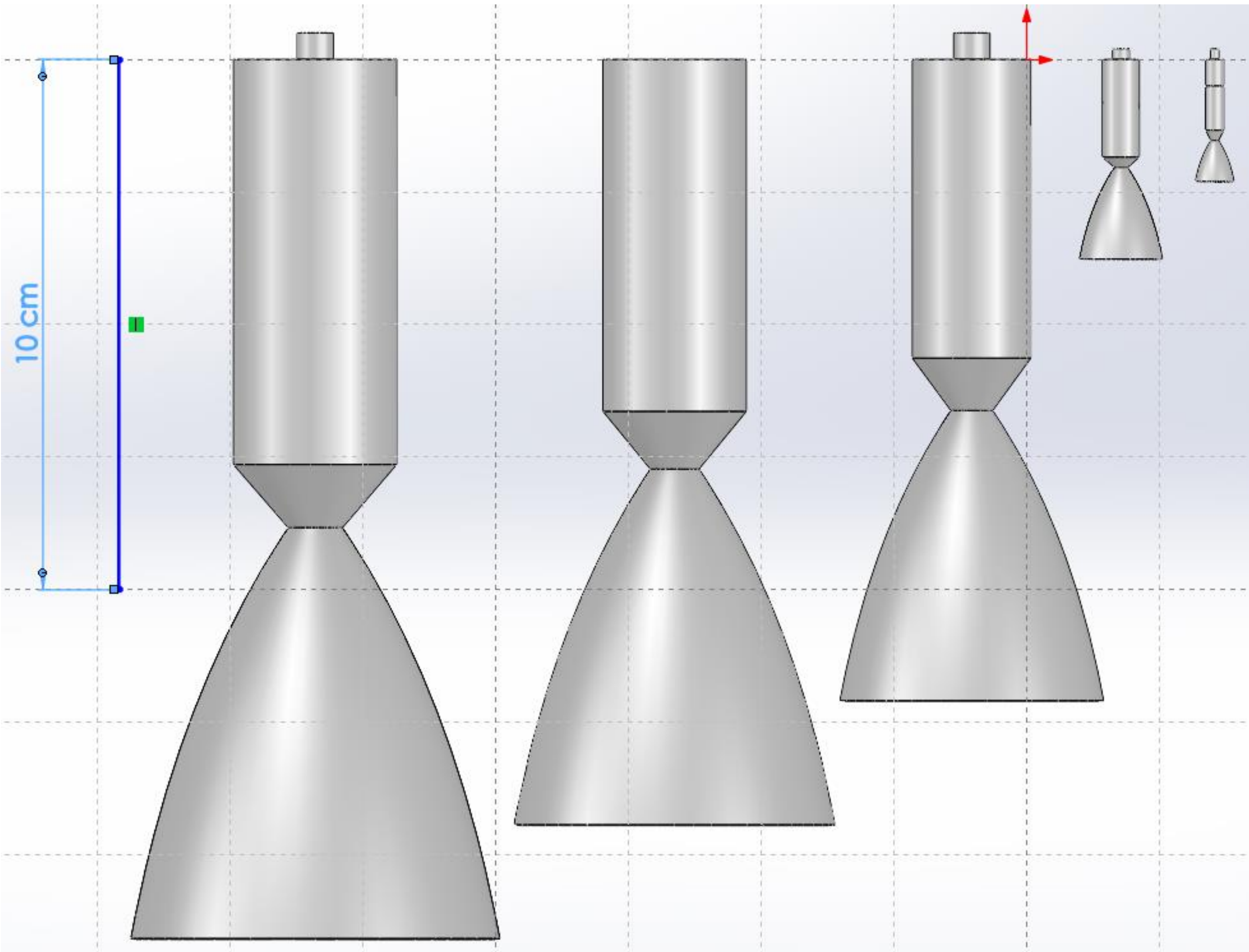


Figure 22 CAD representation of the engines 5-4-3-2-1, from the left to the right

The thermal stand-offs in Figure 22 has not been drawn with holes (they would look like the ones in Figure 20) to make the mass result higher and thus be conservative.
Later, in the final results section, it will be presented which of these engines will be put into the final configuration. The main driver of this choice will be the availability of room in the spacecraft and how to manage it.

4.5 Feed system

The engine needs the propellant to work and the propellant needs a way to be conveyed to the decomposition chamber. The feed system includes at least one tank, in which the propellant is stored, possibly another tank for a pressurant gas, different valve, filters, provisions for filling and removing (draining and flushing). These elements are connected by tubes and the difference of pressure between upstream (propellant tank) and the downstream (engine inlet) let the propellant flow in the right way. The aim of the current paragraph is the preliminary sizing of a complete feed system, trying to define the best solution in terms of type, mass and volume. For this application, turbopumps-feed systems will not be even taken into account because of their unnecessary complexity and their usual large encumbrance.

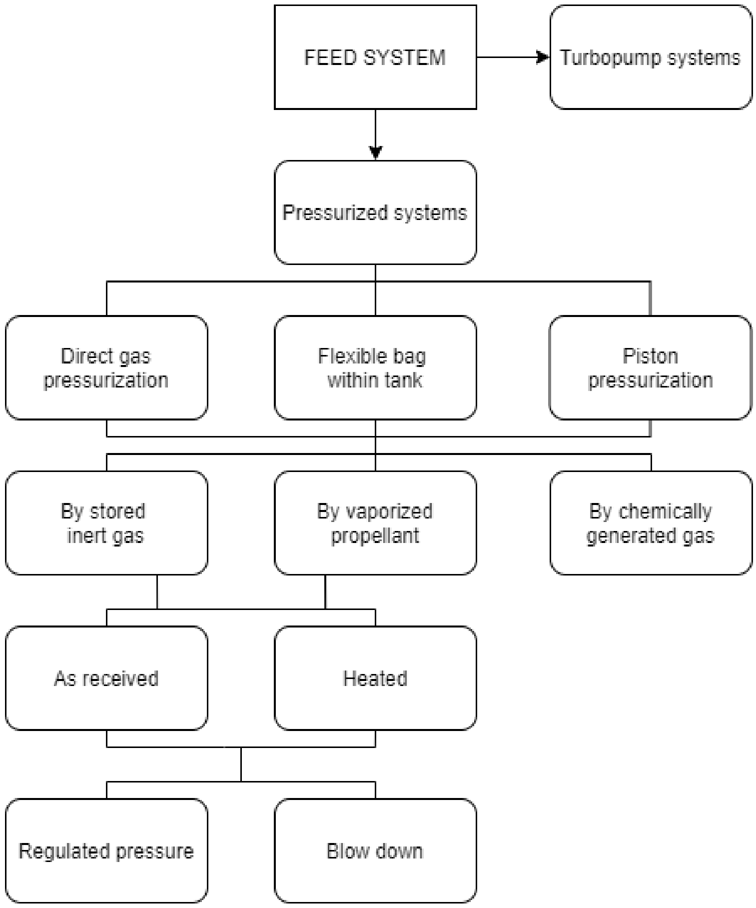


Figure 23 Different types of feed systems

Figure 23 [33] shows various types of feed systems, some of which will be treated in the following sections. Piston pressurization and flexible bags do not seem to be suitable for this mission because of their excessive encumbrance. Moreover, neither the solution with vaporised propellant looks like a good idea, because, as the name says itself, it would need a method to vaporise hydrogen peroxide. Thus, a stored inert gas is the most appropriate choice.

4.5.1 Tanks design

Tanks must perform different functions, for example, trivially to contain propellant or pressurant gas, be compatible with them and to bear high pressures without failures. To this purpose, particular attention has to be paid for the material the tanks will be made of. Common materials for tanks are aluminium, stainless steel, titanium, alloy steel and fibre-reinforced plastics with an impervious thin inner liner or metal to prevent leakage through the pores of the fibres. Tanks can be arranged in a variety of ways and they strongly impact the position of the centre of gravity of the spacecraft. It is well known that the optimum shape of a propellant or gas pressurizing tank is a sphere, because it results in a tank with the least weight, at given material and volume. Unfortunately, spheres are not really efficient for using the space in a vehicle.
What is good for hydrogen peroxide is that it is not cryogenic, so it does not require a way to keep its temperature too low, which would be an issue for a heavy insulation.
Not only does the propellant tank hosts the propellant, but it has to comprise what is called ullage as well. The latter is a necessary space, which allows for thermal expansion of the propellant liquids, for the accumulation of gases originally dissolved in the propellant or for gaseous product from possible slow reactions within the propellant during its storage. Usually, the ullage volume is between 3 and 10% of the tank volume. Ullage also varies along with the propellant discharge, with consequent changes in pressure and temperature (even mass if the tank is not sealed) of the gas.

During the discharge, the propellant could remain trapped in grooves or corners of pipes, fittings and valves, could wet the walls, could be caught in instrument taps. These losses have to be taken into consideration, because they slightly reduce the amount of consumable propellant, resulting in an increase of mass in the entire system.

Another problem to be aware of is the sloshing of the liquid in the tank. When a tank is partially empty, its outlet can be uncovered, allowing gas bubbles to enter the propellant discharge line. This could cause severe problems in the decomposition chamber. Moreover, more importantly, sloshing causes shifts in the centre of gravity of the vehicle, which, in the case of a small spacecraft, is not negligible at all and can lead to several difficulties during the flight.

Independently on the shape, internal volume and pressure are the parameters which mostly affect the design of a tank. As an example, the total internal volume for the propellant tank is determined by the mass of required propellant (output from mission analysis) and the ullage volume to be inserted. Then, the pressure at which the propellant has to be stored will play an important role in the evaluation of the minimum wall thickness of the tank. Usually, for pressurised feed systems, the average pressure in the propellant tank is typically between 13 bar and 90 bar. Instead, for the pressurant tanks, usually the pressure is set between 69 bar and 690 bar [33]. However, some comparisons and analysis can be carried out, fixing some parameters and varying others, in order to identify which quantity has the major impacts on the system. In fact, there are many variables, like the material, regulation pressure, ullage, type of propellant or pressurant, geometry and safety factors.

To size the wall thickness of the tank, the equilibrium of external forces (pressure inside the tank in this case) and internal forces (reaction stresses of the material) can be evaluated.

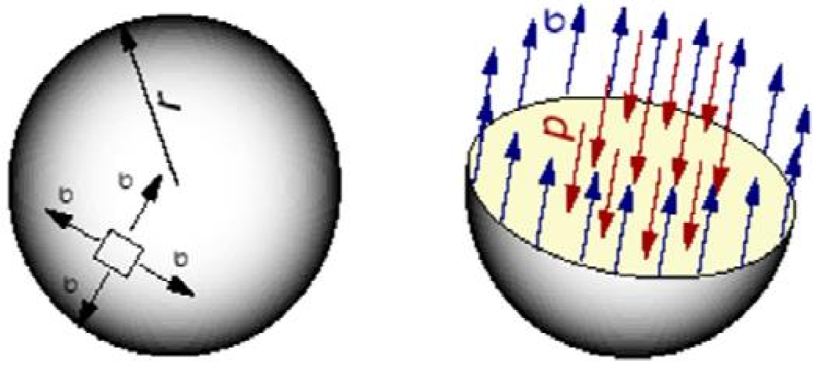


Figure 24 Spherical tank stress [20]

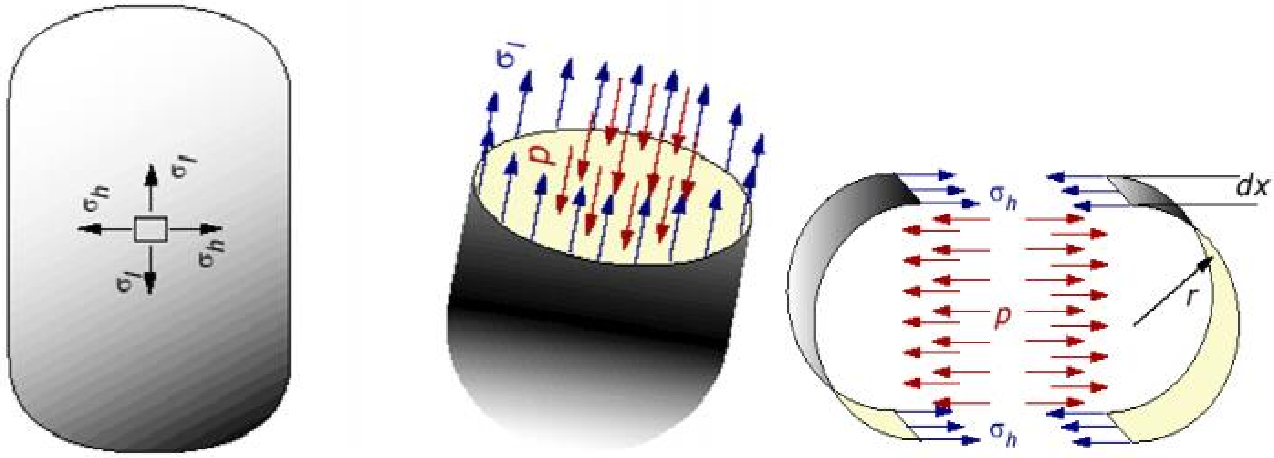


Figure 25 Cylindrical tank stress [20]

Figure 24 and Figure 25 show the stresses arising in the material as the result of the internal pressure. The pressure tends to stretch the material, while the material tends to resist. Here, two different shapes are reported: sphere and pill-shaped. The pill-shaped tank is composed by two semi-spheres merged by a cylinder. The principle with which the equilibrium is written is identical, but it differs between the two shapes because of geometrical reasons.

In the case of the sphere, the maximum force exerted by the pressure is at the diameter of the sphere, where the surface created by the intersection of a plane with the sphere is maximum. That surface is obviously a circle with a radius which depends on the volume of the tank. The resisting portion of the tank is an annulus, with the same internal radius of the circle and an external radius which depends on the wall thickness, which is the goal of this evaluation.

$$p\,\pi r^2 = \sigma\, 2\pi r_m t$$

For thin shells and as a first iteration, one can assume that the medium radius of the annulus is approximately the internal radius

$$r \sim r_m$$

so that the equilibrium can be rewritten as

$$p\pi r^2 = \sigma 2\pi r t$$

Re-arranging this equation, one can obtain the expression for the wall thickness

$$t_{sphere} = \frac{pr}{2\sigma}$$

In the case of the pill-shaped tank, writing the equilibrium of forces for a section of cylinder, it can be noted that the longitudinal stress is the same as the sphere, while the hoop stress is different as follows

$$2\sigma_h t d_x = p\ 2r d_x$$

which leads to

$$t_{pill} = \frac{pr}{\sigma_h}$$

One can immediately compare these results. In fact, if one computes the ratio of the evaluated thickness in the two cases, one obtains

$$\frac{t_{pill}}{t_{sphere}} = \frac{\frac{pr}{\sigma_h}}{\frac{pr}{2\sigma}}$$

If the pressure, the radius and the material are the same, it reduces to

$$\frac{t_{pill}}{t_{sphere}} = 2$$

Showing that the wall thickness in case of a pill-shaped tank is two times the one in case of spherical tank. The wall thickness is what makes the total mass raise, because it defines how much volume is necessary to construct the tank. Note that the mass corresponding to that volume is part of the so-called "inert mass", because it is not useful to feed the engine, but it is compulsory to let the engine be fed.
So, in terms of mass and volume, a spherical tank would be more efficient, but one cannot say the same in terms of the way in which that volume is occupied. At a given internal volume of the tank, the spherical tank can assume one only configuration, because the volume directly defines the diameter. Instead, in the case of the pill-shaped, one can vary two parameters: the internal diameter and the height of the cylinder. This makes possible to design different configurations, like thinner and taller tanks, but remember that the pill-shaped tank requires a higher mass. This solution becomes preferable when the propellant mass is high, because the ratio mass of tank/mass of propellant diminishes, so that the surplus of tank mass is better compensated by the propellant mass.

There is another aspect to be taken into consideration and it is about pressure. The internal pressure in the tank is determined by the feed pressure required by the engine and by the type of feed system chosen. Since no devices which augment the pressure are designed (no volumetric pumps), the pressure in the tank must be sufficiently elevated to cope with the pressure loss between outlet of the tank and inlet of the engine.

Anyway, whatever the pressure is, tanks are not usually designed to bear only the nominal pressure, for safety reasons. They are built taking into account the possibility to undergo complications in the fluidic line,

which could make the pressure strongly and suddenly rise. IN other words, a safety factor k is used and it is conceptually equal to design the tank for a pressure which is k-times higher than the nominal one.
So, the final formulas for the wall thickness are

$$t_{sphere} = \frac{kpr}{2\sigma}$$

$$t_{pill} = \frac{kpr}{\sigma_h}$$

If k is the same, the ratio between them is again 2, always at equal conditions.

The materials taken into consideration are presented in Table 23 Types of alloys utilised in the tank design Table 23, respectively for the pressurant and the propellant tanks.

Material	Yield Strength (MPa)	Tensile Strength (MPa)
Titanium Ti-6Al-4V	888	957
St Steel AISI316L (S31603)	170	485

Table 23 Types of alloys utilised in the tank design [41] [42]

Here following, the results for the sole propellant tank are presented. For further information see Annex III – Propellant tank.

	Value
Regulation pressure	20 bar
Propellant	Hydrogen peroxide (92% wt)
Propellant mass	3 kg
Material for tank	St Steel AISI316L (S31603)
Density of metal	8 kg/litre
Material for bladder	VITON
Density of bladder	1,880 kg/litre
Thickness of bladder	5 mm
Safety factor	3
Pill-shaped internal diameter	1,35 dm

Table 24 Fixed parameters for propellant tank design

One may want to notice that the thickness of the bladder is assumed and not calculated. As first approximation, a bladder 5 mm thick with a diameter equal to the internal diameter of the tank (both sphere and pill-shaped) can represent a good compromise, which makes the mass result a little greater. This is really likely, because, as already said, sloshing in space is not tolerable and surely it will be needed the design of a solution to prevent the propellant from move uncontrollably.
Moreover, using St Steel AISI316L is only a first approximation, because it provides scarce performances. The improvement in the materials which can be utilised is fundamental to save mass and space.

The following results are for a regulated pressure system. The blowdown case will be treated separately.

PROPELLANT TANK (SPHERE)	HYDROGEN PEROXIDE			
mass_gas	0,005437054	kg	5,437054465	g
regulation pressure	20	bar	2000000	Pa
Ullage	10%			
vol_prop_onboard	2,135231317	litre		
Volume_internal	2,372479241	litre		
diametre_internal	1,65475842	dm	0,165475842	m
k_safety	3			
thickness (calculated)	0,001460081	m	1,460080959	mm
diametre_external	1,683960039	dm	0,168396004	m
Volume_external	2,607840485	litre		
Volume_metal	0,127831384	litre	0,000127831	m^3
density_metal	8	kg/litre	8000	kg/m^3
mass_metal	1,022651068	kg	1022,651068	g
mass_bladder	0,202156139	kg	202,1561389	g
mass_propellant	3	kg	3000	g
total_tank_prop	4,230244261	kg	4230,244261	g

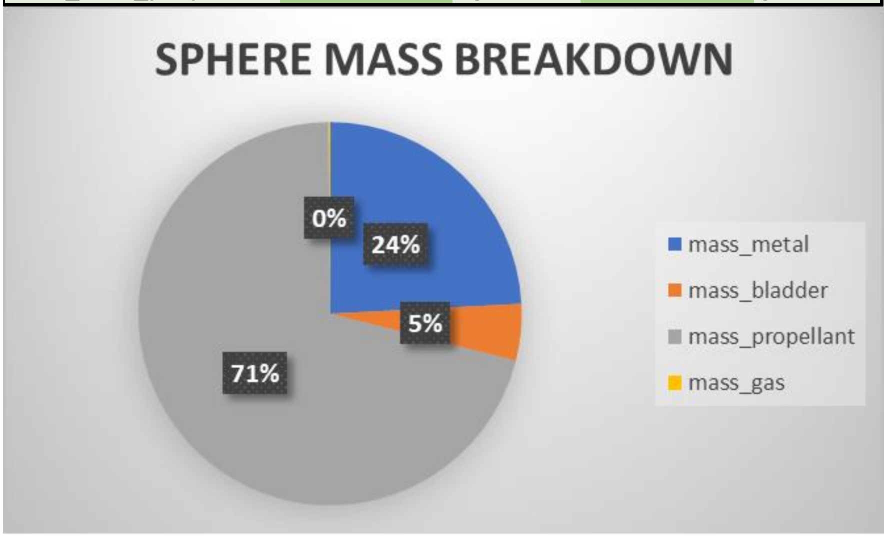


Table 25 Examples of calculations for a spherical propellant tank

TANK PROPELLENTE (PILL - SHAPED)	HYDROGEN PEROXIDE			
mass_gas	0,005437054	kg	5,437054465	g
Volume_internal	2,372479241	litre		
sphere (comparison)	1,65475842	dm		
diametre_sphere (D)	1,35	dm	13,5	cm
height_cylinder (L)	0,757467429	dm	7,574674285	cm
Vol_internal_cylinder	1,084229903			
Vol_internal_sphere	1,288249338			
Vol_interal (check)	2,372479241			
wall thickness	0,002382353	m	2,382352941	mm
diametre_external	1,397647059	dm	13,97647059	cm
Vol_external_cylinder	1,162114377	litre		
Vol_external_sphere	1,429523065	litre		
vol_total_external	2,66320685	litre		
vol_metal_cilynder	0,077884473	litre		
vol_metal_sphere	0,141273728	litre		
vol_total_metal	0,219158201	litre		
density_metal	8	kg/litre		
mass_metal	1,75326561	kg		
mass_propellant	3	kg		
mass_bladder	0,134550486	kg	134,5504864	g
total_tank_prop	4,89325315	kg	4893,25315	g

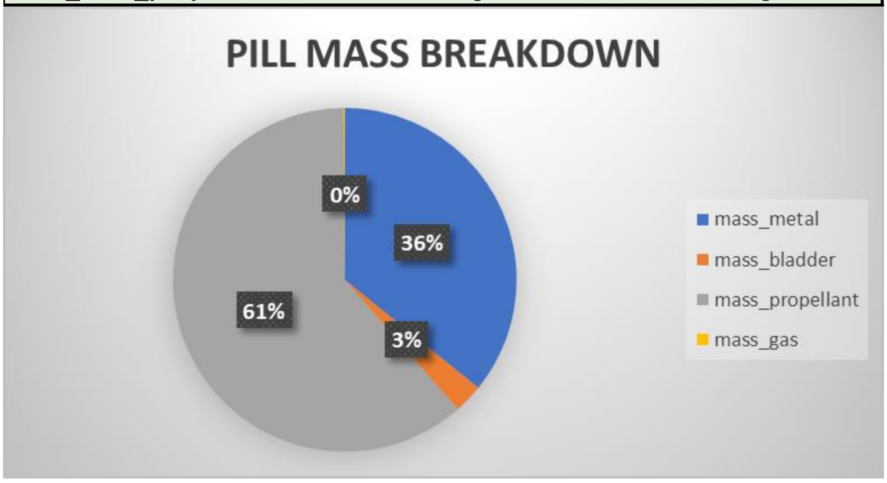


Table 26 Example of calculations for a pill-shaped propellant tank

Table 25 and Table 26 show the difference between a spherical and a pill-shaped tank. If one uses a sphere, he can only control one parameter, at fixed internal volume, which is the diameter. Instead, with a pill-shaped tank, one can modify two parameters, at given internal volume: the diameter and the height. Then, the total resulting height will be the sum of the diameter plus the height of the only cylindrical part. Calculations show that the sphere can be significantly lighter, at the given parameters, because it is more suitable to resist to stresses due to the pressure, resulting in a minor thickness of the wall. The pill-shaped tank results to be heavier, but one can choose to modify arbitrarily one of its two geometrical parameters, the diameter or the height of the cylinder. When one is fixed, at a given internal volume, also the other is defined. The pill-shaped would be preferable, since the available room inside the spacecraft is low, the designer would have more geometrical possibilities among which to choose the optimum shape.

Actually, one can think of splitting one single tank in many, or of building a toroidal tank. These solutions will be discussed later on in the final section of this chapter.

4.5.2 Pressurant mass estimate

Tank pressurization is compulsory to let the propellant naturally flow towards the engine, giving it a differential pressure between tank and engine, talking about both pressure feed systems and pump feed systems. This is the case of a pressure feed system, as already stated. The most common method of pressurization is the use of pressurant gases, among which helium or nitrogen are widely known and utilised. A pressurant gas must be inert, must not condense or be soluble in the liquid propellant. This would increase the mass of required pressurant and thus the inert mass of the system [33]. Sometimes, it is possible to induce a self-pressurization, but this requires a certain type of propellant and it is usually more difficult to control. Even solutions with chemically generated gases would be possible, but in this thesis, for sake of simplicity, only canonical cases will be evaluated.

One can notice that the first part of the gas leaving tie high pressure gas storage tank is at ambient temperature, but if the pressurant expands rapidly, then the gas remaining in the tank undergoes essentially as isentropic expansion, causing the temperature to decrease steadily. So, the last portions of the pressurant gas leaving the tank are much colder than the ambient temperature and readily absorb heat from the piping and the tank walls. The Joule-Thomson effect causes a further small temperature change. Again, for sake of simplicity, these considerations will not be taken into account.

- **Regulated-pressure case**

One can carry out a simplified analysis for the estimation of the required propellant mass on the basis of the conservation of energy principle, assuming an adiabatic process, an ideal gas and a negligibly small initial mass of gas in the piping and in the propellant tank [33]. Here following, the initial conditions in the pressurant tank are expressed with subscript 0, while subscript g and p refer to the gas tank and to the propellant tank respectively.
The first principle of thermodynamics states

$$\Delta U = Q - L \quad (*)$$

Where ΔU is the finite variation of internal energy due to heat exchange Q and work done L. At the initial state, the internal energy of the pressurant is

$$U_0 = c_v m_0 T_0$$

and at the final state

$$U_f = c_v m_g T_g + c_v m_p T_p$$

The work made by the gas on the propellant is

$$L = -pV_p$$

So, equation (*) becomes

$$p_p V_p = m_0 c_v T_0 - m_g c_v T_g - m_p c_v T_p \quad (**)$$

But from the ideal gas law $pV = mRT$ and since $R = c_v(\gamma - 1)$, equation (**) turns into

$$p_p V_p(\gamma - 1) = p_0 V_0 - p_g V_0 - p V_p$$

Thus, the volume of pressurant required is

$$V_0 = \frac{\gamma p_p V_p}{p_0 - p_g}$$

Which leads to the mass of pressurant

$$m_{0_{adiab}} = \frac{\gamma p_p V_p}{1 - p_g/p_0} \frac{1}{RT_0} = \frac{p_p V_p}{RT_0} \frac{\gamma}{1 - p_g/p_0}$$

This relation may spring some comments.
First, one can have a look at the term $p_p V_p / RT_0$: it represents the quantity of gas required to make a V_p volume of propellant flow out the propellant tank. It is useless to say that the more volume has to be ejected at the more pressure, the more mass of pressurant is required. Then, at the denominator one finds the gas constant and the initial temperature in the gas tank. In a sense, storing gas as high temperature, at a given volume and at a given pressure, if the temperature rises the mass decreases, following the ideal gas law $pV = mRT$. A high constant gas R means gas with low molar mass, because $R = \mathcal{R}/\mathcal{M}$.
Then, one can have a look at $\gamma/(1 - p_g/p_0)$, which represents, as a function of the pressure ratio through which the gas expands, a kind of availability of the storage gas. Here, obviously, one can see that the initial pressure has to be the highest possible and, contrarily, the final pressure the lowest possible. In other words, one should give the pressurant the most power possible (p_0 in a sense) and he should utilise it in the best way (so that the final pressure p_g is the lowest acceptable).

In the case of an isothermal expansion, the variation of internal energy is 0, so the expression for the pressurant mass is almost the same, except for the gas specific heats ratio γ

$$m_{0_{isot}} = \frac{p_p V_p}{RT_0} \frac{1}{1 - p_g/p_0}$$

One can immediately notice that

$$\frac{m_{0_{adiab}}}{m_{0_{isot}}} = \gamma$$

to say that the two masses differs for a coefficient, which depends on the type of pressurant is used. Nonetheless, even though an actual process is something between these two cases, considering an adiabatic process is surely conservative.

To sum up, to minimise the pressurant mass required, one needs high storage temperature and pressure, and low final pressure, in addition to low molar masses (high R) and low γ. It is fair to say, however, that a heating system for the pressurant tank could represent a disadvantage more than an advantage, due to the increase in mass and the reliability, but, in a further iteration, it could be taken into account.
There are other important aspects: heat transfers, vaporization of the propellant and heat losses shall be included in a careful analysis, so it is a wise option to think about an excess of pressurant gas. Remember that the above-mentioned equations are valid only under ideal conditions.

Gas	γ	$R\ [J/(kg \cdot K)]$
Nitrogen N_2	1,4	297,7
Helium He	1,667	2077

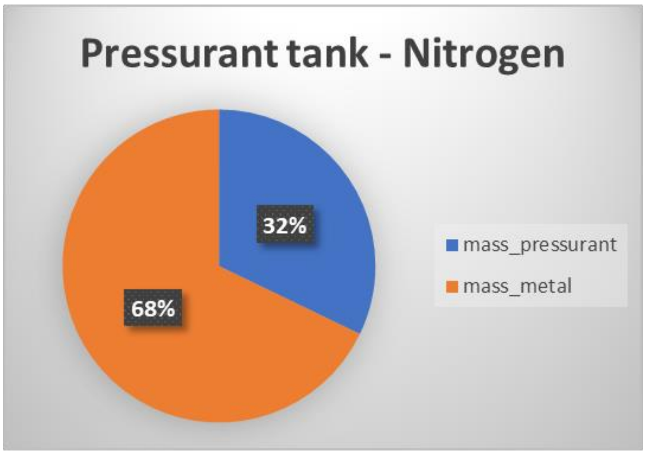
Table 27 Pressurant gases considered

	Value
Storage pressure p_0	350 bar
Storage pressure – end of discharge p_g	25 bar
Regulation pressure p_p	20 bar
Total propellant mass	3 kg
Storage temperature	20°C
Material used	Titanium Ti-6Al-4V
Safety factor	3

Table 28 Fixed parameters

PRESSURANT TANK		NITROGEN		
gamma	1,4			
p0	350	*bar*	35000000	*Pa*
p_end_pressur	25	*bar*	2500000	*Pa*
regulation pressure	20	*bar*	2000000	*Pa*
density_propellant	1,405	*kg/l*		
mass_propellant	2,5	*kg*		
volume_prop_used	1,779359431	*litre*		
majorative factor	1,2		0,705624513	
vol_prop_onboard	2,135231317	*litre*		
V0 (calculated)	0,18395839	*litre*	0,000183958	*m^3*
Diametre_tot	0,705624513	*dm*	0,070562451	*m*
T0	20	*°C*	293,15	*K*
R	297,7	*J/(kg*K)*		
mass_pressurant	0,073776647	*kg*	73,77664674	*g*
k_safety	3			
thickness (calculated)	0,002085883	*m*	2,085883273	*mm*
diametre_total	0,747342178	*dm*	0,074734218	*m*
Volume_total	0,218553167	*litre*		
mass_metal	0,155676493	*kg*	155,6764927	*g*
total_tank_press	0,229453139	*kg*	229,4531394	*g*

Table 29 Example of pressurant gas mass estimate – N_2



PRESSURANT TANK	HELIUM			
gamma	1,667			
p0	350	bar	35000000	Pa
p_end_pressur	25	bar	2500000	Pa
regulation pressure	20	bar	2000000	Pa
density_propellant	1,405	kg/l		
mass_propellant	2,5	kg		
volume_prop_used	1,779359431	litre		
majorative factor	1,2		0,747898819	
vol_prop_onboard	2,135231317	litre		
V0 (calculated)	0,219041883	litre	0,000219042	m^3
Diametre_tot	0,747898819	dm	0,074789882	m
T0	20	°C	293,15	K
R	2077	J/(kg*K)		
mass_pressurant	0,012591249	kg	12,59124905	g
k_safety	3			
thickness (calculated)	0,00221085	m	2,21084955	mm
diametre_total	0,79211581	dm	0,079211581	m
Volume_total	0,260234378	litre		
mass_metal	0,185366224	kg	185,3662238	g
total_tank_press	0,197957473	kg	197,9574728	g

Table 30 Example of pressurant gas mass estimate – He

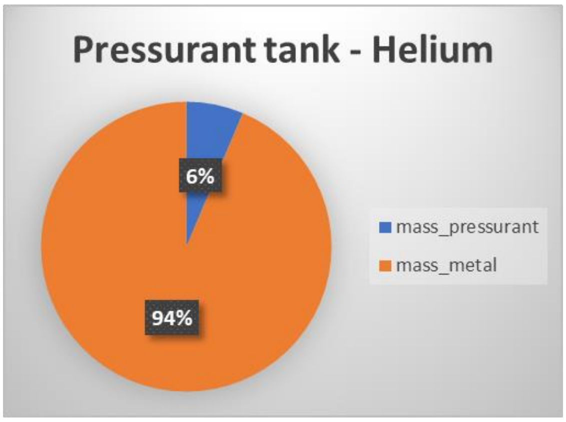


Table 29 and Table 30 show an example with nitrogen and helium respectively, using spherical tanks. These results are for the pressurant tant in case of pressure regulated system. Blowdown will be treated separately. In general, using Helium, the mass of the metal for the tank is greater because of the greater volume required (because of the greater γ), given the other conditions, but the pressurant mass is definitely lower (because of the almost ten times smaller gas constant). Between the two effects, the one which prevails is the second one, and the overall mass is lower with helium. Despite that, helium needs a larger diameter and so it needs a greater volume with respect to nitrogen. To conclude, if one wants to minimise mass helium is the best solution, while if one wants to minimise the total volume nitrogen is preferable.
Then, one may prefer to set very high pressures to save space in terms of minor external diameter.
Here, only spherical shapes are presented because pressurant needs small volume, so a sphere is acceptable, contrarily to what seen for the propellant tank.

Note that if cold gas ACS thrusters were designed, an additional amount of pressurant mass could be required and more precise analyses would be fundamental.
For further information, it is advisable to see Annex IV – Pressurant Tank.

- **Blowdown case**

The evolution of the pressure inside the tank depends on the transformation. One can assume an adiabatic process if all the propellant is used in one shot, while isothermal is the propellant is used with small shots many times.

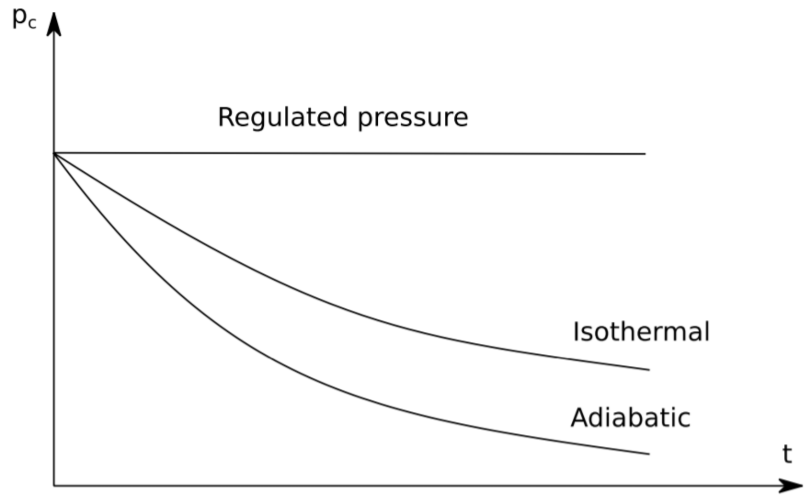


Figure 26 Qualitative comparison of chamber pressure in various cases

Figure 26 [43] clearly shows that, in case of blowdown, independently on the transformation, the pressure in the decomposition chamber cannot remain constant, for no external gas supply is possible and thus, with the propellant flowing towards the engine, the pressurant expands and gradually loses its pressure.
Usually, a blowdown ratio is decided during the trade-off. It is the ratio between the final volume of pressurant and the initial inside the tank

$$B = \frac{V_{gf}}{V_{gi}}$$

Usually, B is about between 3 and 6, but it depends on the requirements of the propulsion system. One can refer to the volume of propellant as well, because the difference between the final volume of pressurant and the initial is just the volume of propellant consumed

$$V_p = V_{gf} - V_{gi}$$

Thus, one can write

$$\frac{V_p}{V_{gi}} = \frac{V_{gf} - V_{gi}}{V_{gi}} = B - 1$$

So, V_p is known, because it represents the quantity of consumable propellant, B is choosen during the trade-off and so V_{gi} is defined

$$V_{gi} = \frac{V_p}{B - 1}$$

Finally, one can find the pressurant mass by the law for ideal gases

$$m_0 = \frac{p_i V_{gi}}{RT_0}$$

Here, some comments similar to the ones for the regulated pressure case may be done. The volume of propellant cannot be modified, once the flight plan is determined. So, at a given V_p, one can augment B in order to have minor initial pressurant volume and so minor initial pressurant mass. However, note that the less the V_{gi} is, the more the gas will have to expand, if V_p is high. And if the pressurant expands a lot, it also loses very much pressure according to the law of ideal gases. This is to say that, if the engine must work with a minimum pressure, the initial pressure inside the propellant tank must be high enough not to fall under a certain value. And now, the higher the initial pressure is, the higher the pressurant mass is and the thicker the walls of the tank are, with a double increase in mass of the system. It is clear that a careful trade-off must be carried out.

Storage pressure p_0	50 bar
Storage temperature	20°C
Total propellant mass	3 kg
Metal used	St Steel AISI316L
Density of metal	8 kg/litre
Material for bladder	VITON
Density of bladder	1,880 kg/litre
Thickness of bladder	5 mm
Safety factor	3
Pill-shaped internal diameter	12 cm
Blowdown Ratio[3]	4

Table 32 Fixed parameters for the design of the tank in a blowdown system

SPHERE case		**T0**	20	*°C*	293,15	*K*
		p0	50	*bar*	5000000	*Pa*
		R	297,7	*J/(kg*K)*		
		m0_pressurant	0,040777908	*kg*	40,77791	*g*
		V0	0,000711744	*m^3*	0,711744	*litre*
		V_internal_tank	0,002846975	*m^3*	2,846975	*litre*
		Internal diametre	0,175844321	*m*	1,758443	*dm*
		k (safety)	**3**			
		thickness (calculated)	0,003878919	*m*	3,878919	*mm*
		diametre_external	0,183602159	*m*	1,836022	*dm*
		volume_external	0,003240649	*m^3*	3,240649	*litre*
		volume_metal	0,000393674	*m^3*	0,393674	*litre*
		mass_metal	3,149390136	*kg*	3149,39	*g*
		mass_bladder	0,228283452	*kg*	228,2835	*g*
		mass_propellant	3	*kg*	3000	*g*
		mass_tot_tank_prop	6,418451496	*kg*	6418,451	*g*
	MASS	**prop/tank (mass)**	0,467402457		46,74	%
		metallo/tank (mass)	0,490677563		49,07	%
		pressur/tank (mass)	0,006353232		0,64	%
		bladder/tank (mass)	0,035566749		3,56	%
	VOLUME	**prop/tank (vol)**	0,658890059		65,89	%
		metallo/tank (vol)	0,121479921		12,15	%
		pressur/tank (vol)	0,21963002		21,96	%
		bladder/tank (vol)	0,000121427		0,01	%

Table 31 Evaluations for the tank - Blowdown spherical case

[3] Magellan Spacecraft had Blowdown Ratio of 4

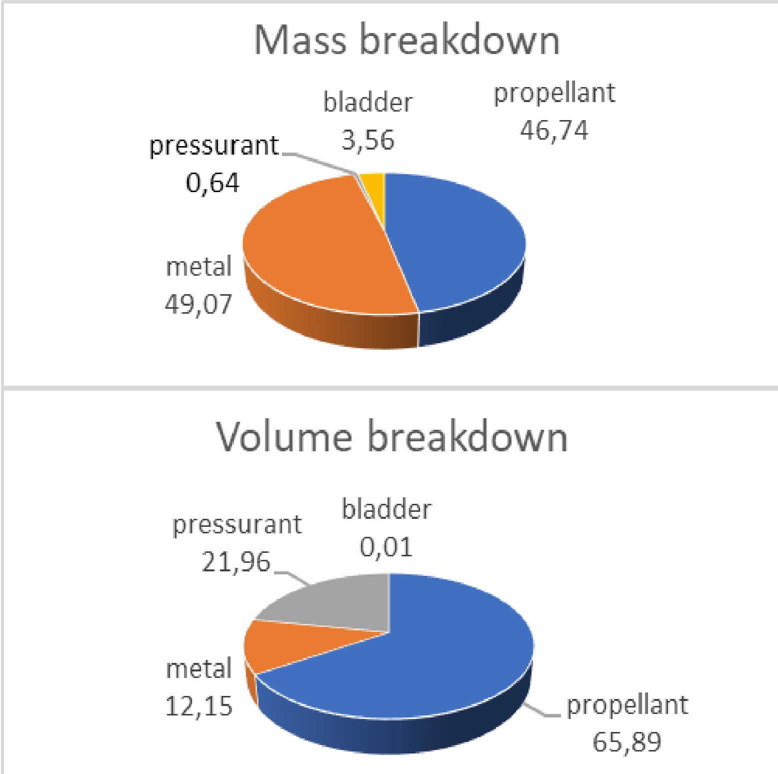


Table 33 Mass and volume breakdown - Blowdown spherical case

PILL-SHAPED case	T0	20	°C	293,15	K
	p0	50	bar	5000000	Pa
	R	297,7	J/(kg*K)		
	m0_pressurant	0,040777908	kg	40,77791	g
	V0	0,000711744	m^3	0,711744	litre
	V_internal_tank	0,002846975	m^3	2,846975	litre
	diametre_internal	0,12	m	1,2	dm
	k (safety)	3			
	height_cylinder (L)	0,171727866	m	1,717279	dm
	Vol_int_cylinder	0,001942196	m^3	1,942196	litre
	Vol_int_sphere	0,000904779	m^3	0,904779	litre
	thickness calculated	0,005294118	m	5,294118	mm
	diametre_ext	0,130588235	m	1,305882	dm
	Vol_cil_ext	0,002300058	m^3	2,300058	litre
	Vol_sphere_ext	0,001166033	m^3	1,166033	litre
	Vol_total_ext	0,003466091	m^3	3,466091	litre
	Vol_metal_cylinder	0,000357861	m^3	0,357861	litre
	Vol_metal_sphere	0,000261254	m^3	0,261254	litre
	Vol_metal_total	0,000619116	m^3	0,619116	litre
	density_metal	8000	kg/m^3	8	kg/l
	mass_metal_cylinder	2,862891586	kg	2,862892	kg
	mass_metal_sphere	2,090033236	kg	2,090033	kg
	mass_metal_tot	4,952924822	kg	4,952925	kg
	mass_bladder	0,106311495	kg	106,3115	g
	mass_propellant	3	kg	3000	g
	mass_tot_tank_prop	8,100014226	kg	8100,014	g
MASS	prop/tank (mass)	0,37036972		37,04	%
	metal/tank (mass)	0,611471126		61,15	%
	pressur/tank (mass)	0,005034301		0,50	%
	bladder/tank (mass)	0,013124853		1,31	%
VOLUME	prop/tank (vol)	0,616034463		61,60	%
	metallo/tank (vol)	0,178620716		17,86	%
	pressur/tank (vol)	0,205344821		20,53	%
	bladder/tank (vol)	0,016314826		1,63	

Table 34 Evaluations for the tank – Pill-shaped case

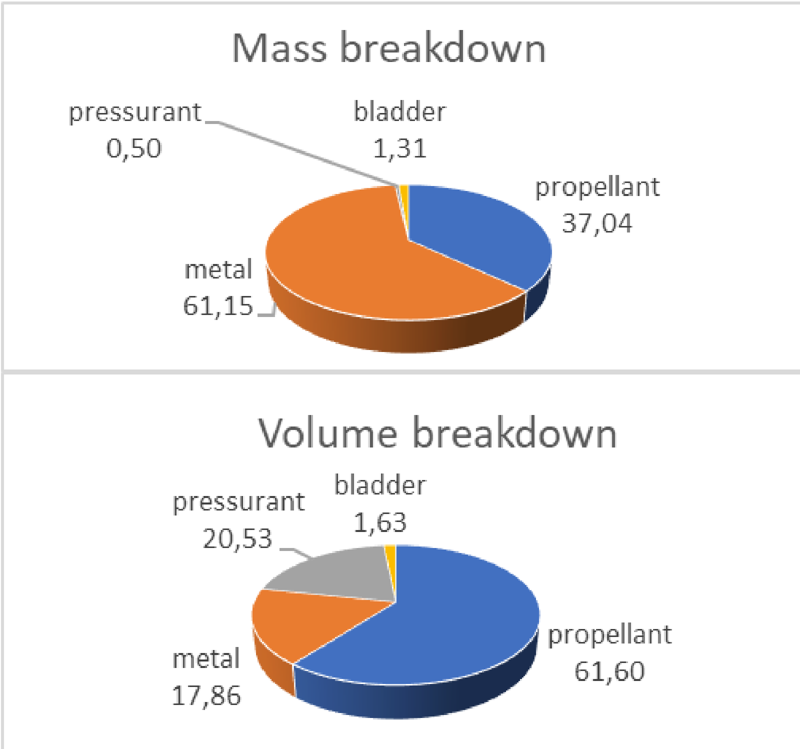


Table 35 Mass and volume breakdown - Pill-shaped case

With respect to the pressure regulated case, blowdown case would require higher masses and volumes. This is due to the significantly greater storage pressure needed to guarantee an acceptably high feed pressure for the engine. Remember that on a blowdown case, no gas refilling occurs, leading to a continuously reducing pressure. This will be analysed in Section 4.5.4.

4.5.3 Regulated pressure

Among the pressure feed systems, regulated pressure feed system is usually the preferable one, looking at the performances of the engines, because it is able to keep a constant pressure in the propellant tank, but it has some disadvantages as well.

Pressure/thrust	Essentially constant
Required components	Regulator Filter Gas valve Gas tank
Gas storage	Separate high-pressure tanks
Advantages	Essentially constant propellant flow Essentially constant thrust Essentially constant specific impulse
Disadvantages	More complex Pressure drop introduced by the regulator High pressure gas storage Shorter burning time

Table 36 Main features of a regulated pressure feed system [33]

Table 36 shows the main features of a regulated feed system. A pressure regulator, situated between the outlet of the pressurant tank and the inlet of the propellant tank, maintains a constant pressure inside the latter, as the propellant flows towards the engines to be consumed. This makes the engine always receive the propellant at an almost constant pressure, letting its functioning provide essentially constant thrust and specific impulse.
On the other hand, the pressure regulator is usually a heavy and cumbersome device, relatively to the small amount of space available inside the spacecraft. So, this would surely be a preferable option if one could build a sufficiently small pressure regulator.
Moreover, in this case at least two tanks are compulsory: one for the pressurant and another for the propellant, contrarily to a blowdown case.
The evaluations in terms of volume and mass for the two tanks have been presented in Sections 4.5.1 and 4.5.2. What it has not been presented yet is the fluidic line that links them to the engine.

Designing a complete fluidic line is a tough task, especially during a high-level study, like this is. This is because many of the components may have to be tailored ad hoc for the mission. In fact, since the available room is extremely low (12U), already existing components could be unsuitable. As an example, a spherical tank would not be the best solution and in general, no component is allowed to be too cumbersome. This is particularly complicated for valves.
Moreover, the fluidic line has to satisfy ECSS's rules [44], which may be strict in some sections.

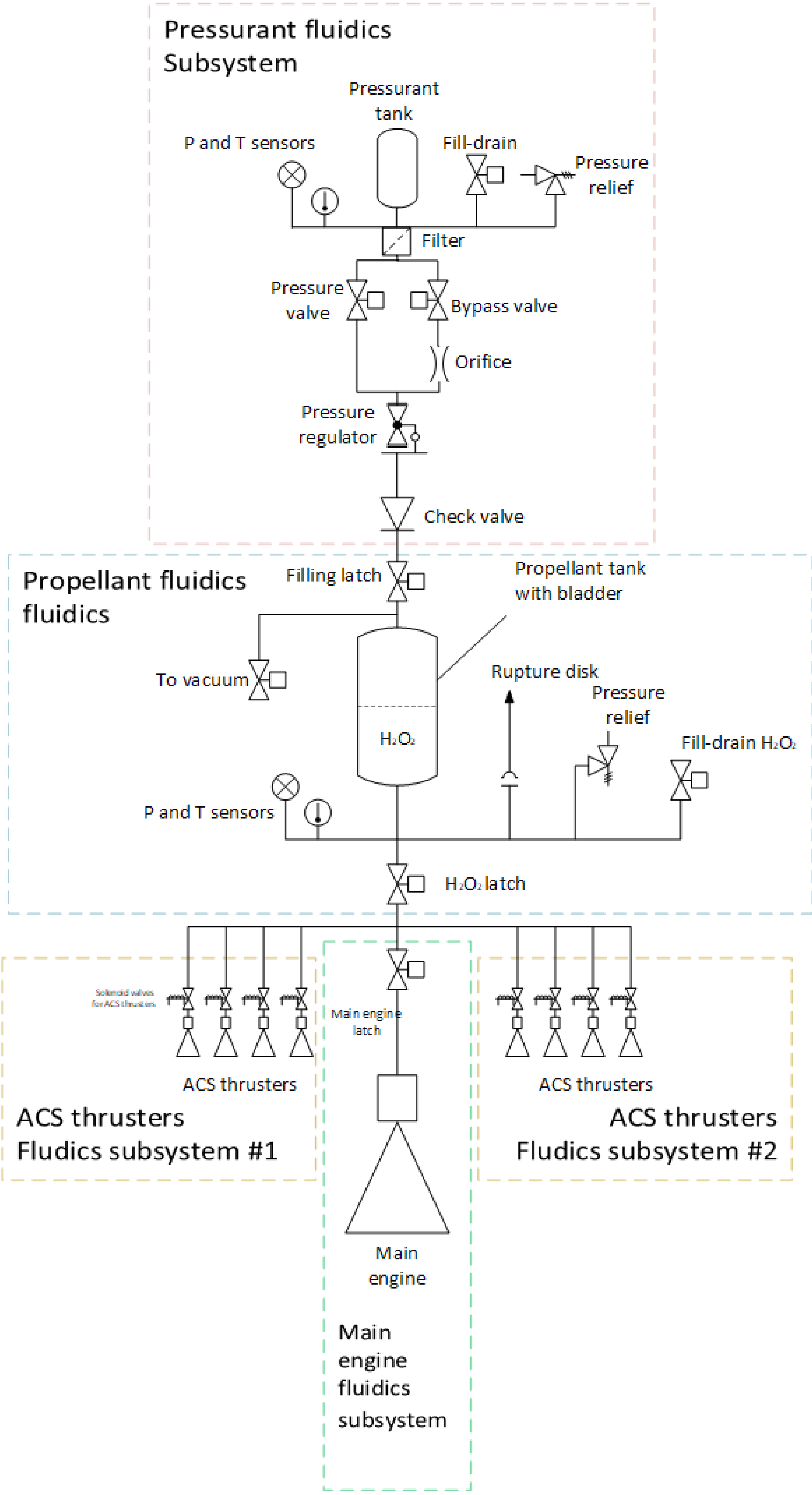


Figure 27 Block scheme of a hypothetical pressure regulated system

Figure 27 shows a hypothetical configuration for a pressure regulated feed system. This would be surely the most suitable looking at the performances of the engine, because it would be able to guarantee almost constant pressure in the decomposition chamber. What it is true, on the other hand, is that it would require much more complexity and components than a simpler system, like a blowdown system.
As it can be seen, two different tanks (described in the sections above) are present, linked by different devices.

Since ECSS's 5.2.1 section speaks about pre-launch and launch activities, the system shall be able to be filled with propellant a short while before launch, so fill and drain devices must be considered. Another important aspect, concerning the handling of hydrogen peroxide, is the so called "don't pour rule. In fact, the propellant tank shall be filled letting the propellant flow from an end to another, being sucked and not injected. At this purpose, the propellant would be put in contact with the "Fill-drain H2O2" valve and sucked by a device connected to the "to-vacuum" valve. So, these two components would increase the dry "useless" mass. Useless is not a proper term, because it is not completely true. Those valves are compulsory to place the propellant in its tank, but, once that operation finished, they would not have any propulsive aim at all. Despite that, they can not be removed.

It is trivial to say that sensor for temperature and pressure serve to give health-monitoring and performances data.

Pressure relief valves are necessary to prevent the line from excessive pressure due to malfunctioning or faults. One may remember that if the fluidic line fails, the whole mission fails.

A filter is put just after the pressurant tank, but why there is not another after the propellant tank? Usually, filters are placed after the outlet of the propellant tank, in order to prevent valves and the engines from receiving impurities. This would be true even in this case, but unfortunately hydrogen peroxide is very reactive and a filter could be a potential way to trigger an undesirable reaction. A careful risk analysis will decide whether or not to put a filter on the propellant line.

After the filter, two different ways are designed, with two different paths. This is for a safety reason. The pressurant is stored at a very high pressure (current design is 350 bar, but however that pressure will be extremely high with respect to that in the propellant tank) so the opening of a single valve would be critical for the hydrogen peroxide. In fact, the characteristic time of actuation of a valve is at most tenths of a second, so, in that time, the pressurant would rush towards the propellant tank, causing a strong blow increasing the pressure extremely rapidly. This would mean an isentropic compression at constant volume, leading to a suddenly increase in the temperature and a great heat flux to the hydrogen peroxide. This could be potentially fatal for the propellant tank, so another solution should be found. The latter is to put another valve together with an orifice. At the opening of the valve, the pressurant would flow with more difficulty into the orifice letting the pressure inside the tank rise much less rapidly. Doing so, the heat would be more easily dissipated through the wall of the tank, instead of the propellant, considerably lowering the risk. The other line would be utilised once the pressure regulator started to work properly. This being said, mass increases because one single line would be split into two.

The pressure regulator is the most critical part in this study, because it has role which is as fundamental as complex. It should maintain a constant pressure in the propellant tank, but the thing is that on the market, at the current state of the art, it may probably be really cumbersome. So, if one would be able to tailor a pressure regulator capable of working with extremely high differential pressures in an extremely small amount of space this would definitely be suitable for this task.

Another important thing to consider is ECSS's section 6.5.4, in particular 6.5.4.1, which is about safety barriers. The precise wording will be reported: "The flight version of the system should be divided into independent subsystems separated by safety barriers such as pyrovalves, latch valves, burst membranes and electrical switches and connectors". And again, in Section 6.5.2 Inadvertent Operation, of ECSS-Q-ST-40C reports: "Inadvertent operation of a safety-critical function shall be prevented by:

1) two independent inhibits, if it induces critical consequences , or
2) three independent inhibits, if it induces catastrophic consequences."

Here, only two are presented and are the "H_2O_2 latch" and the main engine latch, for sake of simplicity; then the rupture disk is also considered. Another barrier could be implemented and it may be a normally-closed valve. Anyway, "H_2O_2 latch" can isolate all the engines, while the main engine latch is devoted to the only main engine, as the name suggests. Each of the eight ACS thrusters has its own solenoid valve.
A more detailed discussion on valves will be presented in Section 4.5.6.

No vent valves are presented in Figure 27, but they would very likely be considered, especially during the flight from Earth to the Moon, to let vapours from hydrogen peroxide be jettisoned.

Another aspect to be considered is the efficiency in regulating the pressure before the propellant tank. Usually, at a first glance, the efficiency would be very high, but the transient could be too sluggish. Another idea would be to consider a cavitating venturi in the propellant line to control the mass flow instead of the pressure. The design of the feed system would requires several iterations and other analyses, but, despite that, one may want to remember that the strength of a regulated pressure feed system is the capacity to maintain almost constant performances, which would be perfect for the flight profile. All of these considerations would imply careful analyses of the risks annexed to each of them. See Section 4.5.6 for further details.

Temperature in the tank shall be monitored as well. It is difficult to make a preliminary estimate of its behaviour, thermodynamically speaking, since continue refilling of pressurant is ensured. Anyway, this problem has not to be undervalued, because it could lead to formation of ice inside the tank, which is not desirable.

4.5.4 Blowdown

A blowdown system consists of having both pressurant and propellant inside a single tank, generally at a higher pressure with respect to the propellant tank in case of the regulated pressure case.

Pressure/thrust	Decreasing with the consumption of the propellant
Required components	Larger and heavier propellant tanks
Gas storage	Inside propellant tank with large ullage volume (30-60%)
Advantages	Simpler system Less gas required Probably less inert mass
Disadvantages	Thrust decreases with burn duration Engine must be stable over a wide range of thrust values Propellant stored under pressure Slightly lower specific impulse

Table 37 Main features of a blow down feed system [33]

Table 37 shows some features of a blowdown feed system. Despite it is simpler, it has disadvantages which could not be negligible. Firstly, since the pressure decreases, as shown in Figure 26, the engine could not work properly and the thrust is intended to decrease as well, which could not be acceptable for the flight plan. Then the specific impulse would decrease in time, reducing the efficiency of propellant consumption. Finally, high pressure would set inside the tank and thus the propellant would be stored under pressure, which could cause different problems.

	9,806	*m/s^2*		
Specific Impulse	155	*s*		
c	1519,93	*m/s*		
Thrust	50	*N*		
MASS FLOW	0,032896	*kg/s*		
	32,89625	*g/s*		
	1,404822	*litre/min*		
m_propellant	2,5	*kg*		
V_propellant	0,001779	*m^3*	1,779359	*litri*
majorative factor	1,2			
V_prop_onboard	0,002135	*m^3*	2,135231	*litri*
m_prop_onboard	0	*0*	3	*kg*
density prop	1405	*kg/m^3*	1,405	*kg/litre*
GAMMA	1,4			
m0_pressurant	0,040778	*kg*		
R	297,7	*J/(kg*K)*		
p0	50	*bar*	5000000	*Pa*
T0	20	*°C*	293,15	*K*
Blowdown ratio	4			
initial gas volume	0,000712	*m^3*	0,711744	*liter*

Table 38 input parameters for the preliminary blowdown analysis

With these initial parameters, supposing a continue nominal functioning by the engine and nitrogen as pressurant, results are as follows.

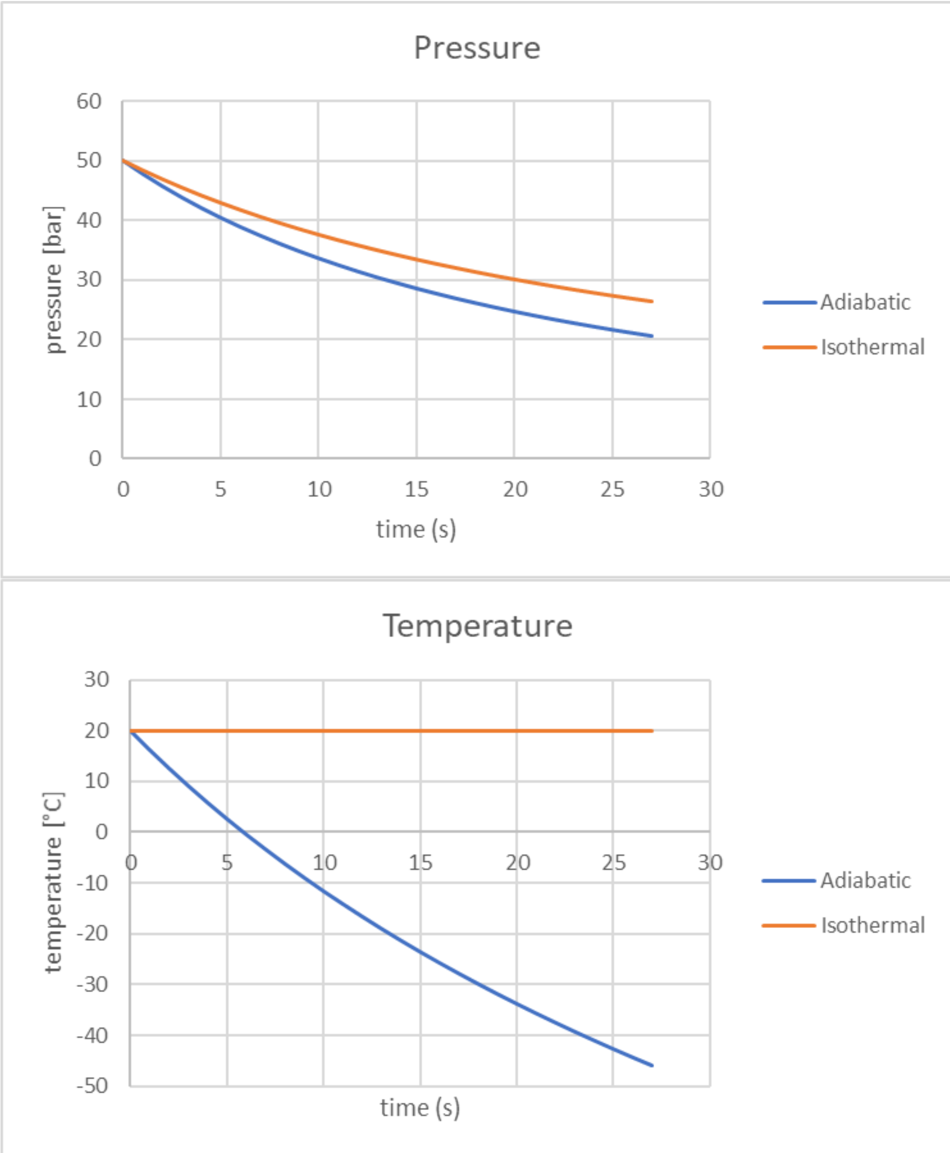


Figure 28 Results from a preliminary analysis of blowdown case

Figure 28 show how pressure and temperature would vary considering ideal transformations and a continue constant mass flow. As the propellant gets consumed, the gas expands and its pressure decreases, even in the best case (adiabatic), reaching low values after a short while. This could be unacceptable for the correct functioning of the engine, supposed to work between 7 and 15 bars (see Section 4.4). Surely, if no device devoted to regulating somehow the pressure is considered, performances will decrease with time. There will be a significant difference between BOL and EOL conditions. The latter should be carefully analysed in a further iteration.
The temperature decreases as well, in the case of an adiabatic process, going towards critical values. One may not forget that hydrogen peroxide could start to solidify, as well as its vapours. Moreover, even plastic sealings may have problems of glass transition. None of these situations would be desirable.

Increasing thrust would mean to increase the mass flow, thus to accelerate the expansion of the gas and in turn to have losses in performances earlier. Helium, due to its much higher gas constant, would lose temperature and pressure more rapidly than nitrogen, at a given mass flow.

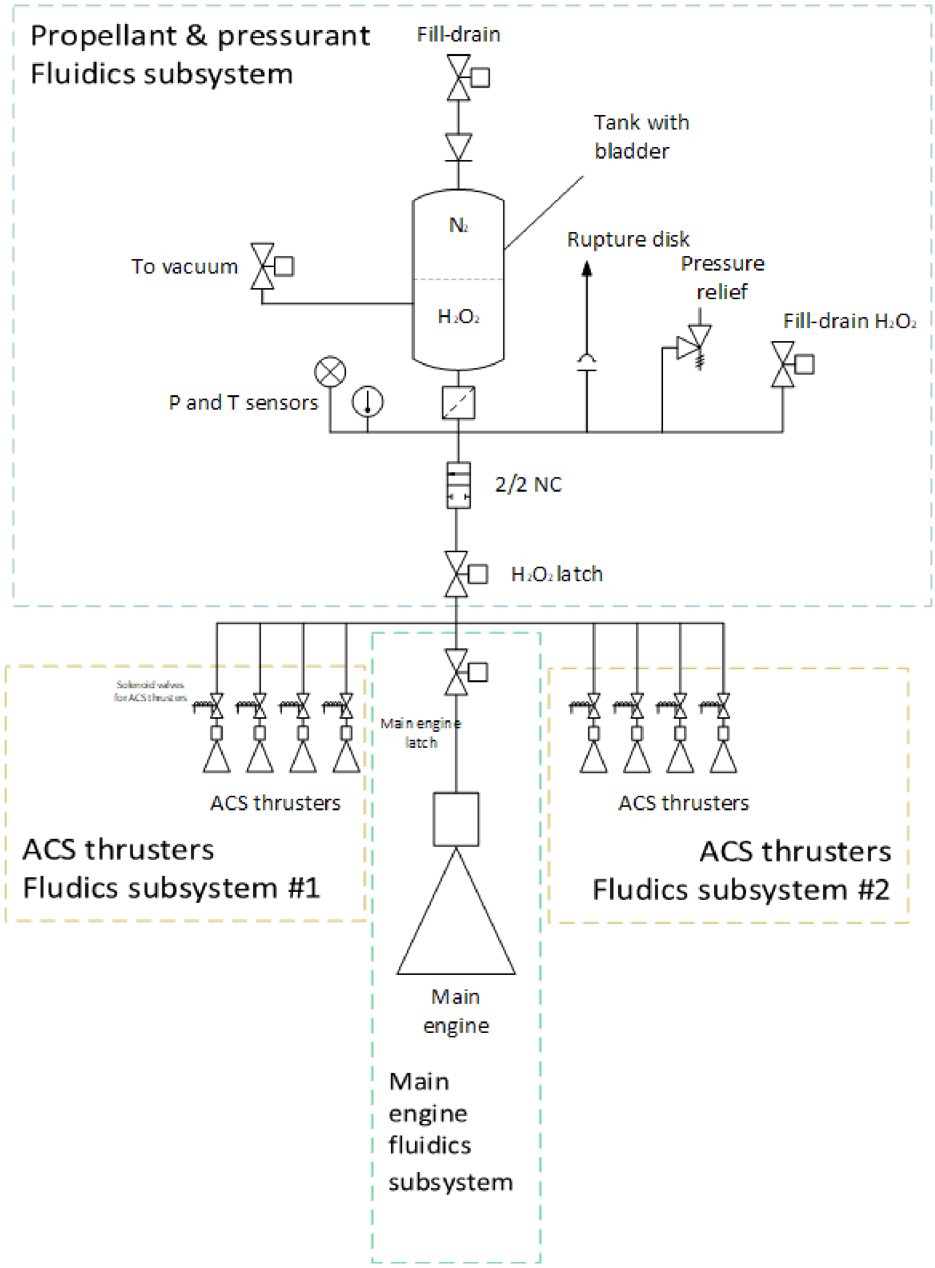


Figure 29 Block scheme of a hypothetical blowdown system

Whatever the variations of the thermodynamical quantities, an example of blowdown system could be represented as in Figure 29. The latter is substantially different from the previous Figure 27 for the pressure regulated example only for the absence of a separate tank for the pressurant. Here, a filter has been positioned after the propellant tank, but what being said in Section 4.5.3 is again true. A two ways normally closed valve, cited in the figure with the name “2/2 NC”, has been inserted to represent the third barrier, as mentioned in Section 4.5.3. A blowdown system is simpler in terms of necessary components, but the tank may require a huge amount of metal mass, if the storage pressure is much high, because stresses could significantly rise. So, the complexity and the aggravation in mass due to the many required valves in a pressure regulated system may not be overcome by a blowdown system, which in turn would result heavier and more cumbersome, in the sense that particular-shaped customised tanks may not be able to cope with high pressures. One great solution would be a blowdown system, but with a device capable of controlling the mass flow arriving to the engine. Pintle valves could be a solution to be designed, even though, actually, since very little mass flows are required, the pintle and its arrangement would be so small that the state-of-the-art-techniques could not be able to build such tiny components.

4.5.5 Blowdown with repressurisation

This section represents a hypothetical idea of an alternative hybrid solution between pure blowdown and pressure regulated feed system. No devices work as pressure regulator. As the propellant gets consumed, the pressure in the tank decreases. When the pressure reaches such low values that the engine cannot work properly anymore, a control logic drives a solenoid valve, which makes some pressurant flow inside the tank, repressurising it. This process is repeated until it is necessary. In this case, the advantage of the pure blowdown is achieved (no pressure regulator), likewise the one of the pressure regulated (low pressure in the propellant tank). Here the problem is that a fast and precise control logic has to be implemented to monitor the pressure inside the tank and consequentially open the valve at the right time for the right duration in order to achieve a correct repressurisation. In this case, thrust and specific impulse would not be constant, but it is possible to find a way to make them vary between an arbitrarily wide range of values, acceptable for the engine. It is all about to manage to implement a strong control logic.

Just like in the case of pressure regulated system, evaluations to estimate how much pressurant gas will be needed are tough. This is because one should define the range of pressure acceptable inside the propellant tank and the management of the internal temperature. As a first extremely coarse approximation, one could think that, given the consumed propellant mass flow and the other parameters, the mass of pressurant required should be at least that of a pure regulated pressure case. This is because in that case, the gas would continuously flow from the pressurant tank to the peopellant tank, without delays. Instead, if the propellant tank were not constantly refilled, its gas inside would lose more pressure and more temperature, so a greater amount of pressurant mass would be required to restore the initial pressure.

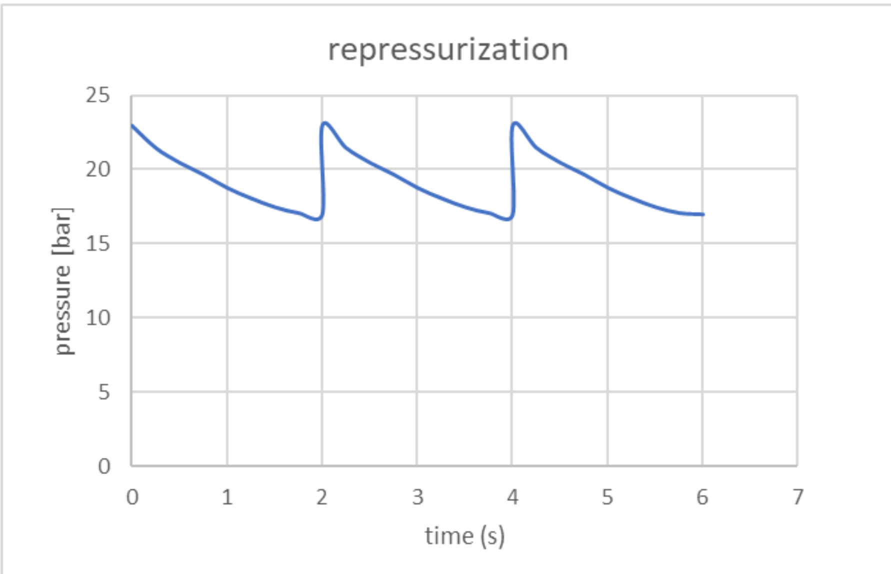


Figure 30 Qualitative example of pressure behaviour across 20 bar

Figure 30 shows a qualitative pressure discharge-recharge in a hypothetical repressurized system, which would work from 23 bar to 17, with refilling each 2 seconds. The right timing of the refilling should actually be decided by a computer, receiving data from sensors monitoring the tank. The mean value across which to work should be decided as a function of the performances of the engine.

Another idea could be the “one-shot repressurisation”, instead of multiple ones. In this case, the range of pressure drop should be much higher, under the same other conditions as before. The advantage would be to have less complicated control logic to decide when and how to refill the pressurant tank.

4.5.6 Valves and tubes

Liquids and gases in the propulsion system need to be controlled and conducted to the intended components: valves and tubes play this role. The required characteristics of valves are reliability, lightweight, leakproof and capability to withstand vibrations and loud noises.

Often the design details, such as clearance, seat materials or opening time delay present development difficulties. What is even worse is that a valve failure can cause a failure of the propulsion unit itself.
In this study, the main driver of the design of the system is the availability of space. Given the amount of propellant required, tanks are compulsory, and the only thing one can do is to reduce their inert mass and their volume, changing their shape, material etc. Tubes are compulsory as well and, given the mass flow rate, one can immediately know what their encumbrance will be. It is trivial to say that at least one engine is compulsory, searching a trade-off between encumbrances and optimal thrust to be provided. Even valves are compulsory, but they are the element on which one can work the most, primarily thanks to the developing nowadays techniques for working metal, just as additive manufacturing. The presence or the possibility to design miniaturised valves here in this case can make the difference between the technological success of this mission or the contrary. Technological progress is what will allow engineers to build smaller and smaller devices.

Looking at the two similar scheme of fluidic lines, in Figure 27 and Figure 29, many components can be seen, even though they are likely less the necessary number. As an example, it is sufficient to think about the deactivation barriers for the engine, compulsory from the ECSS’s rules. This is for safety reasons, since if one only valve were present and it failed, the engine would start to work uncontrollably, with critical consequences. That is why redundancy is applied to the propellant line, with valves capable of shutting off the line. Normally closed and latch valves are common to play this role.
Among the factories present on the internet, Ham-Let, Swagelok, Marotta, Vacco, Ariane-Group, Rafael, MOOG, IHI-Aerospace and others may be considered as references for types of COTS available valves and their encumbrances.

		PR = pressure reg, BD = blowdown			
Type of device	**Manufacturer**	**#PR**	**#BD**	**Power [W]**	**mass [g]**
Relief valve (manual)		2	1	0	ND
Fill-drain valve (manual)	Omnidea-RTG	3	2	0	91
Latch valve (electric)	Marotta	5	2	8	100
Normally-closed valve (electric)		0	1		
Pressure regulator (electric)	Omnidea-RTG	1	0	ND	1100
Check valve (manual)	Marotta	1	1	0	40
Rupture disk (manual)		1	1	ND	ND
Solenoid ACS valve (electric)		8	8	0,25	5
Vent valve (manual)	Rafael	1*	1*	0	60
Sensors (electric)		4	4		
Pressurant filter	Omnidea_RTG	1	0	0	76
Propellant filter	Omnidea_RTG	0	1	0	110

Table 39 Sum-up of devices in the fluidic line

Table 39 shows an overview of the components (excluded tanks and pipelines and fittings) in the fluidic lines for both blowdown and pressure regulated. Vent valves have a * because they are not in the scheme in Figure 27 and Figure 29, but they would surely have to be implemented. Note that the aim of that table is only to provide a general view of what could be put on the spacecraft in terms of electrical power and mass, but, due to the encumbrances, it is not possible to fit all those devices in LuNaDrone. As an example, the only pressure regulator by Omnidea-RTG has physical dimensions equal to 158 x 140 x 80 mm^3, which are far more than it is permitted for a 12U spacecraft. This is to say once again that customization for the valves is the only way to make a feed system compatible with such low room. Moreover, not each component may be suitable for the type of fluid flowing into. As an example, latch valves are used both for propellant and gas and they will have to be designed to work with those fluids properly. Titanium valves are not suitable for hydrogen peroxide, while St Steel valves are, but St Steel valves could not be suitable for high pressure gas, and titanium could be preferable. Again, here it is really difficult to state, at this level of detail, how much the system will weigh, consume and its materials.

Another example is the latch valve for the main engine, which could be of many different types. MOOG proposes solenoid normally closed valves for different levels of thrust [45]. For a 40N monopropellant thruster a 230-grams normally closed valve with 26.5W of power consumption is presented, but without reference to the encumbrance.

One of the valves which deserve particular attention in the scheme is the one controlling the main engine. Here, various different solutions may be considered and, again, depend on other questions, like the possibility of throttling.

There are studies that show the possibility to design a flow control valve with a moving pintle that occludes a site positioned between the inlet and the outlet of the valve where the fluid flows. It could theoretically completely block the flow as well, if the travel of the pintle were long enough. So, one could think of utilising it both as a barrier and as a flow control, but there are various issues about that. Since low mass flow rate are required for the typical range of thrust of this mission (30-100N), it would be technically difficult to construct a tiny throat and a suitably small pintle to occlude it.

Another solution may be a cavitating venturi nozzle, which is a device that allows a liquid flowrate to be fixed or locked. This flowrate is not dependent on downstream process conditions or fluctuations. In function, this is similar to a sonic nozzle's velocity shockwave used with gases. A Sonic Nozzle's flowrate is adjusted with inlet pressure and is not sensitive to downstream conditions. The Cavitating Venturi, however, uses the liquid's vapor pressure point to limit or lock the flow. The throat of a Cavitating Venturi is sized such that the differential pressure generated from the inlet section to the throat reduces the liquid's absolute pressure to its vapor pressure point and it starts to vaporise or boil. These vapor bubbles begin to physically block the throat passageway. This prevents any additional increase in flowrate. If the inlet pressure is increased, this also raises the throat pressure, taking the liquid at the throat out of its vapor pressure point range. Additional flow may now pass through the Venturi which in-turn generates a higher differential pressure. This decreases the throat pressure to the vapor pressure point again and a new higher fixed flowrate is found.

Another possibility is to merge the pintle with the cavitating venturi effect [46]., to give the possibility of having a continuous throttling. The cavitating pintle acts as a cavitating venturi in order to choke the mass flow and make it independent of downstream pressure. The pintle changes the venturi throat area and thus varies the fluid mass flow without changing the upstream pressure.

Minimum mass flow range	30 $[g/s]$
Maximum mass flow range	300 $[g/s]$
Maximum operating pressure	80 $[bar]$
Venturi throat diameter	2.2 $[mm]$
Upstream throat radius	3.3 $[mm]$
Venturi divergence angle	10 $[deg]$
Pintle apex angle	10 $[deg]$
Maximum pintle stroke	11 $[mm]$
Useful pintle stroke	7 $[mm]$

Table 40 Variable area cavitating venturi properties

Looking at Table 40, it can be seen that a 2.2 mm throat diameter has been designed for higher flowrates than those hypothesised for this study. This means that eve lower throat diameter and thus thinner pintle would be required.

The simplest solution, despite less precise, could be the pulsed utilisation of the engine, with an only solenoid valve with a suitable response time. Currently, T4i's engine can be tested at 25Hz with 20ms of burn-state and 20 ms of shutoff-state. This would have to carefully analysed with the flight profile, in order to find the best solution. With properly varying the duty cycle of the engine, one could obtain the desired profiles of thrust, considering switch-on and switch-off transients.

Device	**PROs**	**CONs**
Pintle valve	Varying mass flow	Difficult to precisely move the pintle for tiny movements, bulky because of the motor for the pintle and the pintle itself
Cavitating venturi	Choking the mass flow, relatively simple and lightweight	One-lever control: upstream pressure
Pintle + cavitating venturi	Varying and choking mass flow with high precision with pintle insertion in venturi area	Complex and may not achieve the velocity of actuation required, cumbersome, need of a motor
Pulsed utilise (with latch valve)	Simple and lightweight, no motor required	No fine control on pressure or mass flow

Table 41 PROs and CONs of the different solutions

Table 41 PROs and CONs of the different solutionsTable 41 shows a summary of what being said above. Each of those solutions should be carefully analysed with the cases of both pressure regulated and blowdown system and may lead to changes in the presented fluidic schemes. In case of blowdown system, as an example, one should check that the pintle travel could guarantee the required mass flow rate and the correct pressure for the engine. What is important to say is the fact that each solution has strength and weaknesses and they have to be meticulously analysed in terms of performances, mass and encumbrances.

One may want to pay attention to thermal issues as well. Some of the valves in the fluidic line may have constraints in their operative temperature, possibly varying in the worst case from a few Celsius degrees above zero to a few tenths, looking at data sheets, because of sealings[4]. They can stiffen due to low temperatures or even crystallise. Usually, at least two levels of sealing are required, but three is preferable. In this study, no deepening on this aspect will be done and sealings will be considered already present in the valves.

[4] Often sealings are made of Viton (the same material for the bladder) or Silicon/ Teflon (check compatibility with H_2O_2)

For what concerns pipelines, steel 316L tubes can be utilised, because they are fully compatible with hydrogen peroxide and COTS. In addition, they are quite easy to shape, to find the better configuration inside the spacecraft. Just like valves, they can suffer from vibrations. As a first approximation, vibrations depend on the velocity of the fluid inside the tubes. In particular, one can say that the upper limit to avoid vibrations is 5 m/s for liquids and 10 m/s for gases (information coming from a private call). In this case, propellant mass flow rate is known and some evaluations may be conducted to choose the diameter of the tubes. It is useless to say that, at a given mass flow rate, the wider the diameter, the higher the mass and the volume of the pipe. This first approximation gives an idea of what type of tube would be better. Here, the comparison between 1/4" and 1/8" [47] is done.

OD_4	0,25	*inch*	0,635	*cm*
OD_8	0,125	*inch*	0,3175	*cm*
tube wall_4	0,035	*inch*	0,0889	*cm*
tube wall_8	0,028	*inch*	0,07112	*cm*
ID_4	0,18	*inch*	0,4572	*cm*
ID_8	0,097	*inch*	0,24638	*cm*
internal_area_4	0,025447	*sq. Inch*	0,164173	*cm^2*
internal_area_8	0,00739	*sq. Inch*	0,047676	*cm^2*
weight_4	0,08	*lb/ft*	0,119053	*kg/m*
weight_8	0,029	*lb/ft*	0,043157	*kg/m*
working_pressure_4	5100	*psig*	351	*bar*
working_pressure_8	8500	*psig*	586	*bar*

Table 42 Parameters of the tubes

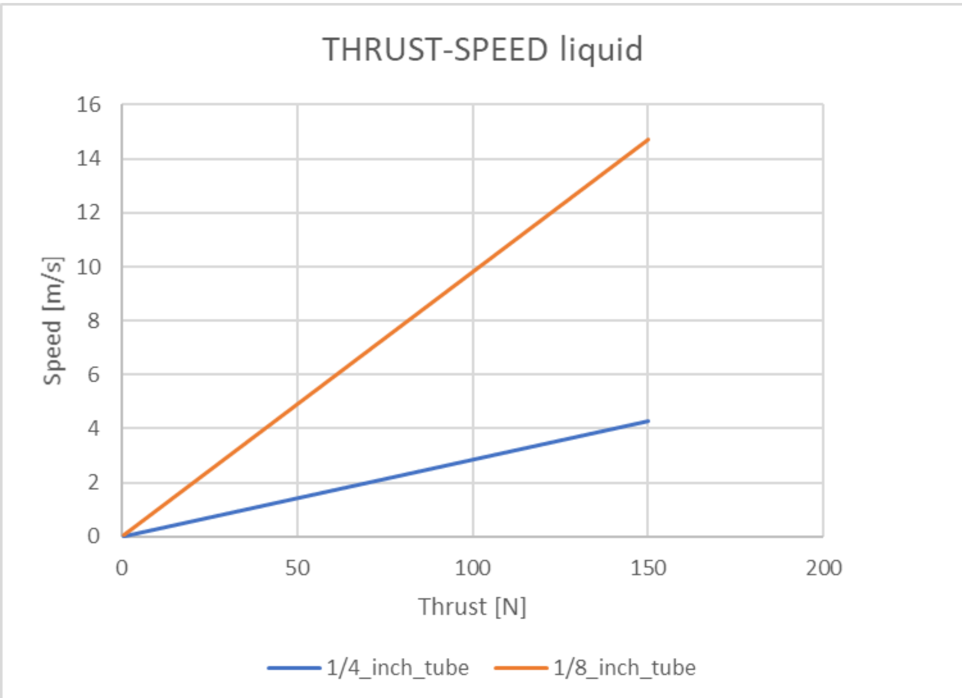


Figure 31 1/4" and 1/8" tubes comparison

Figure 31 shows that 1/4” tube would not have any problems for mass flow rates corresponding to a vast range of thrust. Instead, halving the diameter to 1/8” means to significantly reduce the internal area and thus, at given mass flow rate and density, to increase speed. At this purpose, the design of the whole fluidic will include 1/4" tubes except for the tubes linking the ACS thrusters, because they only need a really small mass flow rate. This is conservative in terms of mass and encumbrances and surely discourages distributed pressure loss along the pipes. No evaluations were done about the pressurant, because it is not immediate to know the required mass flow rate.

4.5.7 Pressure drops

The ideal case would be the one in which the propellant maintained constant its pressure between tank outlet and engine inlet, but, since it has to flow inside tubes and pass obstacles (filters, valves…) various dissipative effects will actually make it have losses. To evaluate how much pressure the propellant will lose, one can imagine differentiating the drops in distributed and concentrated. As the names suggest, distributed pressure drops will be considered to occur in a finite space (the length of a tube, for example), while the concentrated ones in an infinitesimal space (valves, filters, fittings…). This is acceptable as a first approximation and gives a first estimate about what the pressure must be inside the propellant tank to ensure a correct feed to the engine. After having a model of the propulsion system, one can evaluate the total pressure drop, which depends on various parameters concerning the devices, like tubes diameter and roughness, types of valves and concerning the type of propellant (density, viscosity, mass flow rate required etc…).
To have a rough idea of what the pressure drop could be, if no precise information can be found on the data sheet of the components, an online pressure drop calculator may be useful [48].

Calculation output

Flow medium:	Hydrogen Peroxide / liquid
Mass flow::	0.0329 kg/s
Weight density:	1405 kg/m³
Dynamic Viscosity:	1.245 cP
Element of pipe:	circular
Dimensions of element:	Diameter of pipe D: 0.18 in. Length of pipe L: 1 m
Velocity of flow:	1.43 m/s
Reynolds number:	7359
Velocity of flow 2:	-
Reynolds number 2:	-
Flow:	turbulent
Absolute roughness:	0.05 mm
Pipe friction number:	0.05
Resistance coefficient:	9.96
Resist.coeff.branching pipe:	-
Press.drop branch.pipe:	-
Pressure drop:	142.36 mbar 0.14 bar

Figure 32 Example of pressure drop along pipes

Figure 32 shows the expected pressure drop per metre of pipe. IN this case the pressure drop is moderate, because the internal diameter of the pipe and the mass flow rate[5] are not too low.

Pressure drops across valves are strongly dependent on the mass flow rate and the type of actuation the utilise. Usually, manufacturers test their products and generate graphs or coefficients to determine the pressure drop to be expected. Sometimes it is not easy to find them, so here again the online calculator may be useful.

[5] Calculated with 50N thrust and 155 s of specific impulse

Calculation output

Flow medium:	Hydrogen Peroxide / liquid
Mass flow::	0.0329 kg/s
Weight density:	1405 kg/m³
Dynamic Viscosity:	1.245 cP
Element of pipe:	Globe valve
Dimensions of element:	Diameter of pipe D: 0.18 in.
Velocity of flow:	1.43 m/s
Reynolds number:	7359
Velocity of flow 2:	-
Reynolds number 2:	-
Flow:	turbulent
Absolute roughness:	
Pipe friction number:	
Resistance coefficient:	26.43
Resist.coeff.branching pipe:	-
Press.drop branch.pipe:	-
Pressure drop:	377.67 mbar 0.38 bar

Figure 33 Example of a pressure drop across a globe valve

Figure 33 refers to a globe valve, which is the one with the greatest pressure drop in that online calculator. On the data sheets one, may find data about C_d (discharge coefficient), C_v (flow coefficient) or even Lohms[6], which are all numbers referring to the link between geometrical dimensions, mass flow rate and type of fluid in valves. In this case, since neither a precise fluidic scheme nor neither the type of feed system have been defined yet, only a rough preliminary analysis will be carried out.

Here below both regulated pressure and blowdown systems will be considered, focusing only on the propellant line and in particular the section linking the propellant tank to the engines.

Looking at Figure 27 for the pressure regulated system, only two latch valves are present, but, as written in Section 4.5.3, a filter could be added and another valve. Moreover, the catalytic bed has to be taken into account as well as the pipelines. Bends, fittings and other devices may be worth considering, because a single line will be split into others to feed ACS thrusters.

Figure 29 for the blowdown system is not so far from the other one, because it includes the above-mentioned filter and another valve, so there is no need to split the analysis into two different cases.

	p_drop	**p[bar]**
Chamber pressure		12
catalyst	20%	15
injector	2 bar	17
3 valves	1 bar each	20
filter	2 bar	22
bends + fittings	1 bar	23
tubes	0.5 bar	23,5
tank outlet	1 bar	24,5

Table 43 Example of possible pressure drops

[6] The Lee Company has developed the Lohm system for defining and measuring resistance to fluid flow. Just as the "ohm" defines electrical resistance, the "Lohm", or "liquid ohm" can be used as a measure of fluid resistance. The Lohm is defined such that 1 Lohm will flow 100 gallons per minute of water with a pressure drop of 25 psi at a temperature of 80°F [52].

Now, from Table 43 the resulting pressure that should be maintained inside the propellant tank is 24.5 bar, but in Section 4.5 the reference regulation pressure was 20 bar. Moreover, the chamber pressure may be risen to 15 bar depending on a more detailed trade-off. Increasing the pressure in the decomposition chamber could enhance performances, but increase mass due to the thicker walls of the engine. The estimate of Table 43 is much conservative and it would be probable and desirable to have around 20 bar in the propellant tank for the actual configuration. However, in the Annexes there are graphs which describe what would change with other pressures.

4.5.8 ACS thrusters

For what concerns ACS thrusters, no deep analysis have been done. The actual configuration includes 8 hydrogen peroxide monopropellant rockets of 1 Newton of thrust, with about 2.5 cm of length and a few grams of mass. Each of them is preceded by a LEE IEP solenoid valve [49] of 4.7 grams of mass, about 0.5 ms of response time and 4100 Lohm rate. Thanks to the latter it is possible to know the predicted pressure drop, following the instruction on The Lee Company's website [50].

Thrust [N]	mass_flow [kg/s]	vol_flow [L/min]	vol_flow [GPM]	DELTAP [PSI]	DELTAP[BAR]
0	0	0	0	0	0
0,25	0,000164481	0,007024111	0,00185558	0,193506638	0,013341818
0,5	0,000328963	0,014048221	0,003711161	0,774026552	0,053367273
0,75	0,000493444	0,021072332	0,005566741	1,741559741	0,120076364
1	0,000657925	0,028096443	0,007422322	3,096106207	0,213469092
1,25	0,000822406	0,035120554	0,009277902	4,837665948	0,333545457
1,5	0,000986888	0,042144664	0,011133482	6,966238965	0,480305458
1,75	0,001151369	0,049168775	0,012989063	9,481825258	0,653749095
2	0,00131585	0,056192886	0,014844643	12,38442483	0,853876369
2,25	0,001480331	0,063216996	0,016700224	15,67403767	1,08068728
2,5	0,001644813	0,070241107	0,018555804	19,35066379	1,334181827
2,75	0,001809294	0,077265218	0,020411384	23,41430319	1,61436001
3	0,001973775	0,084289328	0,022266965	27,86495586	1,921221831
3,25	0,002138256	0,091313439	0,024122545	32,70262181	2,254767287
3,5	0,002302738	0,09833755	0,025978126	37,92730103	2,614996381
3,75	0,002467219	0,105361661	0,027833706	43,53899353	3,00190911
4	0,0026317	0,112385771	0,029689286	49,53769931	3,415505477
4,25	0,002796181	0,119409882	0,031544867	55,92341836	3,85578548
4,5	0,002960663	0,126433993	0,033400447	62,69615068	4,322749119
4,75	0,003125144	0,133458103	0,035256028	69,85589629	4,816396395
5	0,003289625	0,140482214	0,037111608	77,40265517	5,336727307

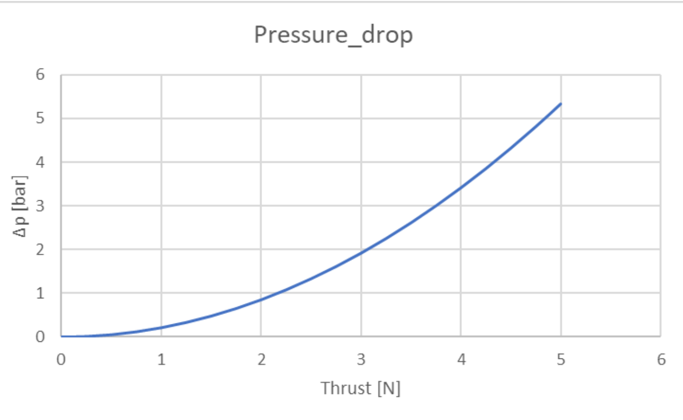


Table 44 Pressure drop of IEP series solenoid valves by The Lee Company

Constructing a monopropellant rocket for thrust minor than 1 N could be difficult (from a private call) and moreover, given the extremely low mass flow rates, the pressure drops would be too high, as shows Table 44, where a parabolic-like behaviour of the pressure drop as a function of the thrust (and thus the mass flow rate) can be seen.

Actually, since the number of ACS thrusters and their positioning strongly depend on many other parameters concerning other subsystems, it is not known if the solution of monopropellant rockets would be the most suitable. At this purpose, it would be possible to implement cold gas ACS thrusters, fed by nitrogen inside the pressurant tank or another, if present. In that case, without further details, the reference components are the following:

- Omnidea-RTG's 0.01-1 N thrust, MEOP 5.2 bar, 89 grams mass, St Steel/Ti6Al4V/Vespel material [51]
- MOOG's proportional flow control valve (PFCV)[7], MEOP 186 bar, 115 grams mass, throttle rate <25ms, St Steel and Vespel material, 1,000,000 cycles [52]

[7] This valve, on these conditions, would be surely oversized, but it is a good reference

4.5.9 Results

After collaborating with the other two colleagues, the current chosen configuration is a pressure regulated blowdown feed system, with one titanium tank for the pressurant at 350 bar and one Steel AISI316L for the propellant at 20 bar. The latter has been designed with a toroidal shape, because, after trying to insert the various components in the 12U spacecraft, the best possible choice was a toroidal shape with the engine inside the vacuum space into the minor diameter. After some calculations, a toroidal tank with circular section was not a possible solution, because to obtain the required internal volume the encumbrances would have been excessive. So, a toroidal tank with a rectangular section has been utilised as reference, despite it is not the actual configuration and it would not be suitable for managing the stresses due to the internal pressure (sharp corners). In fact, instead of a rectangular section, the following step was about the same section but with the angles fitted with 10 mm radius.

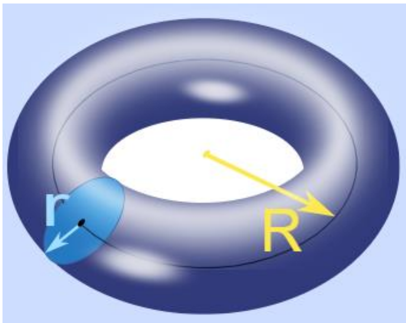


Figure 34 Example of geometrical parameter of a toroid

Figure 34 is not representative of the actual tank, but it is useful to understand how calculations were made. The internal volume of a toroid is given by

$$V_{int} = 2\pi RA$$

where R stands for the radius shown above and A for the area of the section.

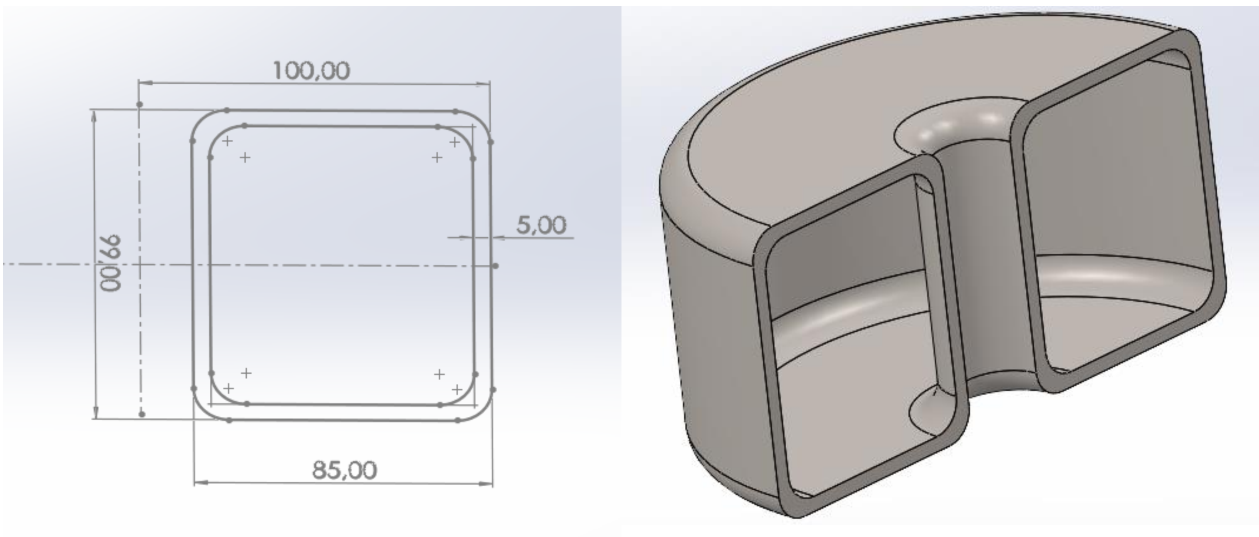


Figure 35 Current toroidal tank

Figure 35 shows the current configurations and all the measures necessary to explain the next steps.

This type of toroidal shape is already present in literature. The area of the section here can be written as follows:

$$A = \pi r^2 + 2(h - 2r)r + 2(b - r)r + (b - 2r)(h - 2r)$$

where r is the fitting radius, h the internal height of the section and b the internal base of the section in Figure 35. Comparing the volume required to the volume of this toroidal shape, one can see that the second one is bigger than the first one, so the required volume is available.

ext_diameter	200	*mm*	2	*dm*
ext_radius	100	*mm*	1	*dm*
int_diameter	30	*mm*	0,3	*dm*
int_radius	15	*mm*	0,15	*dm*
thickness	5	*mm*	0,05	*dm*
base_int	75	*mm*	0,75	*dm*
R	57,5	*mm*	0,575	*dm*
height_ext	99	*mm*	0,99	*dm*
height_int	89	*mm*	0,89	*dm*
Fiilet radius r	10	*mm*	0,1	*dm*
A	6589,159	*mm^2*	0,658916	*dm^2*
Volume	2380552	*mm^3*	2,380552	*litre*
Volume_int_required			2,372479	*litre*

Table 45 Actual toroidal tank - geometrical parameters

Note that the 5 mm thickness has been hypothesised. For a first verify of the admissible stresses with, one can refer to the following formulas [53]

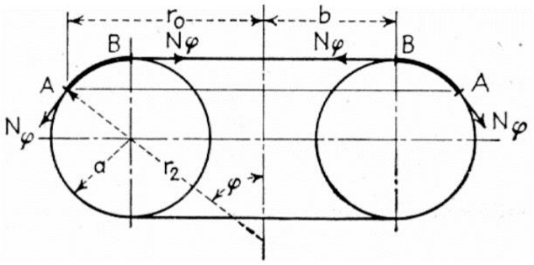


Figure 36 Circular toroid

$$\sigma_\varphi = \frac{pa}{2t}\frac{r_0 + b}{r_0} \qquad \sigma_\vartheta = \frac{pa}{2t}$$

These two formulas refer to Figure 36, a circular case. To adapt them for the current case, b will be equal to R and a will be the average between the internal height and the internal base of Figure 35. So, the actual case is compared to a toridal tank with circular section of diameter equal to that average.

Remembering that AISI316L has 170 MPa as yield strength, the results are as follows:

t	5	*mm*	0,005	*m*
k_safety	6			
p	20	*bar*	2000000	*Pa*
a	82	*mm*	0,082	*m*
b	102	*mm*	0,102	*m*
r0 min	20	*mm*	0,02	*m*
r0 max	184	*mm*	0,184	*m*
sigma r min	6,00E+08	*Pa*	600	*MPa*
sigma r max	1,53E+08	*Pa*	153	*MPa*

Table 46 Current tank evaluations

The stresses in this case (600 MPa) exceed the 170 MPa limit imposed by the material, but the safety factor has been risen to 6 take into consideration the impropriety of the formula. So, 5mm of thickness could probably be not enough with Steel, but they probably would with titanium, which in tun has issues of compatibility with hydrogen peroxide. Putting a safety factor of 3 instead of 6 would make the minimum thickness become 1 cm, which would lead to a high increase in mass and to a reduction of the internal available volume. So, to have the required internal volume, one should have to increase the height of the tank. The main problem here is the performance of Steel 316L: 170 MPa of yield strength and 8 kg/litre density are not possible solutions for this study.

Note that, for the sake of simplicity, accurate analysis has not been carried out, and the aim of this rough analysis is to give an overview of how the room inside the spacecraft could be utilised. Anyway, the final CAD file will include the tank shown in Figure 35, even though it is not to consider as a suitable choice. Moreover, the engine inside the tank may provoke thermal issues, that have not been analysed. In addition, the external diameter of the current tank is 20 cm, which is the maximum possible, because the dimensions of the 12U spacecraft are 20x20x30 cm.

If one had to modify some parameters (like pressure of regulation, type of gas ecc.), the result presented in the annexes for the spherical and pill-shaped tanks would be qualitatively correct yet.

For what concerns valves, the most critical components are the pressure regulator and the main engine valve. Here two options will be presented.

- OPTION 1: pressure regulator + simple latch main engine valve
- OPTION 2: no pressure regulator but latch valve for the pressurant + cavitating venturi valve

The cavitating venturi valve may be heavy and cumbersome as well as the pressure regulator, so, in terms of mass and encumbrances, the two options may be similar.

4.6 Final results and comments

Component	#	mass [kg]
propellant		3
prop tank	1	4,8
bladder		0,2
pressurant		0,074
press tank	1	0,156
tubes	0,12 kg/m	0,36
main engine	1	0,2
ACS thrusters + valve	8	0,8
check valve	1	0,04
press_regulator	1	1
latch valve	5	0,5
orifice	1	0,05
relief valve	2	0,2
fill-drai valve	3	0,3
total mass		11,68

Table 47 Gross estimate of the total mass of the propulsion system

Table 47 shows a first iteration estimate of the total mass of the propulsion system. This is composed by many items:

1) Mass of propellant: decided considering the flight profile
2) Mass of the propellant tank: is derived from the CAD file on Solidworks
3) bladder, pressurant and pressurant tank masses: evaluated in Section 4.5.2
4) tubes: 3 metres have been considered (in excess to take into account connections ecc)
5) mass of main engine: from CAD file on Solidworks
6) ACS thrusters + valve: average on datasheets in literature[8]
7) Check, latch, orifice, relief, fill-drain valves masses: average on datasheets in literature
8) Pressure regulator mass: literature

The mass of the pressure regulator may be too high to this purpose, but it can be intended in another way: let us suppose to have a 1 kg pressure regulator. If it really weighted 1 kg then it would have dimensions not compatible with the available room on the spacecraft. So, the estimate in Table 47 is to be intended considerably excessive.
It is important to note that, from the calculations coming from the flight profile, a 50N of thrust has been chosen as a good reference [13]. Despite that, a 75N engine has been designed and put into the CAD file. This is because, at this low level of detail, there is no certainty that a 50N engine could actually provide 50N of continuous thrust. So, with a margin of 50%, with a 75 N engine one can say that at least 50N of continuous thrust are surely available. Moreover, even though a 75N thrust were entirely available, a pulsed utilise,

[8] Omnidea-RTG's cold gas + solenoid valve has a mass of 89 grams each. Other manufacturers provide 1N monopropellant rocket with valve at about 200 grams. The actual configuration of LuNaDrone includes micro-thrusters of about 10 grams in excess with 4.7 grams solenoid LEE IEP valves.

properly adjusting the duty cycle, could provide a wide range of level of thrust below 75N. In other terms, 75N engine assures a 50N thrust and is more cumbersome than a 50N one, so this choice is merely due to conservativity.

Here below, the actual configuration of the propulsion system in the case of pressure regulated system is presented as a CAD file, but, for sake of simplicity, only a few valves are presented[9].

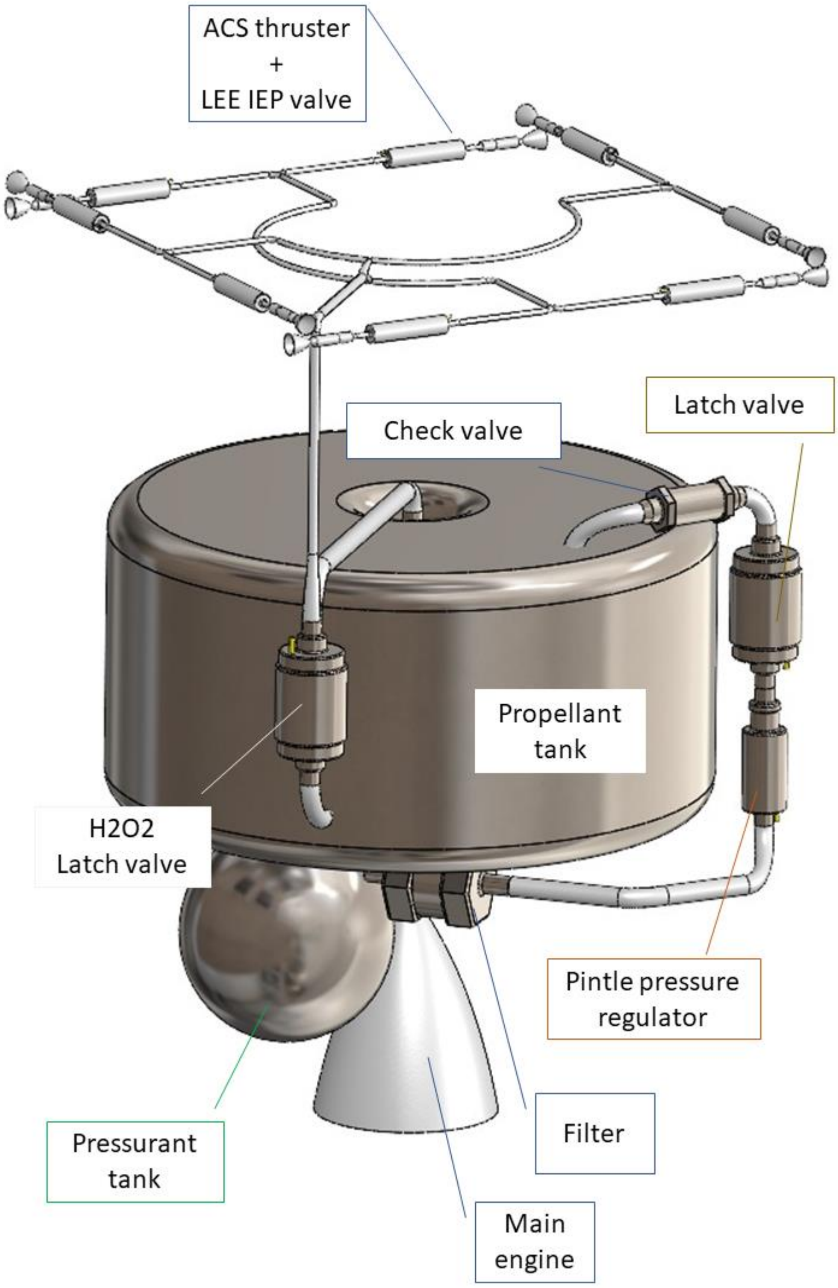


Figure 37 Simplified propulsion system

[9] Gentle courtesy by Gael Latiro [14], who has taken care of assembling the spacecraft after brainstorming with the author of this thesis

In Figure 37 a simplified view of the propulsion system is presented. Substantially, there is only a direct line from the pressurant tank to the engines; no fill-drain and other secondary devices useful for the generation of thrust are present. ACS thrusters and LEE IEP valves are in real dimensions, as well as the two tanks and the main engine.
On the other hand, the main engine valve is drawn as a normal latch valve and the pressure regulator as a pintle valve operating with gases, called “pintle microvalve” by the designers [54], that unfortunately is devoid of the range of operating pressure.

The total mass of the devices in Figure 37 is about 5,4 kg, without considering pressurant gas and propellant and using properly Titanium, Inconel and Steel. With pressurant and propellant the total mass would rise to about 8,5 kg, that seems to be a value blow which it will be really difficult to reduce, considering that it does not include many necessary devices.

It is fundamental to say that the representation of Figure 37 does not claim to be an actually working system, which follows ECSS’s rules and easy to test. It could indeed neither be possible for thermal issues and for structural requirements, because tanks, pipelines and engines have necessarily to be anchored somewhere. To conclude, this is only a rough idea and numerous deeper analysis have to be carried out.

Here below, some other ideas, possibilities and issues will be presented.

- **Engine**

Configuration	**PROs**	**CONs**
One single engine	Simple, reliability of a single engine and its valve, low horizontal encumbrance (one single nozzle)	Its length may exclude other internal configurations for the other devices
3-4 little separated engines nadir pointing	Relatively simple, shorter engines so less vertical encumbrance	Reliability linked to 3-4- engines and their valve. If one of these engines fails, flight becomes really difficult, more horizontal encumbrance (various nozzles), increase in mass
3-4 little engines with the same decomposition chamber nadir pointing (see Annex II - Engine)	Relatively simple, very low vertical encumbrance[10], one single valve, slight increase in mass	High horizontal encumbrance, reliability linked to the difference in the functioning of the 3-4 nozzles
3 separated engines, 1 big nadir pointing and 2 smaller ram and anti-ram pointing	Simple, horizontal movements quicker and more efficient due to the higher available horizontal thrust	Reliability linked to 3 engines and their valves, undesirable encumbrance, difficulties in the fluidic line
4 medium thrust engines, 1 nadir pointing, 2 ram and anti-ram pointing and 1 azimuth pointing	Better flight control due to the 4 directions of the thrust	The length of each engine obstacle the other devices, reliability

Figure 38 shows what is intended for nadir and ram directions, for the main engine/s discussion.
In general, increasing the number of engines, at given total thrust, increases the total mass and the total volume. Moreover, the more the engines, the less the reliability, because, in a series model, the total reliability is given by the product of each one of the single engine. So, the convenience in designing a higher number of engines should be found in the improvement of the flight performances, in terms of readiness at changing directions and of increased endurance. Careful risk analysis would have to be carried out. Heaters for the catalyst may be considered.

[10] This configuration includes 3-4 nadir pointing engines with a common decomposition chamber at 90° from the axis of the nozzles

Throttling is surely possible, but it strongly depends on the actuation velocity of the valve connected to the engine. At least two different throttling strategies are suitable: a pulsed utilise with constant mass flow rate and a proper duty cycle or a continuous utilise with varying mass flow rate. The choice, if throttling is required, has to consider the power consumption of the valves, their encumbrance and their compatibility with the desired duty cycle.
However, heaters and valves must be taken into consideration for the power budget.

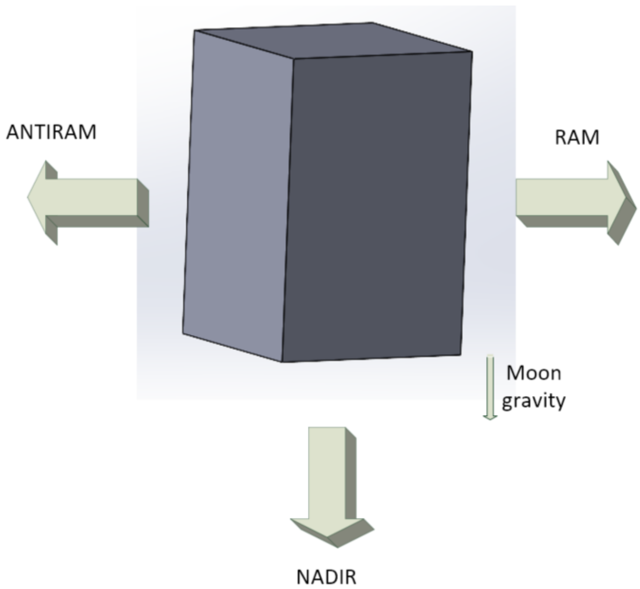


Figure 38 Auxiliary image of the spacecraft

- **ACS thrusters**

The only two realistic solutions are monopropellant and cold gas systems. The two types of engines are about the same in terms of mass and encumbrances, even because little and lightweight valves are suitable, as shown in this thesis. Moreover, if one chooses to utilise 1N thrusters, mass and volume are not a big problem. The main issue is indeed the line bringing them the propellant, because it is strictly affected by the positioning of the tanks. Moreover, the position of the thrusters itself is also affected by the positioning of the tanks, because they are the main responsible for the movement of the centre of mass.
For the actual configuration, in which the main engine is the only responsible for every movement and ACS thrusters only serves to correct the attitude and rotate the spacecraft for the various flight segments,8 1N thrusters positioned at the top of the spacecraft seems to be the best solution. Likely, the centre of mass should be in the lower parts of the spacecraft, because most of the devices are located there.
Other possible configurations could be 8 small thrusters at the bottom (modifying the management of the internal volume, because the fluidic line would obstacle the other devices this way) , or maybe 4 at the bottom and 4 at the top. One may want to notice that, in the last case a second latch valve would be necessary to shut the secondary line that another group of 4 engines would create.
All these above-mentioned ideas have necessarily to consider compatibility issues, because monopropellant engines would reach high temperatures and that may cause problems to the navigation systems or the electrical wires.
Cold gas system would not have thermal problems, but it would require a much higher mass of gaseous propellant, because of the very low specific impulse. This would lead to an increase of the pressurant tank mass, if one should decide to store the propellant together with the pressurant (and thus the two should be the same gas). At this purpose, helium would not be very suitable, because its density is extremely low,

despite its specific impulse higher than hydrogen peroxide). Another possible solution would be a separate tank for the gaseous propellant.
It would be easier to store a little more liquid propellant (hydrogen peroxide in this case) in the main tank for monopropellant rockets for ACS.
To conclude, one should estimate the amount of mass required for the ACS manoeuvres and then, after careful analyses, choose which solution could fit best.

- **Tanks**

The number of tanks strongly depends on the choice of the ACS thrusters and of the feed system. What concerns ACS thruster has been discussed before. For a pressure regulated system, splitting the main tank into others is not a good choice, because it would require more pressure regulators, despite the wall thickness would decrease for the minor diameter, at the same regulation pressure. Moreover, a more complicated fill system should be designed and many other valves would be necessary, likely.
A higher number of tanks does not seem a suitable solution.
The problem of the sloshing may be managed with different devices like bladders, diaphragms and others. Lightweight honeycomb structures may also be taken into consideration.

- **Thermal compatibility**

In the current configuration, the principal problem is created by the main engine positioned inside the internal diameter of the toroidal tank (see Figure 37). While pre-heating the catalyst would be a good idea for the readiness of the performances at the start of the engine, heating of the propellant tank could present a high risk situation, due to the high reactivity of the hydrogen peroxide. The range of admissible temperatures for the electrical components shall not be neglected.

- **Structural issues**

Usually, components of the propulsion system can not be utilised with structural function. The current configuration would need a structure where to anchor the ACS thrusters, the tanks and the engine. Deep analysis for this aspect is required.

Everything being said shall follow ECSS rules, obviously.

5 Conclusions and recommendations

This thesis started with the aim to present a high-level feasibility study of a particular mission on the Moon, whose purpose is to explore the inside of lunar caves. Rocket propulsion has been chosen to design a drone able to autonomously fly, eliminating the issues relative to the presence of obstacle on the lunar soil.

For what being said, a 12U non-orbiting autonomous drone capable of moving in an environment without atmosphere is made of propulsion system for most of its internal volume, if the desired flight time is various tenths of seconds. In fact, given the specific impulse of the engine, the necessary mass of propellant is fixed and it must be stored onboard.
The main issues are linked to the mass and volume of the different devices.

The current technology of the propulsion is mature for this type of missions. On the other hand, miniaturisation of valves is yet to be fully achieved and qualified. With the state-of-the-art COTS components, for what it is possible to find freely on the internet, this mission seems not to be feasible, but just because of the restrictive 12U constraint.

Future developments should aim to reduce, where possible, mass and volume of the different devices. As already said, the amount of propellant cannot be modified, once the flight profile is set. Valves and tanks are surely the components on which to focus. The two main aspects to be considered are:

- the design of new lightweight valves
- the development of lightweight materials with high yield strength for the tanks, compatible with hydrogen peroxide

The raising and continuous improvement in additive manufacturing could represent a strong positive factor.

These two improvements would bring many advantages to the whole space sector and not only that.

Annexes

Annex I - Chemistry

Here following there are the result of a computation about the chemistry for hydrogen peroxide using CproPep, freely available on the internet. Some of these will be taken as reference values.

Computing case 1
Frozen equilibrium performance evaluation

Propellant composition

Code	Name	mol	Mass (g)	Composition
1044	HYDROGEN PEROXIDE 90%	0.0003	1.0000	196H 179O
1044	HYDROGEN PEROXIDE 90%	0.0003	1.0000	196H 179O

Density : 13.992 g/cm^3
2 different elements
H O
Total mass: 2.000000 g
Enthalpy : -6573.28 kJ/kg

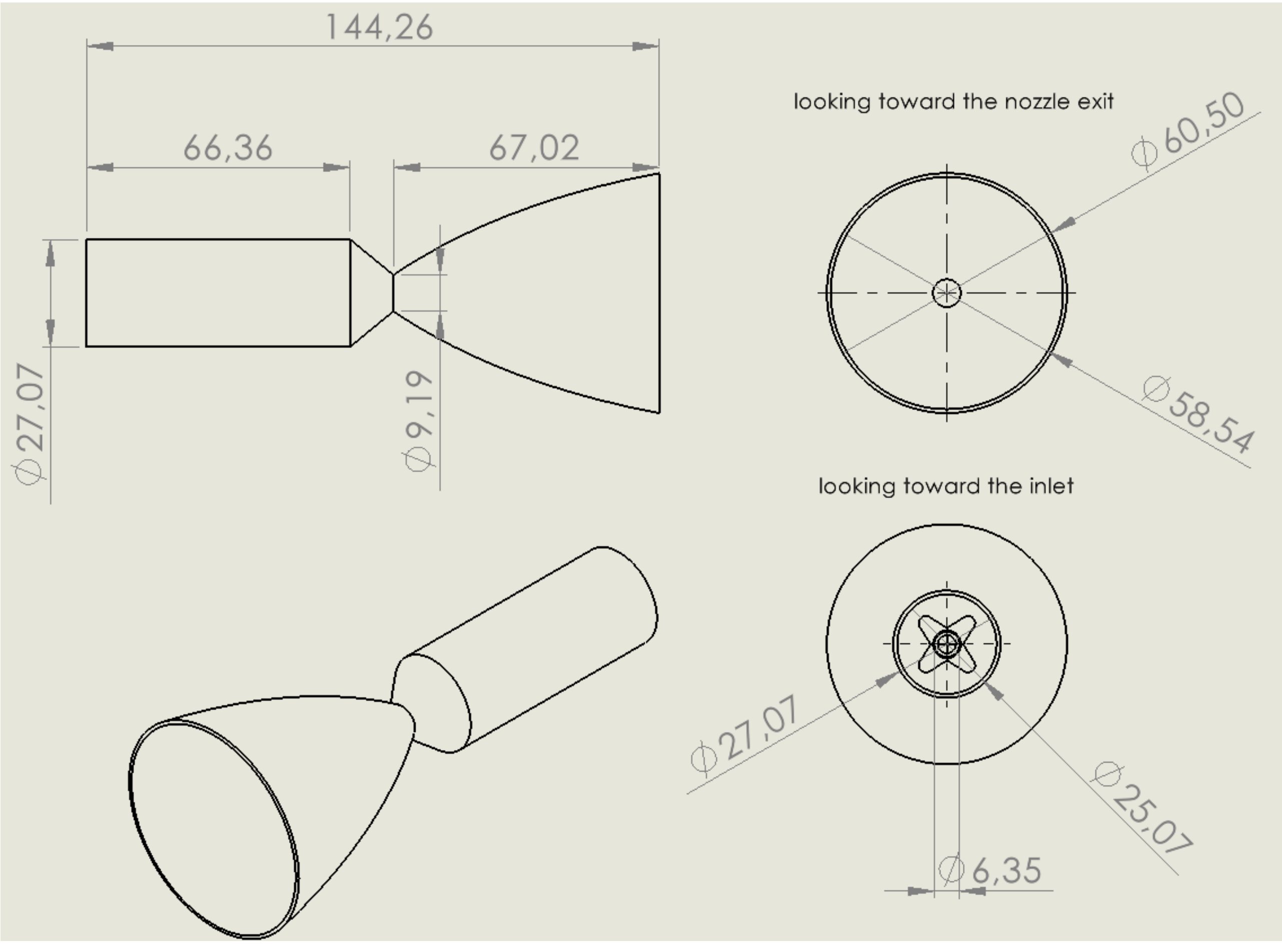
9 possible gazeous species
2 possible condensed species

	CHAMBER	THROAT	EXIT
Pressure (atm) :	9.869	5.425	0.006
Temperature (K) :	1019.590	897.350	165.374
H (kJ/kg) :	-6573.275	-6788.808	-7934.300
U (kJ/kg) :	-6956.793	-7126.345	-7996.505
G (kJ/kg) :	-16838.951	-15823.721	-9599.359
S (kJ/(kg)(K) :	10.068	10.068	10.068
M (g/mol) :	22.104	22.104	22.104
(dLnV/dLnP)t :	-1.00000	-1.00000	-1.00000
(dLnV/dLnT)p :	1.00000	1.00000	1.00000
Cp (kJ/(kg)(K)) :	1.79253	1.73365	1.44310
Cv (kJ/(kg)(K)) :	1.41638	1.35750	1.06696
Cp/Cv :	1.26557	1.27709	1.35254
Gamma :	1.26557	1.27709	1.35254
Vson (m/s) :	696.68176	656.55513	280.57995
Ae/At :		1.00000	69.99999
A/dotm (m/s/atm) :		94.76749	6633.72349
C* (m/s) :		935.28246	935.28246
Cf :		0.70199	1.76403
Ivac (m/s) :		1170.65819	1687.56651
Isp (m/s) :		656.55513	1649.86323
Isp/g (s) :		66.94999	168.23923

Molar fractions

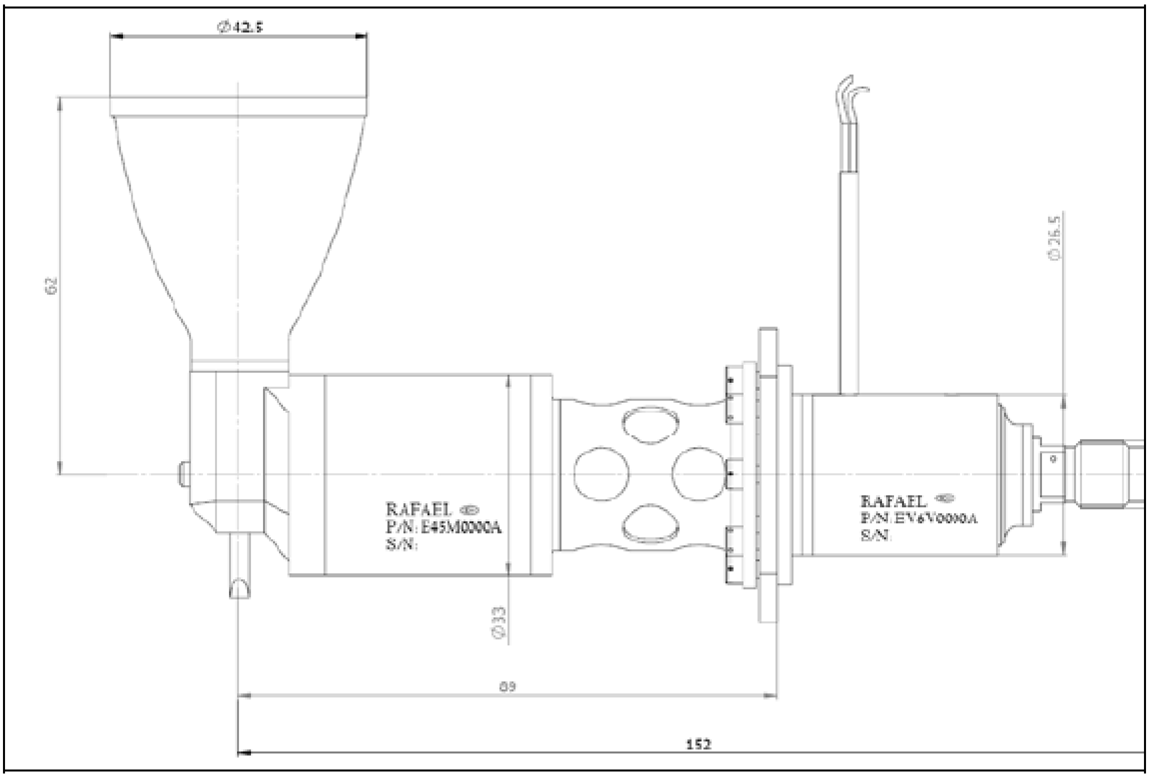
H2O	7.0758e-001	7.0758e-001	7.0758e-001
OH	2.7217e-007	2.7217e-007	2.7217e-007
O2	2.9242e-001	2.9242e-001	2.9242e-001

Annex II - Engine



Annex II - 1 Dimensions of the engine in mm

The dimensions shown in Annex II - 1 belong to the 75N engine in the current configuration of LuNaDrone. The inlet of the engine has been designed with a 1/4" pipe splitting into four, linked to the decomposition chamber. These dimensions are derived from a first iteration and have to be taken as an example and not as a final configuration.



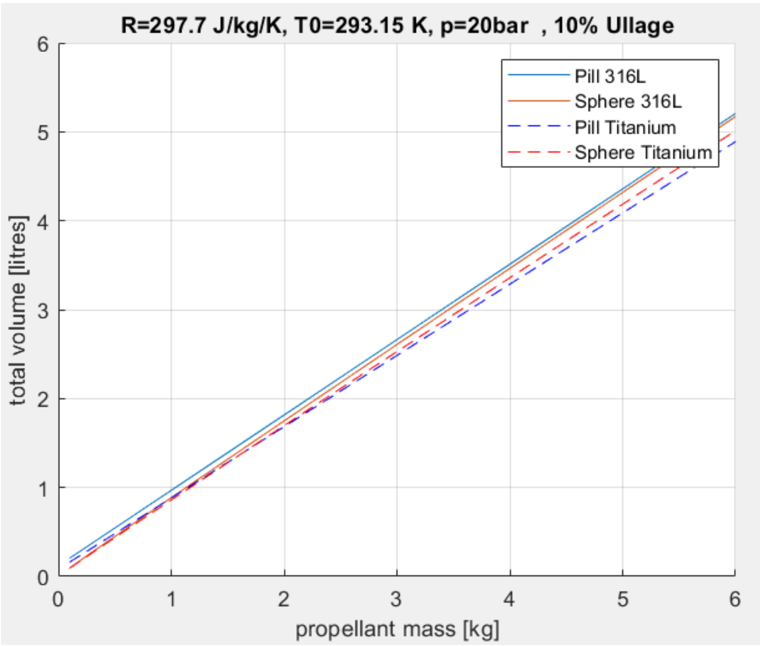
45 N Thruster

Annex II - 2 45N thruster by RAFAEL, with not coincident nozzle and combustion chamber axis

Annex III – Propellant tank

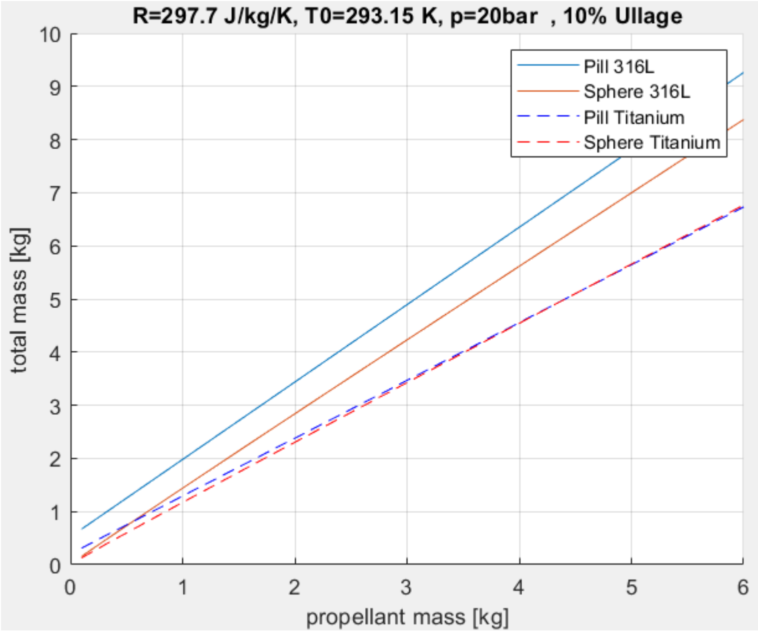
Here one can find other information about how different parameters affect the propellant tank, supposed to be spherical or pill-shaped and made of stainless steel AISI316L or titanium, in case of a regulated pressure system. Titanium Ti-6Al-4V is only implemented to have a comparison with steel, but it cannot be utilised alone, because of its incompatibility with hydrogen peroxide. One could say that, if it were possible to create a material with those performances but also compatible with hydrogen peroxide, the results would be similar to the following.

- **Variation of propellant mass onboard**



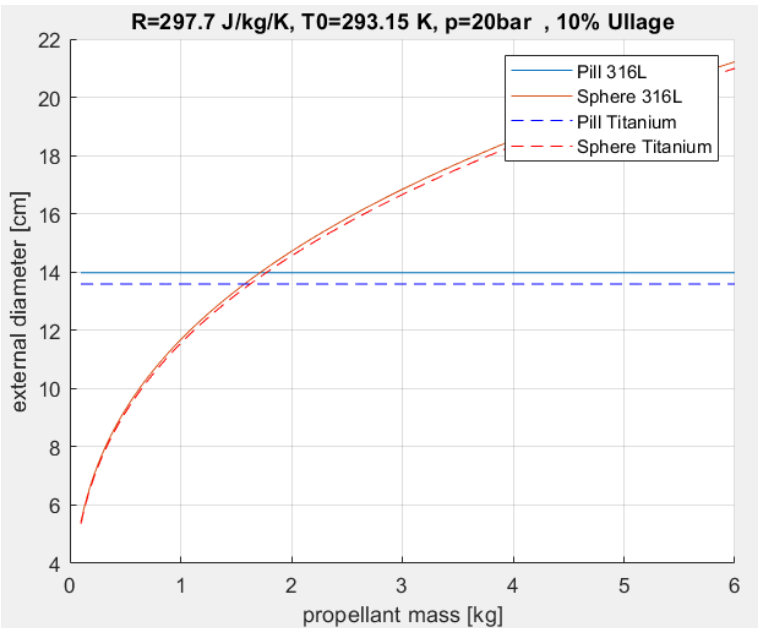
Annex III - 1 Total volume as a function of the propellant mass

Propellant mass and propellant volume are proportional and thus also the total volume of the tank is such. Since each kilogram of propellant occupies less than a litre, the resulting total volume is composed by metal, and ullage and it seems that for each kilogram of propellant less than a litre will be occupied. This is not the case for the total mass, as follows.



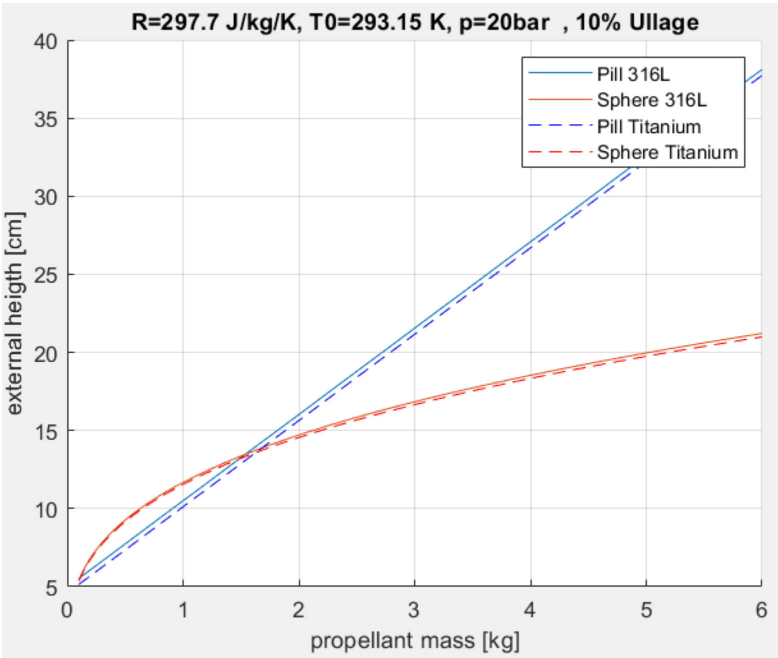
Annex III - 2 Total mass as a function of the propellant mass

The total mass is composed by propellant, ullage gas, bladder and metal. As it has been evaluated, the mass of the bladder is constant, if one does not change the diameters if the two tanks. The mass of gas depends on the ullage, which in turn depends on the volume of propellant and thus on the propellant mass, but it is the same both for the pill-shaped and the sphere. What makes the difference here is the mass of metal utilised, remembering that here the regulation pressure is assumed to be constant. As it can be clearly seen, titanium is significantly lighter than steel and results in almost equal masses in the two cases of sphere and pill. This is because of its high yield strength, which makes the wall thickness become very low and similar between the cases of sphere and pill. Then, since titanium has a density which is almost half than steel, the resulting mass is much lower. As said before, titanium cannot be utilised.



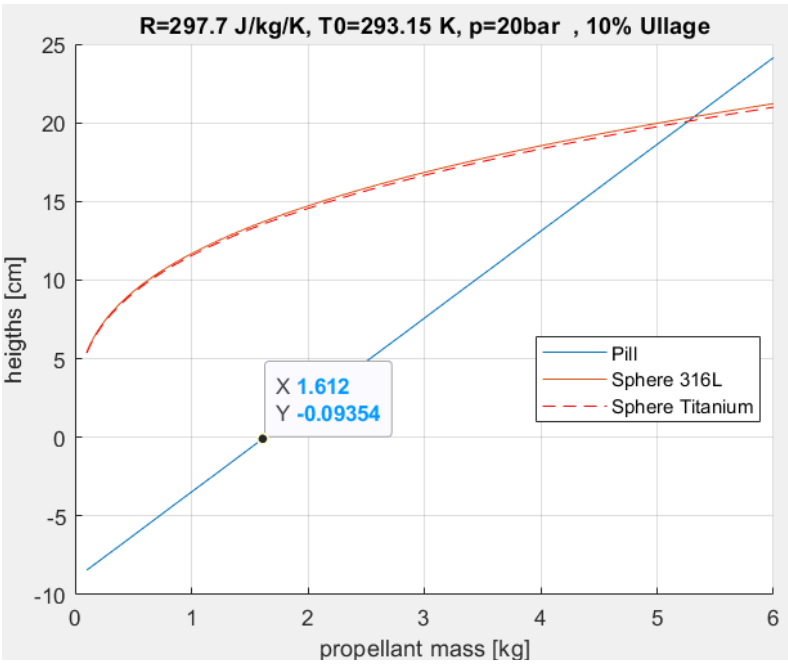
Annex III - 3 External diameter as a function of propellant mass

The external diameter, in case of pill-shaped, is constant, because it is the parameter one chooses as an input. The difference between steel and titanium is always the same, the different yield strength which leads to different wall thickness. The external diameter for the sphere rises with the propellant mass, because it is the only free parameter, defined by the propellant volume and thus mass. Remember that great diameters for the sphere could imply difficulties in the management of the room in the spacecraft.



Annex III - 4 External height as a function of the propellant mass

Note that, in this graph, the reasonable values for the pill-shaped are those for propellant masses higher than the point where pill and sphere encounter each other. The reason is shown in the next graph.

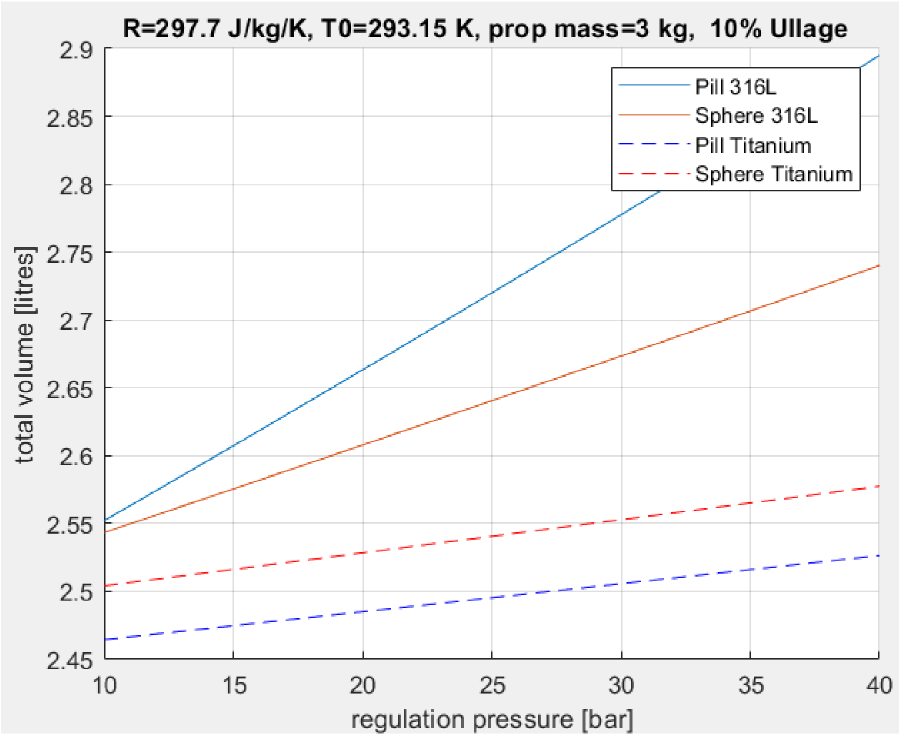


Annex III - 5 Height of the cylinder starting from negative values

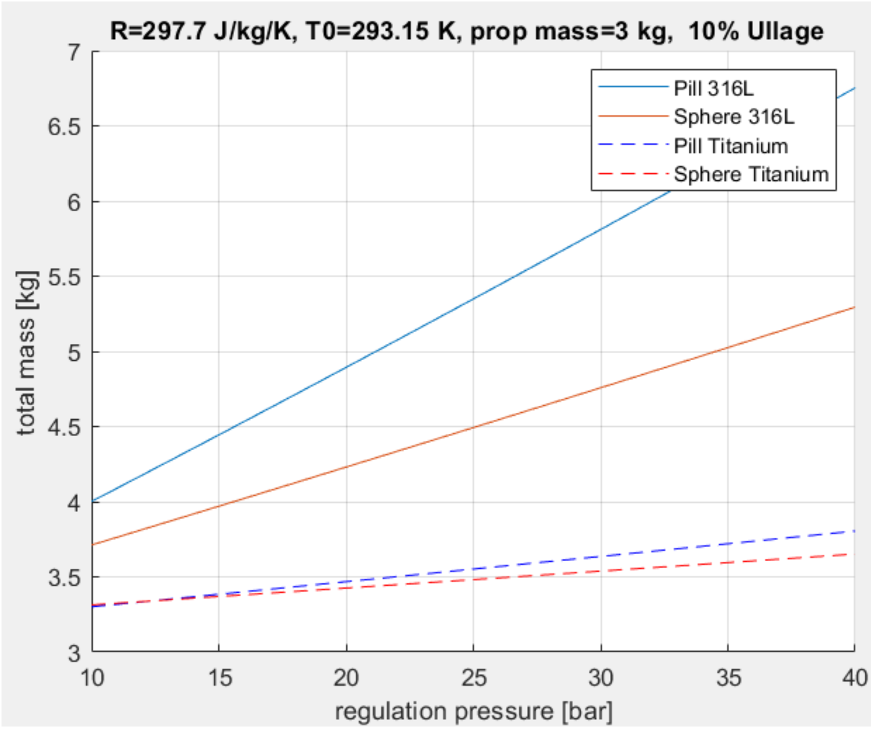
The only values for the only cylinder height (pill case, no matter the material, since the height of the cylinder is fixed after fixing the diameter) which have to be taken into account are the positive values. Negative values are an "error" of the calculations. Fixing the diameter equals to fix an internal volume. If the propellant mass requires a smaller internal volume, the script tries to equal those volumes, going towards negative value to subtract the fixed volume of the sphere. So, the reasonable values are only the ones after about 1.6 kg of propellant, in this case, as shown by the box in Annex III - 5. Varying the diameter of the cylinder will make the results change, but always according to this logic.
That being said, considering only the reasonable values, external height for the sphere is simply the diameter. External height for the pill is the height of the cylinder plus two halves an external diameter of the sphere (or cylinder, since they are the same, which is chosen as an input). Here it can be seen that the external height is linear, because the diameter of the cylinder is fixed and so the additional volume is only due to the height. So, if the volume required rises, so does the height, linearly. These calculations give a first idea of what the encumbrances of the tanks will be, in terms of height and diameter.

- **Variation of the regulation pressure**

Regulation pressure is an important parameter, which has to be designed in order to guarantee the correct pressure drop which the propellant will undergo before arriving to the decomposition chamber. The following graphs mean to give an idea of how much the result may vary, varying the pressure, fixing ullage and propellant mass.

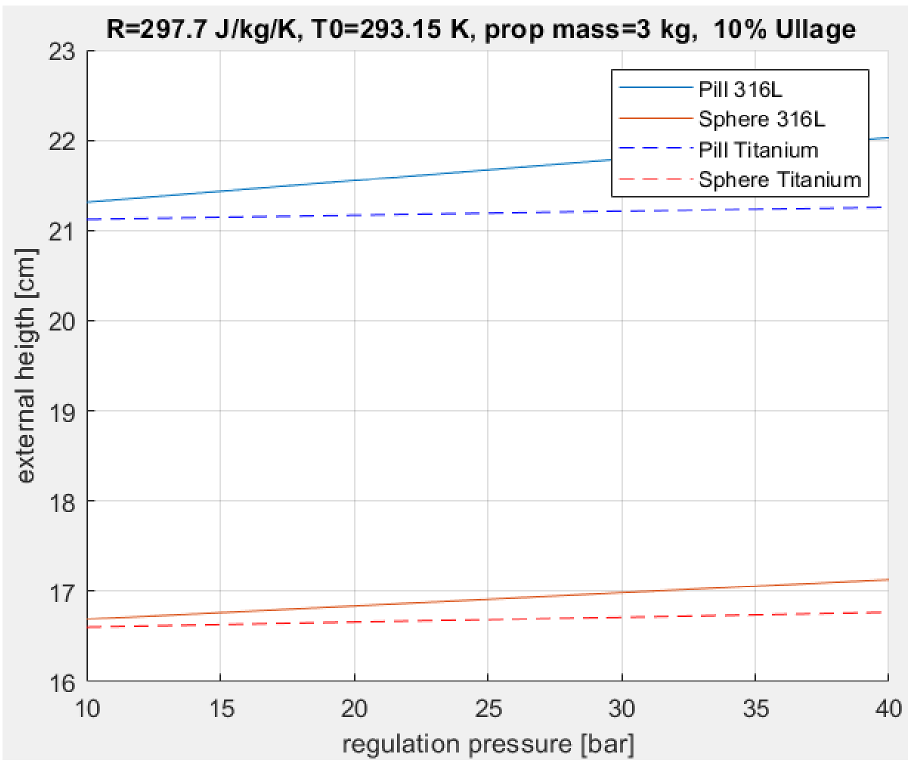


Annex III - 6 Total volume as a function of the regulation pressure

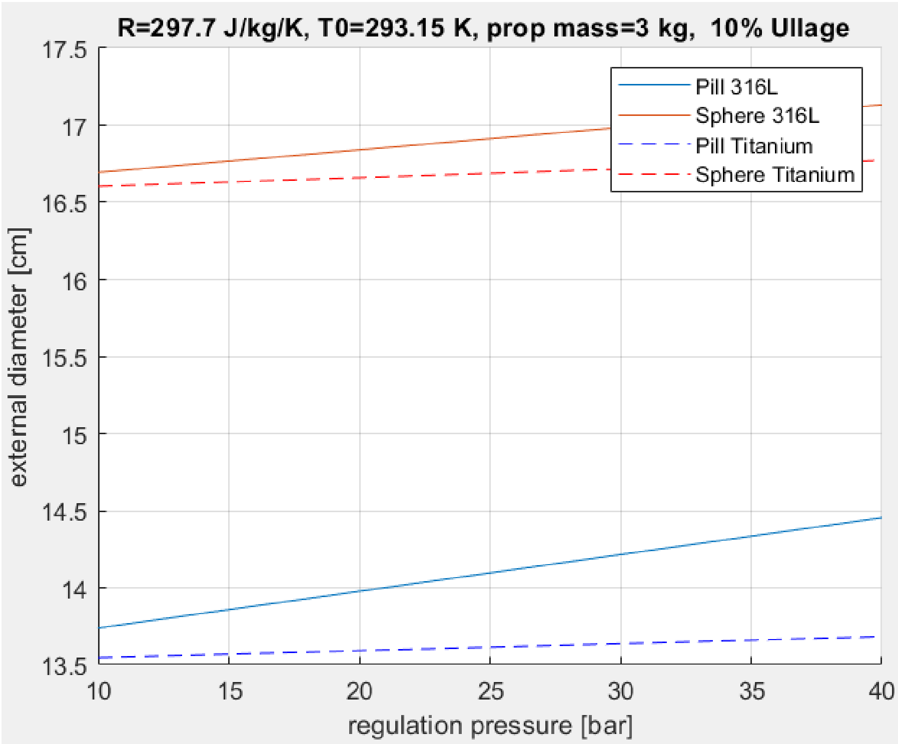


Annex III - 7 Total mass as a function of the regulation pressure

As it may be easy to predict, augmenting the pressure of regulation makes volumes and masses rise, again with differences between materials and shape of the tank. Remember that here, for the pill case, the diameter of the cylinder is fixed at 1.35 dm.



Annex III - 8 External height as a function of the pressure of regulation

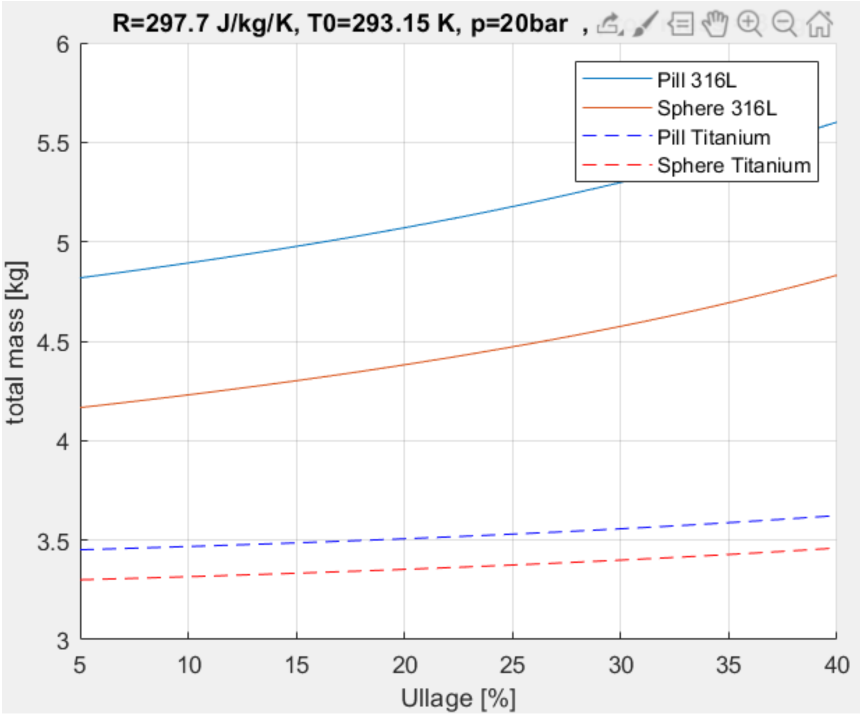


Annex III - 9 External diameter as a function of the pressure of regulation

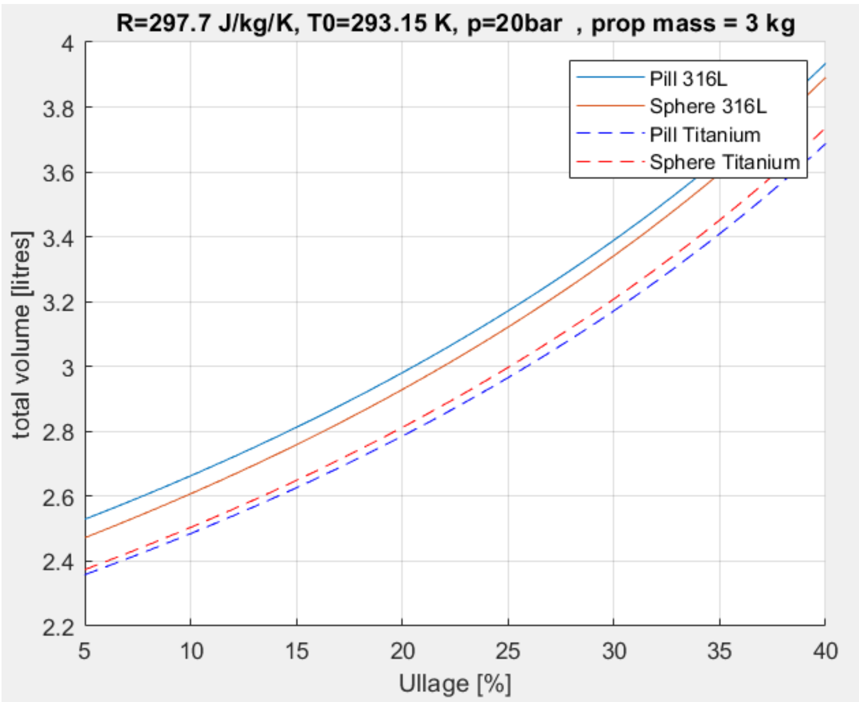
The problems reported in the example varying the propellant mass are not present here, because the volume of propellant is high enough not to make calculations find a negative height for the cylinder. However, the regulation pressure, making masses and volume rise, makes also geometrical parameters rise. In the last two graphs, the effect of the pressure is clearly visible with the increasing wall thickness, which substantially is the difference between a random value of the curve and the fixed internal diameters.

- **Variation of the ullage**

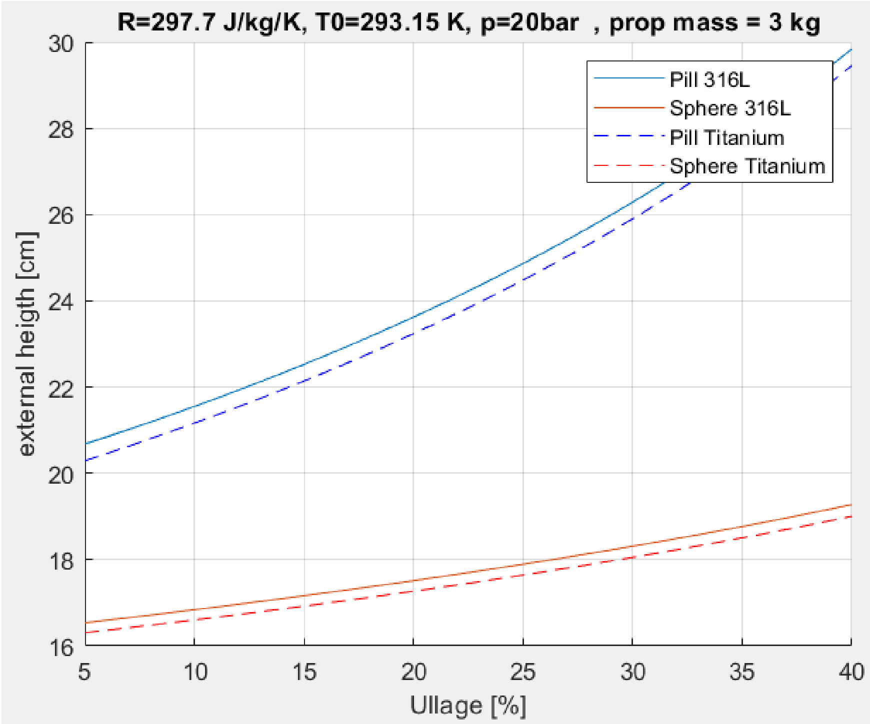
Ullage is necessary for the propellant tank, but the more the ullage, the more the volume, so the optimal value has to be found. In the previous graphs, an ullage of 10% had been utilised, because it is a common value, but the aim of this thesis does not include to find the best value, but only how the value can affect the system.



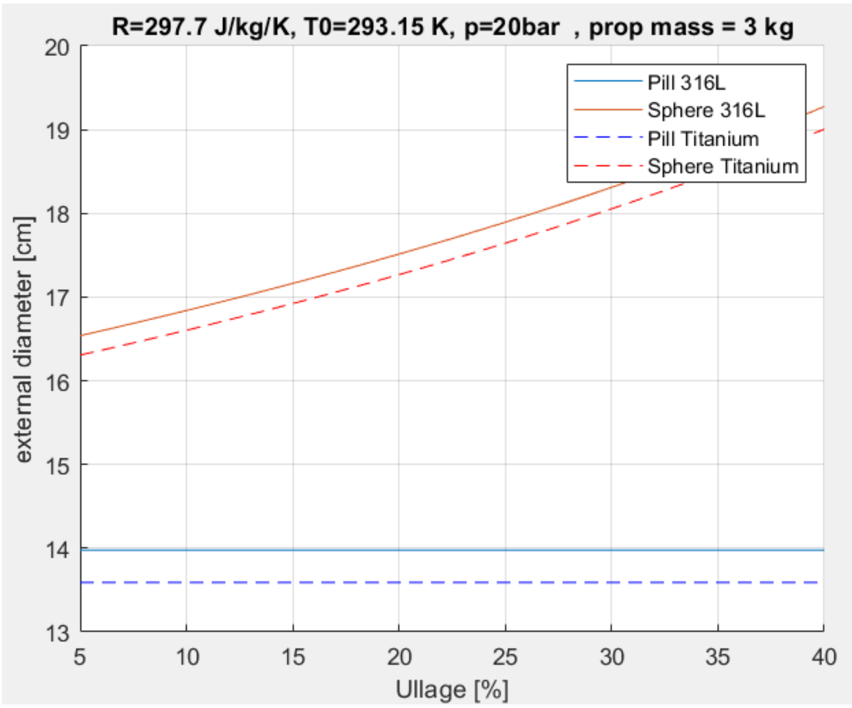
Annex III - 10 Total mass as a function of the ullage



Annex III - 11 Total volume as a function of the ullage



Annex III - 12 External height as a function of the ullage

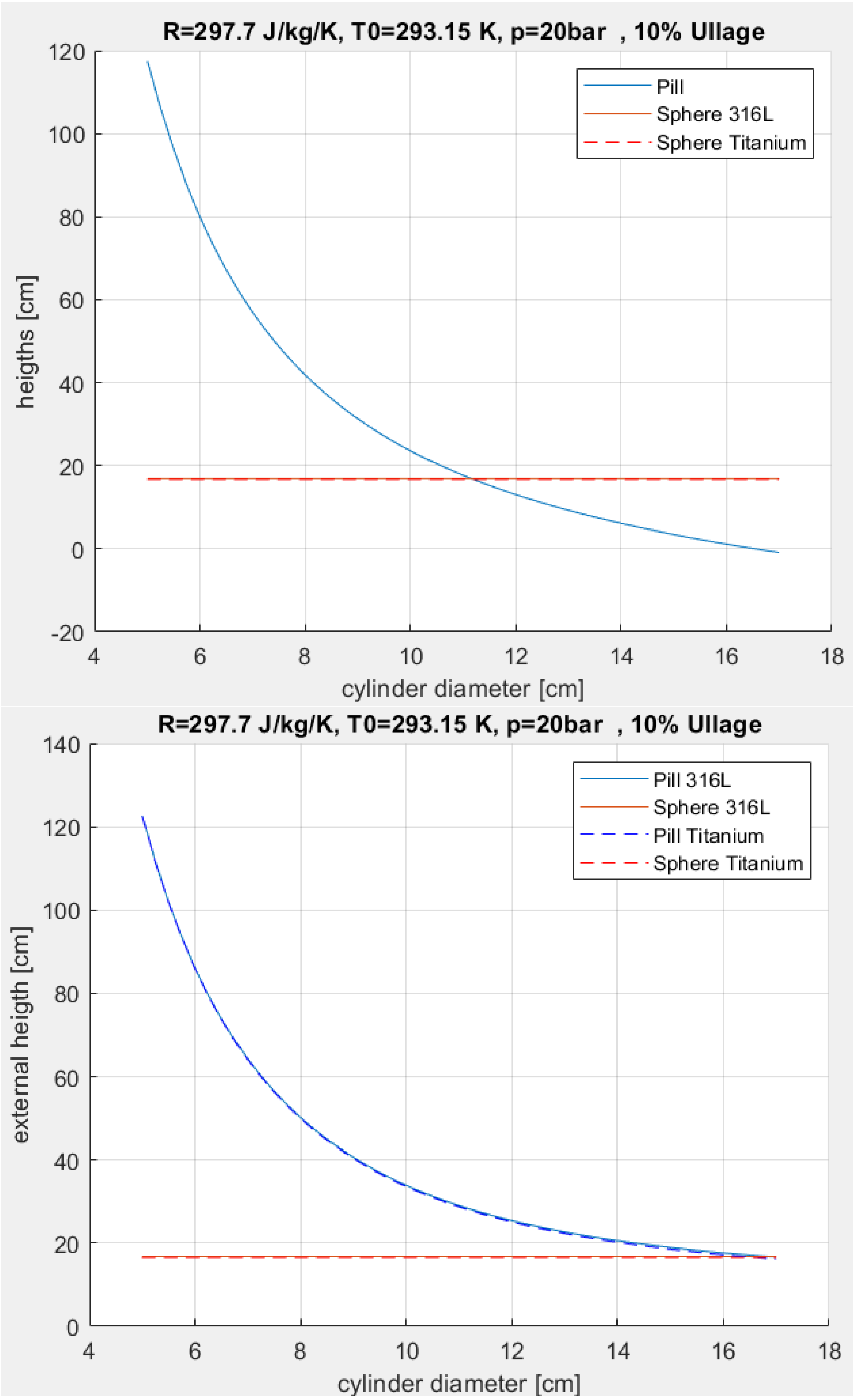


Annex III - 13 External diameter as a function of the ullage

As it can be seen, major ullage leads to major inert volume and mass, because the amount of propellant is always the same. It would make no sense to choose high values of ullage, because encumbrances would rise too much. Moreover, this effect with steel as material is even more underlined, both in terms of mass and volume.

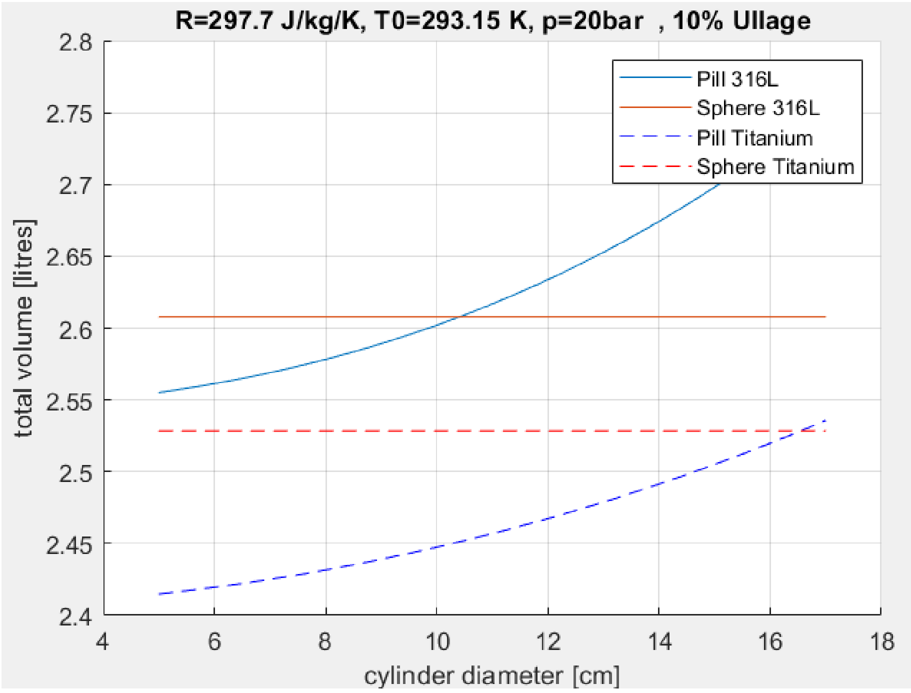
- **Variation of the diameter of the pill**

The diameter of the cylinder, at given mass of propellant, ullage and pressure of regulation, has to be decided principally for the encumbrances inside the spacecraft. Fixed the internal volume, the lower the cylinder diameter, the greater the height will be. This affects masses and volume, as shown here below. The parameters of the spherical tank will not vary, because the amount of propellant if sixed. Variations will appear only for the case of the pill-shaped tank.

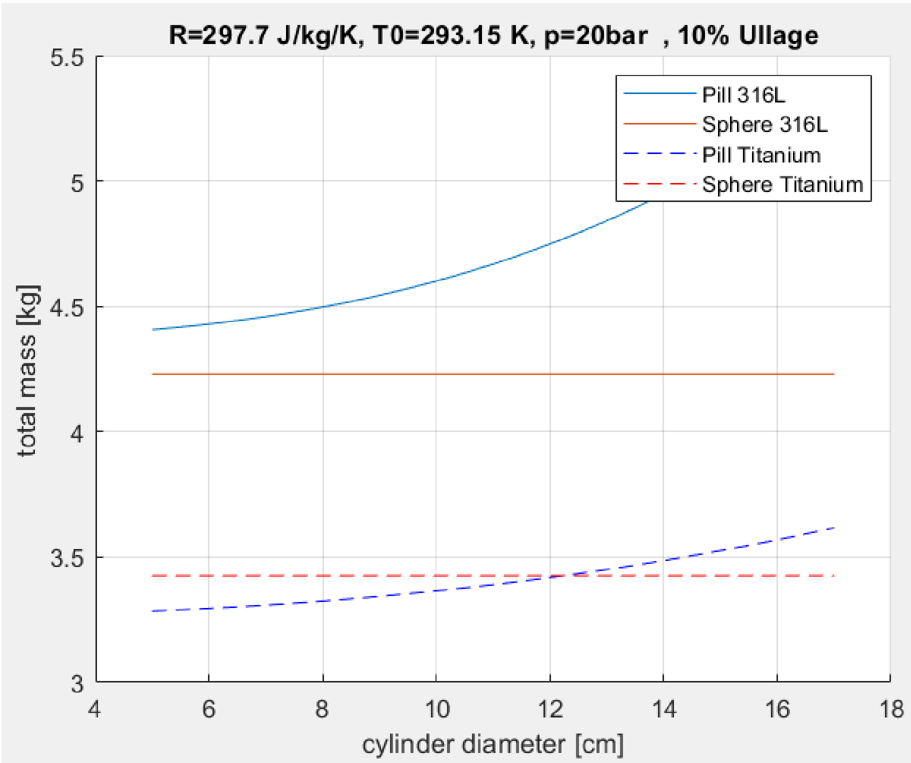


Annex III - 14 Heigth as a function of the diameter of the cylinder

Annex III - 14 shows again that the height of the cylinder may become negative, as already seen before. So, the only acceptable values are those before the cylinder diameter for which the resulting cylinder height is zero or, also the cylinder diameter equals the diameter of the sphere in the sphere case. In fact, it can be clearly seen that around values of 16.5 cm of diameter, the external height is about 16.5 cm, which means that the pill degenerate into a sphere.

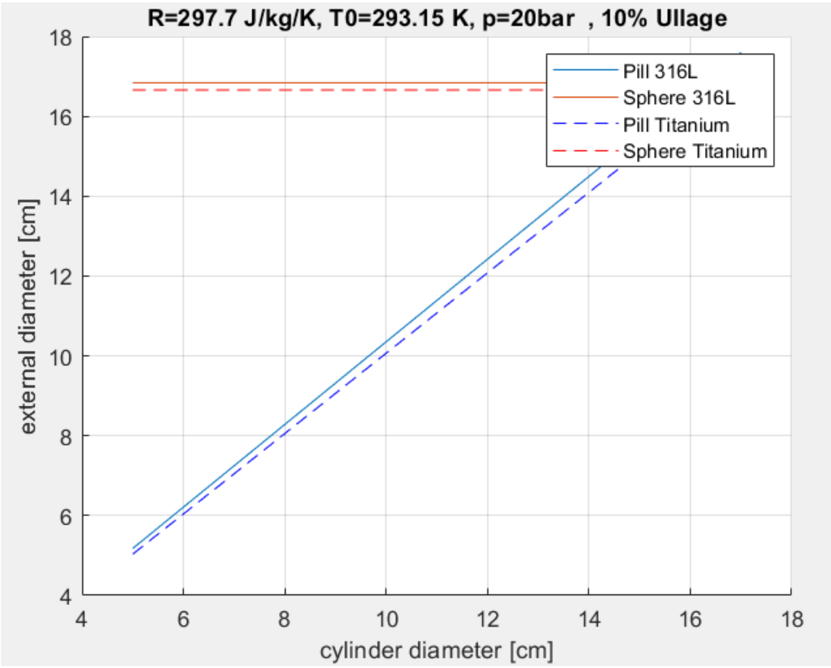


Annex III - 15 Total volume as a function of the diameter of the cylinder



Annex III - 16 Total mass as a function of the diameter of the cylinder

Looking at Annex III - 16 one may think that reducing the diameter of the cylinder would be a good idea, but, as an example, if one set 10 cm, then the external height (see Annex III - 14) would be more than 30 cm, which is the maximum dimension of the 12U spacecraft (if one decides the 20x20x30 cm configuration). So, the design of the tank, both being pill-shaped and spherical, must include careful analyses.
These comparisons may be carried out also varying the mass of propellant or other parameters, but the results would be analogous.

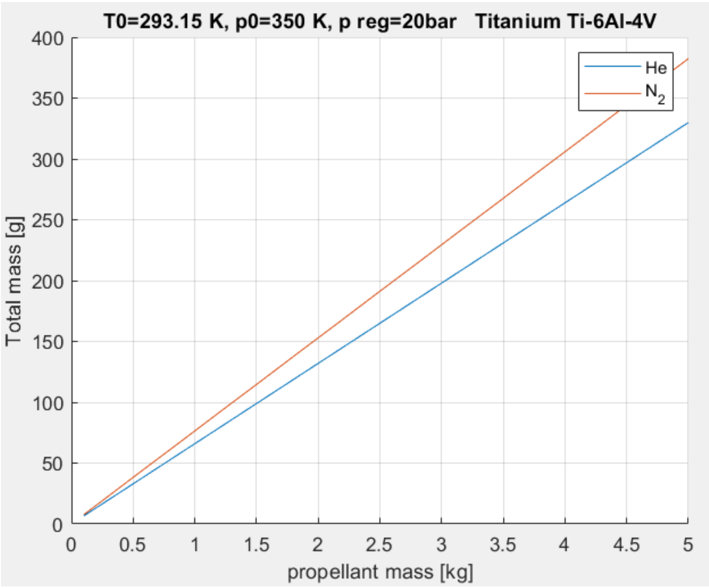


Increasing the diameter of the cylinder will increase the stresses ant thus the wall thickness.

Annex IV – Pressurant Tank

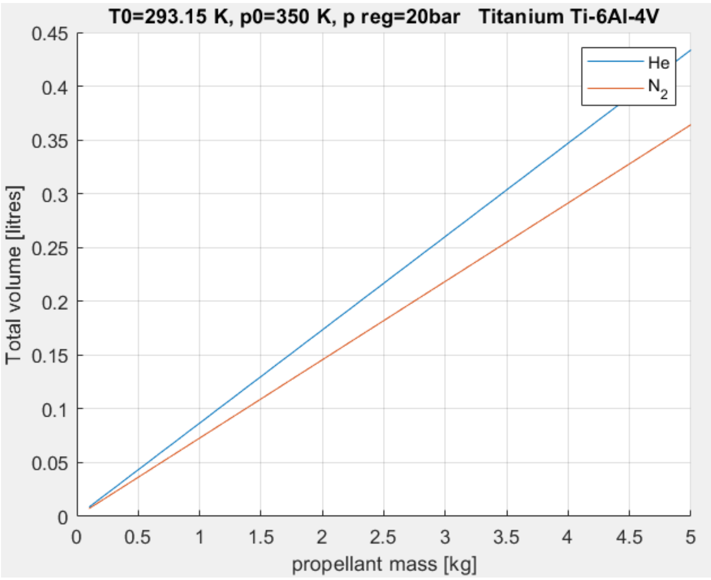
Here one can find other information about how different parameters affect the pressurant tank, supposed to be spherical and made of titanium. Titanium Ti-6Al-4V is much lighter than St Steel AISI 316L and has noticeably higher yield stress, so, since there are not problems of compatibility here with gases, it is more suitable than steel. Note that helium is a tiny molecule, so, despite it has good performances, it has also bad sides. In fact, Helium tends to escape from the fluidic line through very little possible openings. At this purpose, if one decided to use Helium, he should consider welded connections rather than threadings.

- **Variation of propellant mass onboard**

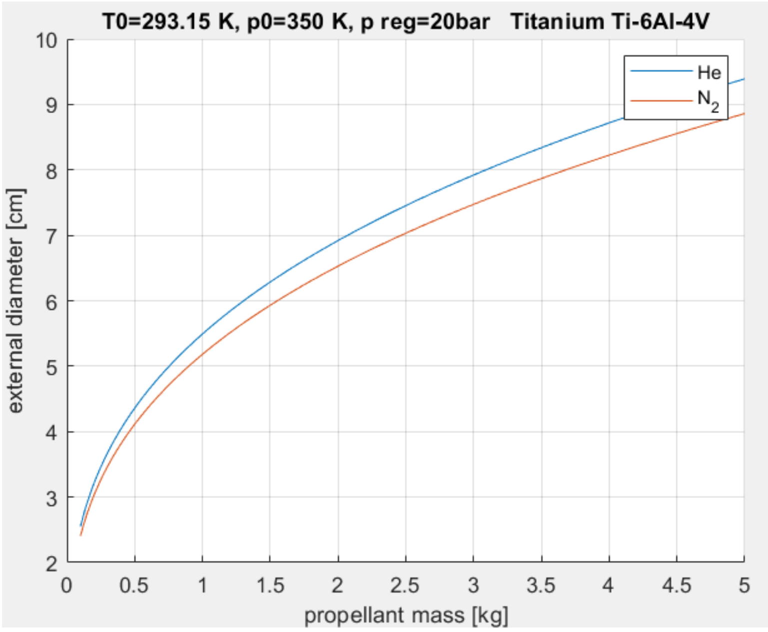


Annex IV - 1 Total mass as a function of the propellnt mass

As shown here above, using helium is a better solution with respect to nitrogen. Obviously, the more propellant is boarded the more mass the pressurant tank will have.



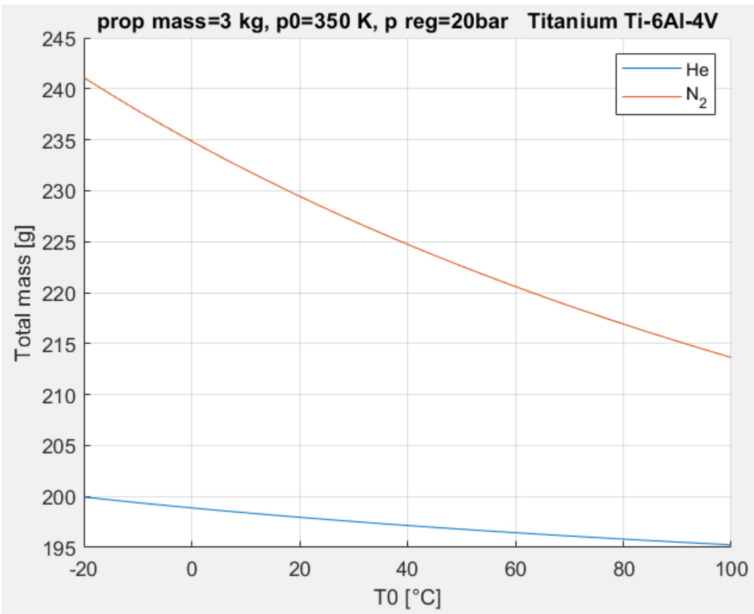
Annex IV - 2 Total volume as a function of the propellant mass



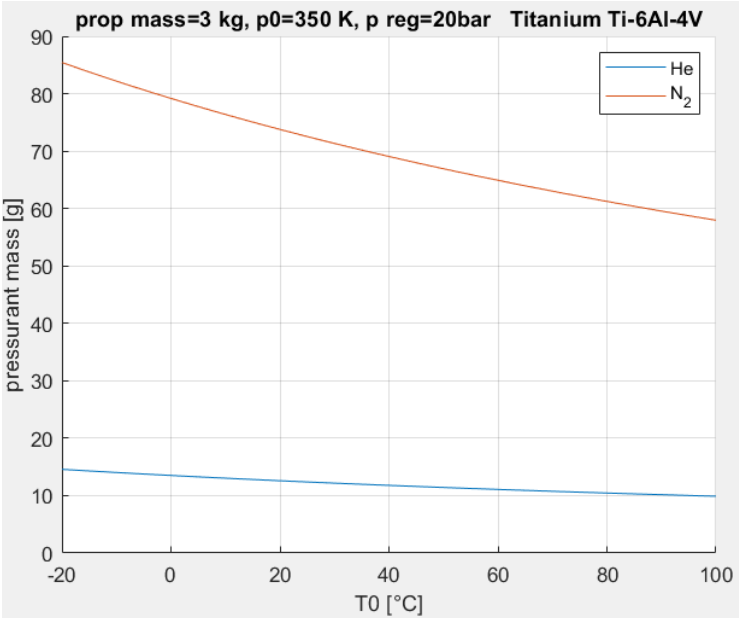
Annex IV - 3 External diameter of the sphere as a function of the propellant mass

Again, with the increase in the propellant onboard, the diameter of the sphere will increase and thus the volume occupied.

- **Variation of the storage temperature**



Annex IV - 4 Total mass as a function of the temperature

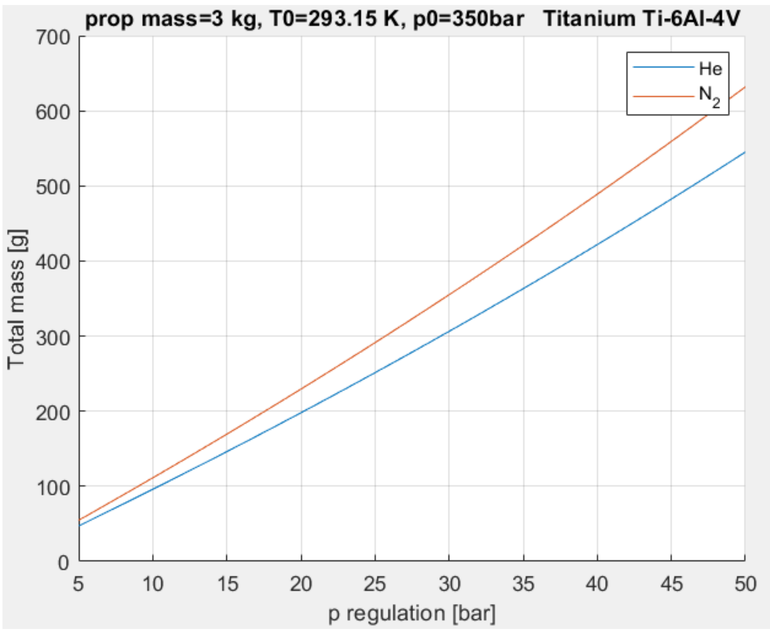


Annex IV - 5 Gas mass as a function of temperature

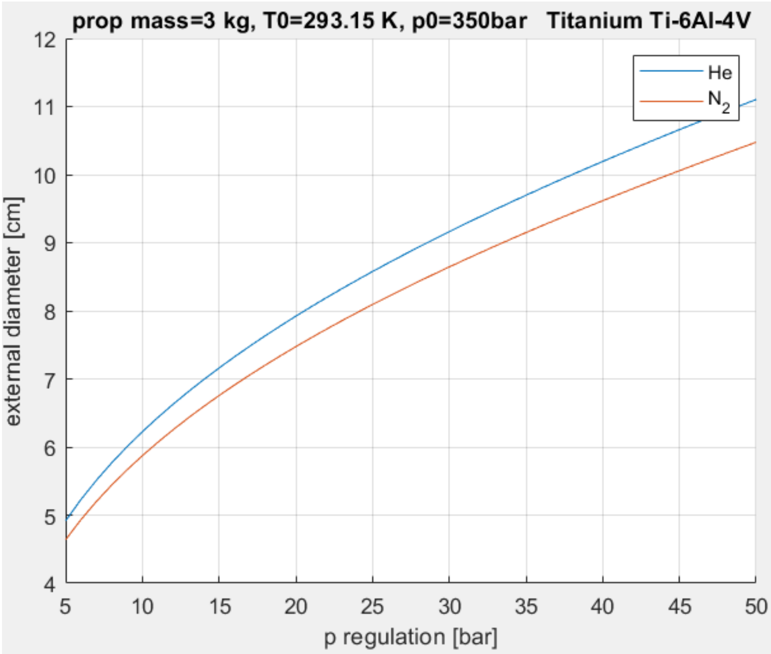
If the temperature rises, at the given conditions, the mass of the pressurant slightly decreases, considering that it would make no sense neither storing at very low or very high temperatures. This effect is bigger using nitrogen.

- **Variation of the pressure of regulation in the propellant tank**

Note that the following charts are made considering the pressure at the end of the discharge always five bars greater.



Annex IV - 6 Total mass as a function of the pressure of regulation

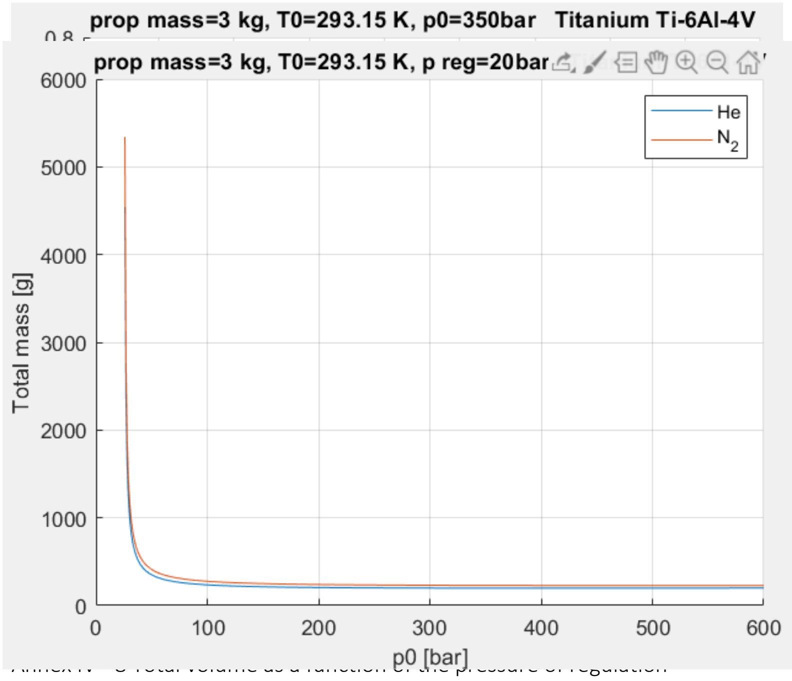


Annex IV - 7 External diameter as a function of the pressure of regulation

Here the results are similar to the ones presented above. The pressure of regulation of the propellant tank makes the pressure at the end of the discharge in the pressurant tank necessary higher and to mass and volume increase. As before, helium favours the mass and nitrogen the volume.

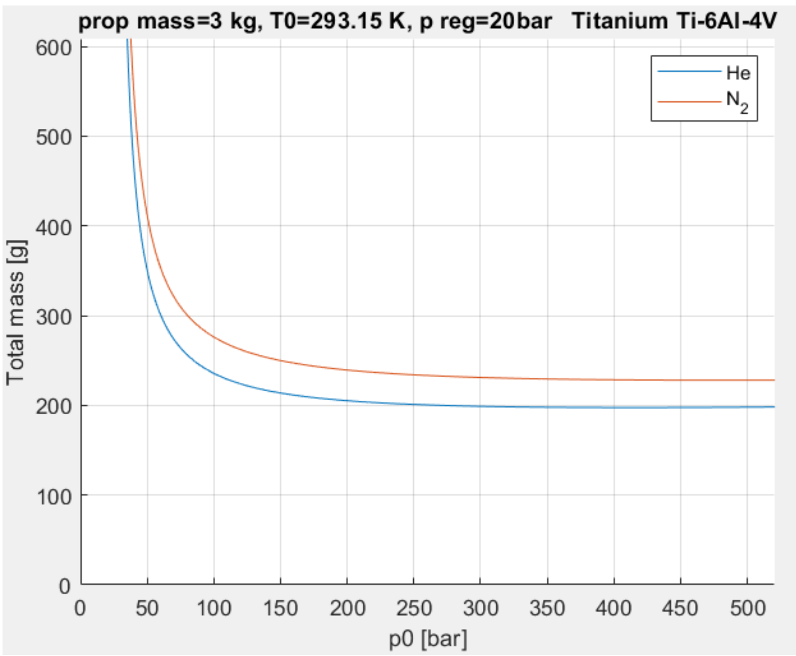
- **Variation of the pressure of storage**

The pressure of storage of pressurants can be very high, since they do not require much room usually. Here in this case the volume occupied is fundamental.

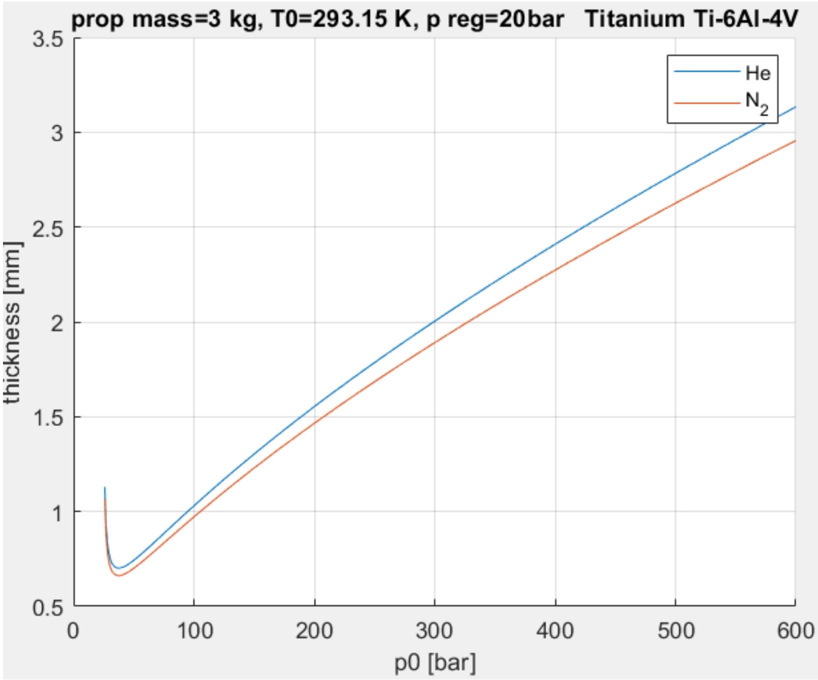


Annex IV - 9 Total mass as a function of the pressure of storage

As shown, mass reduces with increasing pressure of storage, but after around 200 bar seems not to be any gain. So why to choose higher?

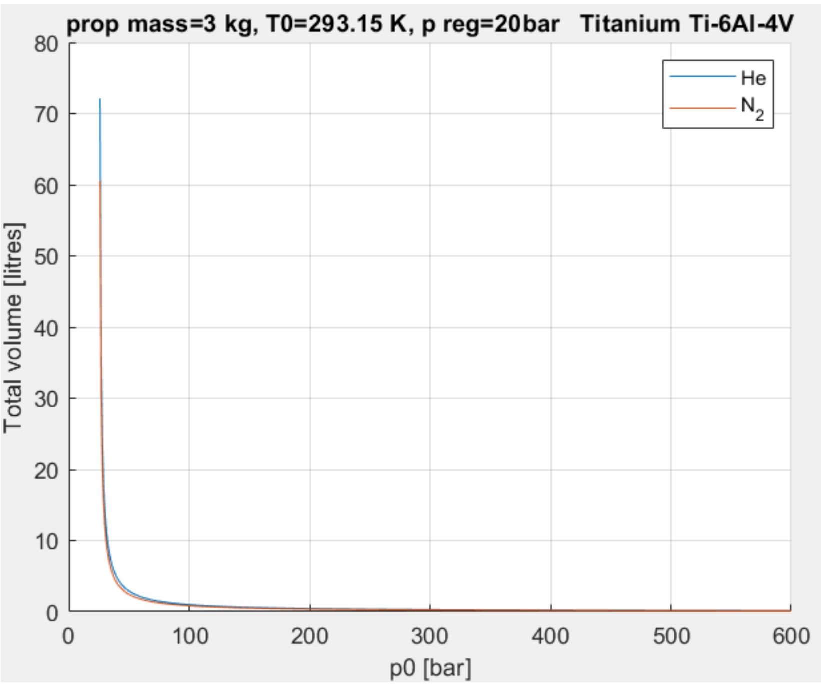


Annex IV - 10 Total mass as a function of the pressure of storage (zoom)

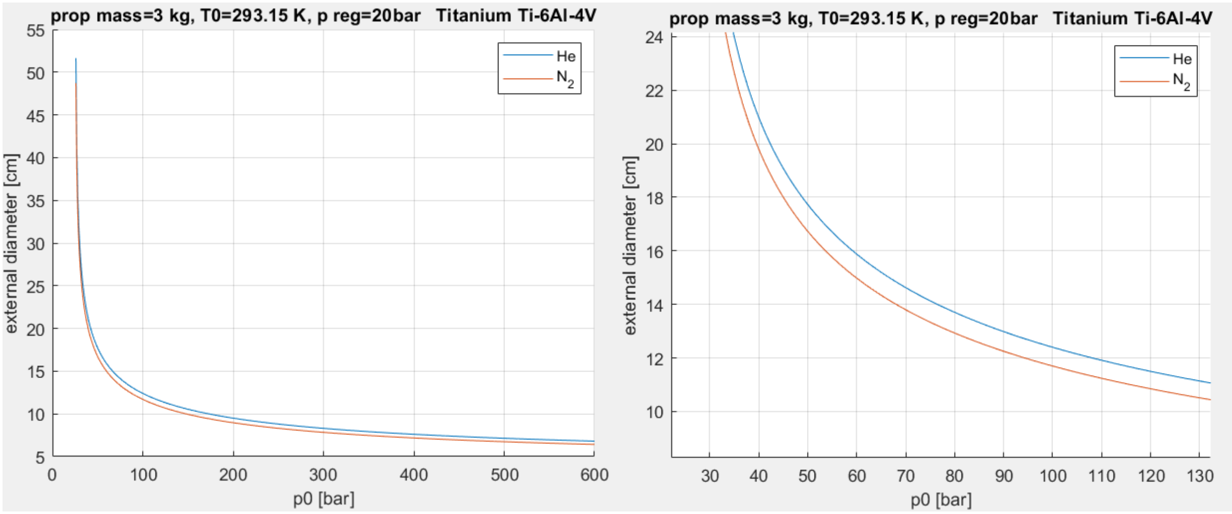


Annex IV - 11 Wall thickness as a function of pressure of storage

Annex IV - 11 shows that there is a minimum for the wall thickness. Initially, the thickness decreases because the storage pressure is really close to the pressure at the end of discharge, so high volumes are needed and thus wide diameters. Here, the minimum value of thickness is obtained with pressures around 37 bar.

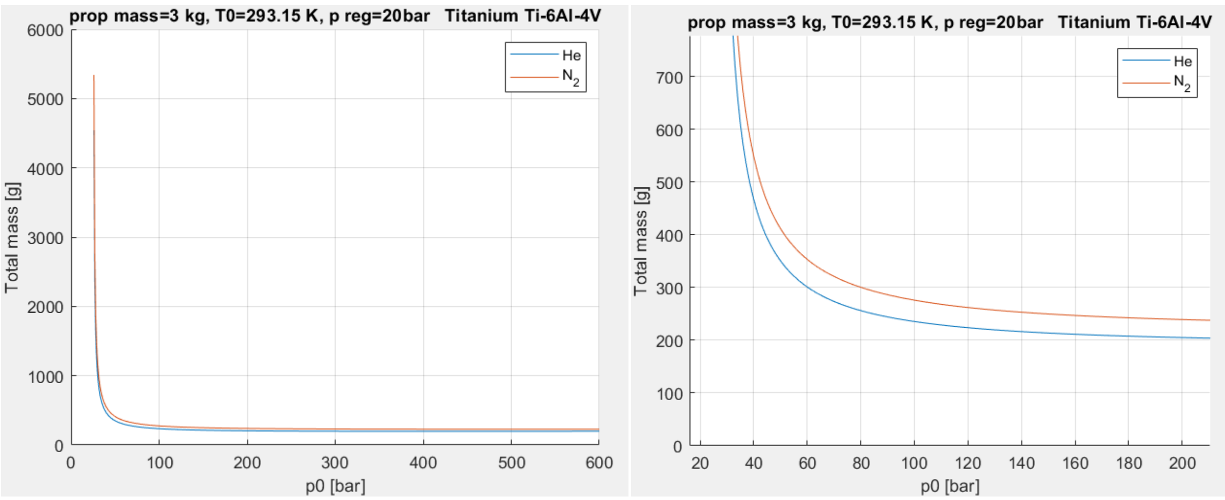


Annex IV - 12 Total volume as a function of the pressure of storage



Annex IV - 13 External diameter as a function of the pressure of storage

Here it is clear that, for pressures around 37 bar, the external diameter is still much greater than one may desire, because of encumbrances. The sphere in general is not really efficient to the management of spaces, so even half of a centimetre could make the difference.



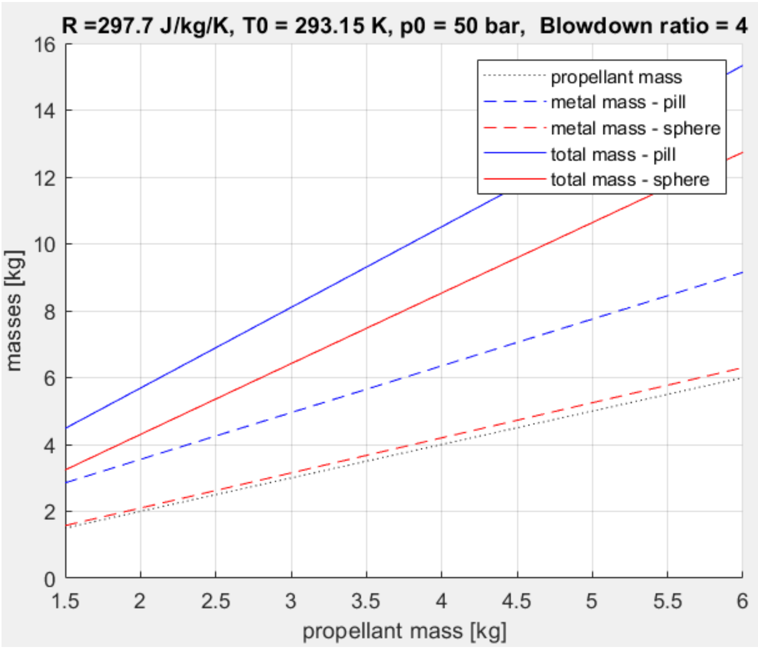
Annex IV - 14 Total mass as a function of the pressure of storage

As a confirm of what being said before, for pressures around 37 bar mass would be noticeably greater than with higher pressures. After around 200 bar, no appreciable differences in mass are seen, but this is not true for the external diameter. To conclude, one may prefer to set very high pressures to save space in terms of minor external diameter.

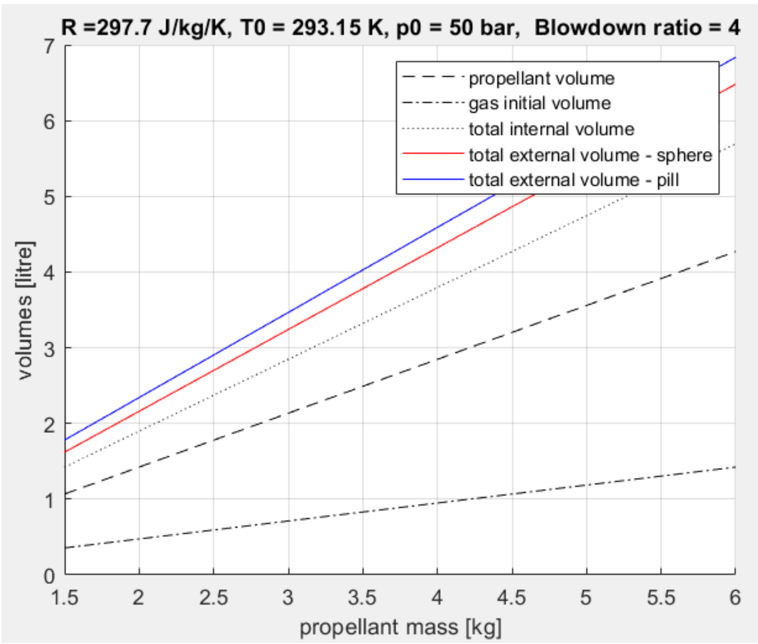
Annex V – Propellant Tank - Blowdown

Here some results of the propellant tank design in case of blowdown feed system will be presented, even though it is not a convenient option. Since all the gas required for the expansion must be stored together with the propellant in the same tank, pressures will rise, so masses and volumes will do. The decided internal diameter of the cylinder of the pill is 12 cm. Lately it will be varied to see what it affects. Here St Steel AISI316L is utilised, because titanium is not compatible with hydrogen peroxide. Results about mass with Titanium would be about a half of those with steel, because the density is almost half and the yield strength is much higher..

- **Variation of the propellant mass**

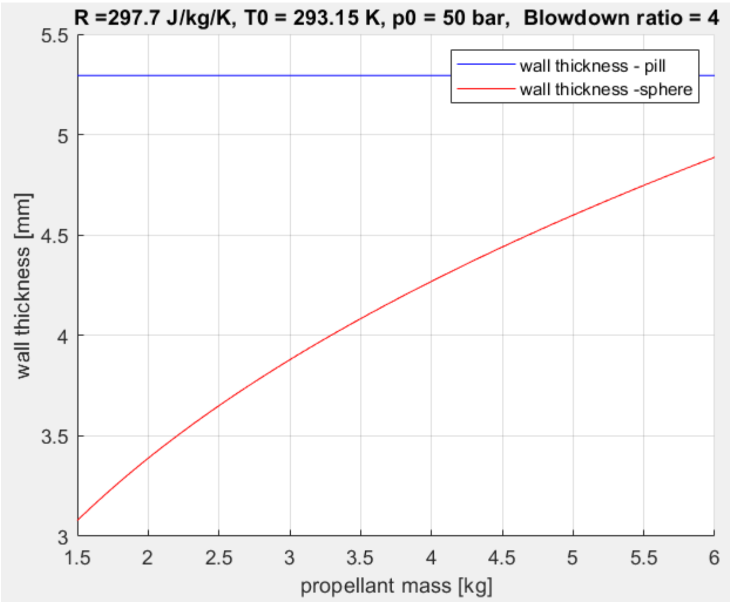


Annex V - 2 Distribution of masses as a function of the onboard propellant mass



Annex V - 1 Distribution of volumes as a function of the onboard propellant mass

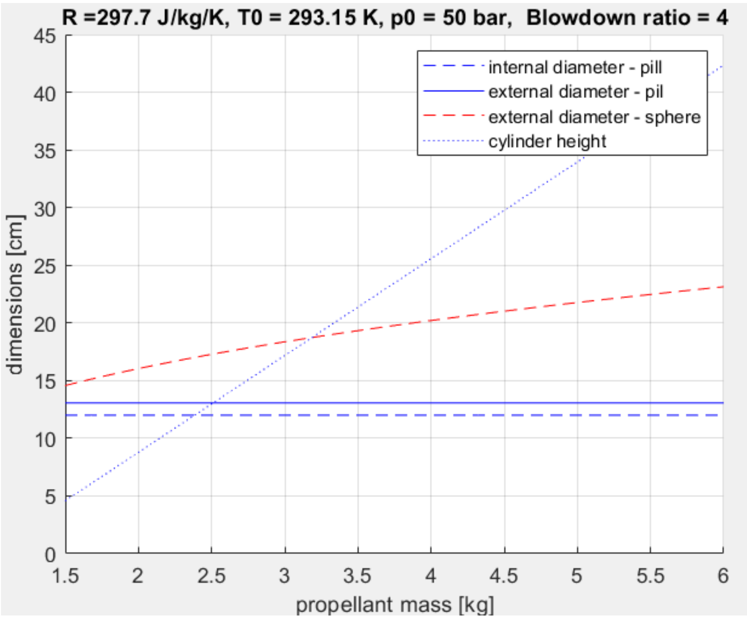
Increasing the propellant to be stored means to increase its volume and since the blowdown ratio is fixed, the consequent mass and volume of pressurant required increases. Note that, differently from the pressure regulated case, here the total resulting mass is always beyond two times the mass of propellant. This is due



Annex V - 3 Wall thickness as a function of the onboard propellant mass

to the large amount of metal mass required to bear high pressure in the tank, as a consequence of high wall thickness.

The wall thickness for the pill is constant, because it depends on its internal diameter, which is fixed at 12 cm here. The wall thickness of the sphere, instead, rises as the volumes increases because of the increasing stresses.

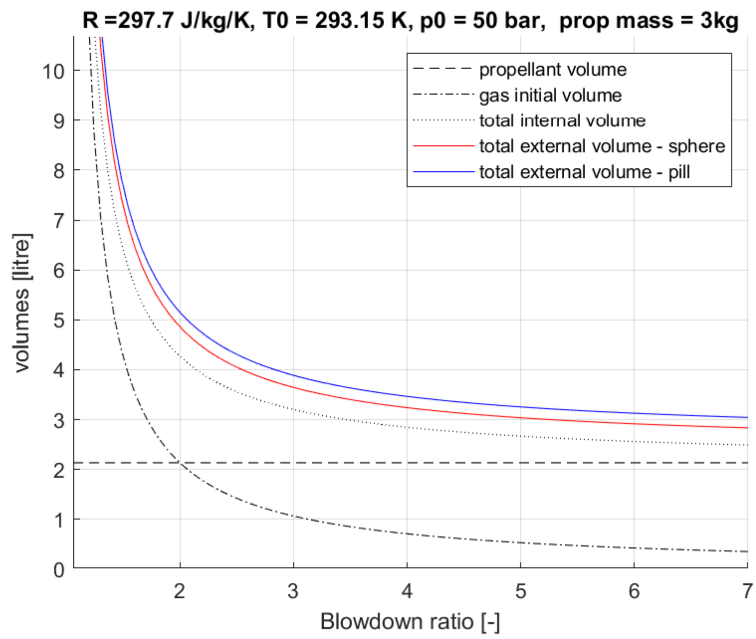


Annex V - 4 Geometrical dimensions as a function of the onboard propellant mass

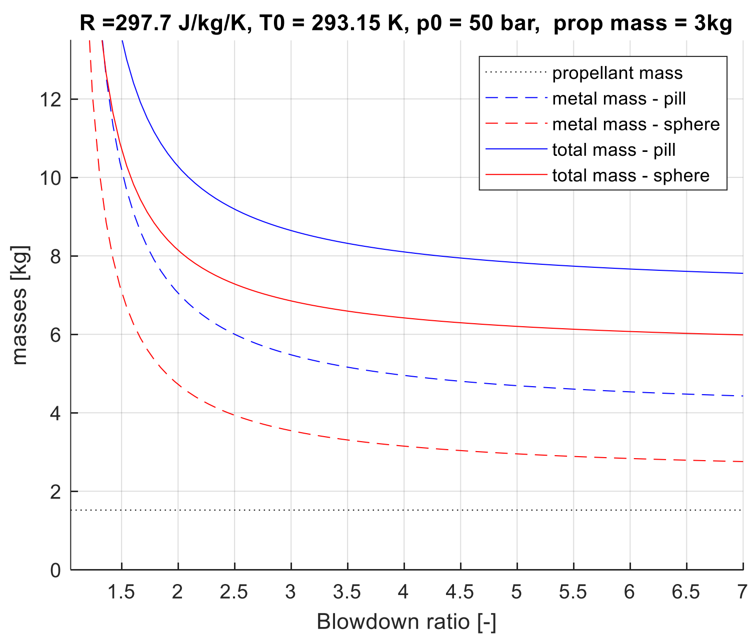
AS just said, the diameters, both internal and external, remain constant. The height of the cylinder rises linearly, because once the internal diameter is fixed, the volume of the sphere is fixed and the additional volume depends only on the height of the cylinder. In the case of 3 kg of propellant, the height results to be about 17 cm, which, added to the external diameter of the sphere of the pill of about 13 cm, is about 30 cm, the maximum dimension of the 12U spacecraft. This is to say that, using this configuration, would mean to occupy almost all the available room in the spacecraft for the propellant tank, without considering valves, tubes ecc.

- **Variation of the blowdown ratio**

Blowdown ratio affects the quantity of gas to be initially present in the tank. The higher the blowdown ratio, the bigger the expansion of the gas and thus the higher the descent of the pressure.

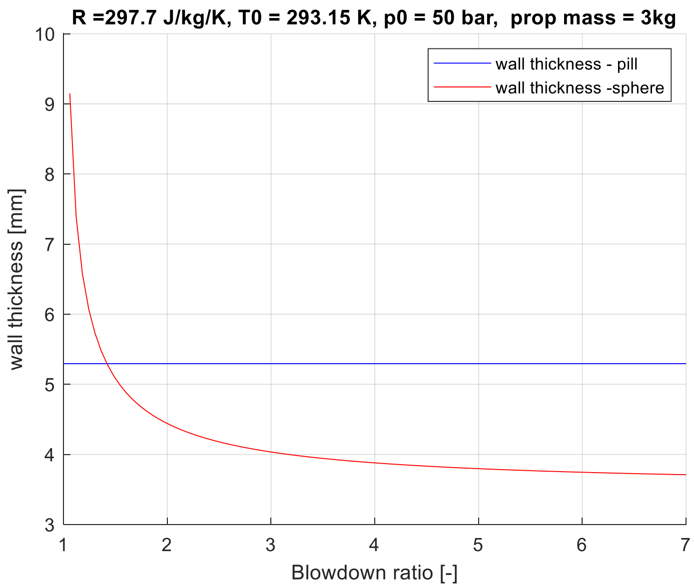


Annex V - 6 Distribution of volumes as a function of the blowdown ratio

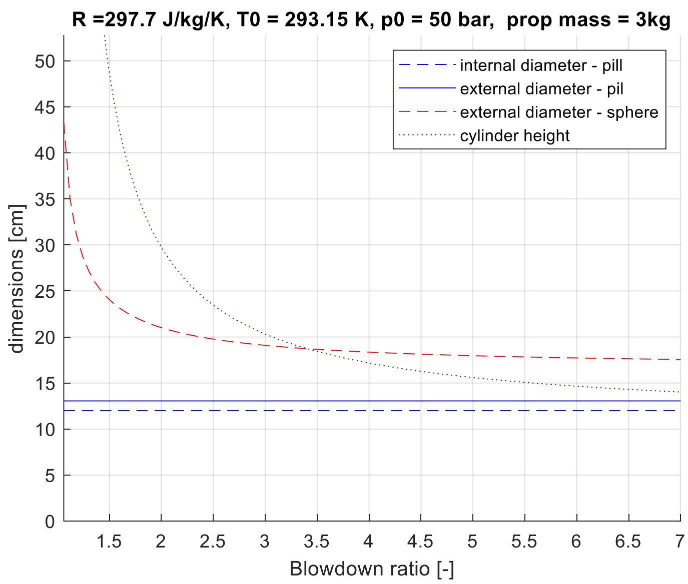


Annex V - 6 Distribution of masses as a function of the blowdown ratio

As shown in the last wo graphs, if the blowdown ratio il low volumes and masses reach extremely high values. This is because, even though the propellant mass is constant, having low blowdown ratios means that the gas has to expand less than what it would do if the blowdown ratio were high. Despite that, low blowdown ratio means that the initial volume of the gas has to be high, so that the total internal volume increases, as shown in the figures. On the other hand, the less the gas expands, the less the pressure drops, so this could even be even advantageous. In turn, this transforms into an increase in the dimensions. If the blowdown ratio increases, the situation is the contrary of what just described.



Annex V - 7 Wall thickness as a function of the blowdown ratio

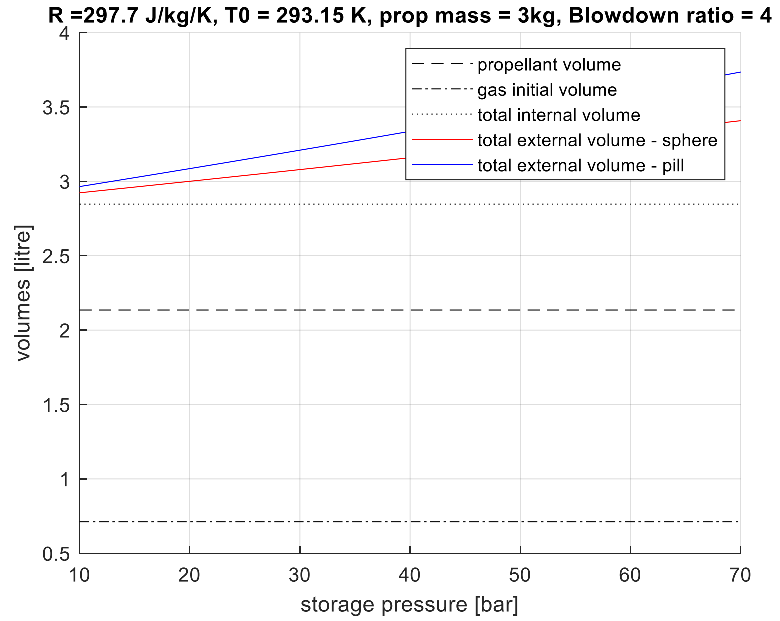


Annex V - 8 Geometrical dimensions as a function of the blowdown ratio

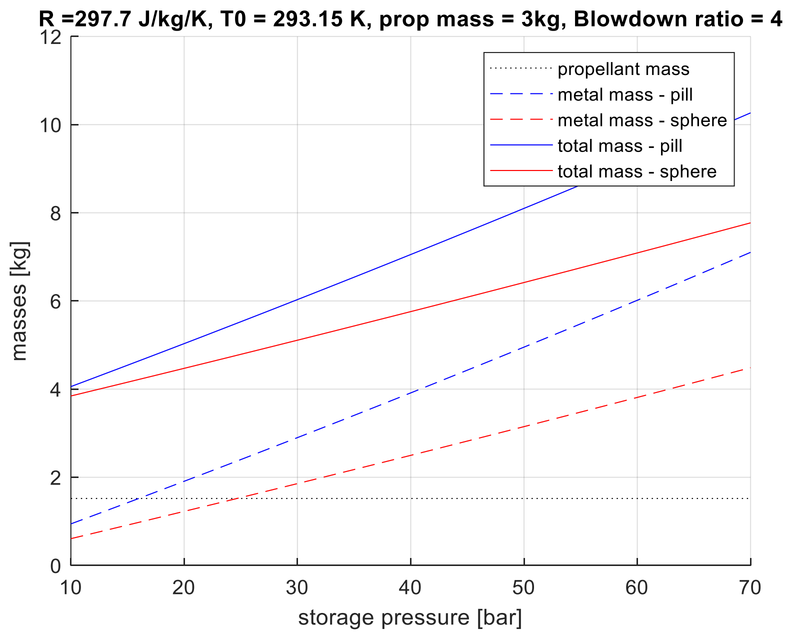
As seen before, since the internal diameter of the cylinder is fixed, so is its thickness and thus the external diameter. As usual, this is not true for the sphere, which has one only parameter to be evaluated, given the internal volume, that is the diameter.

- **Variation of the storage pressure**

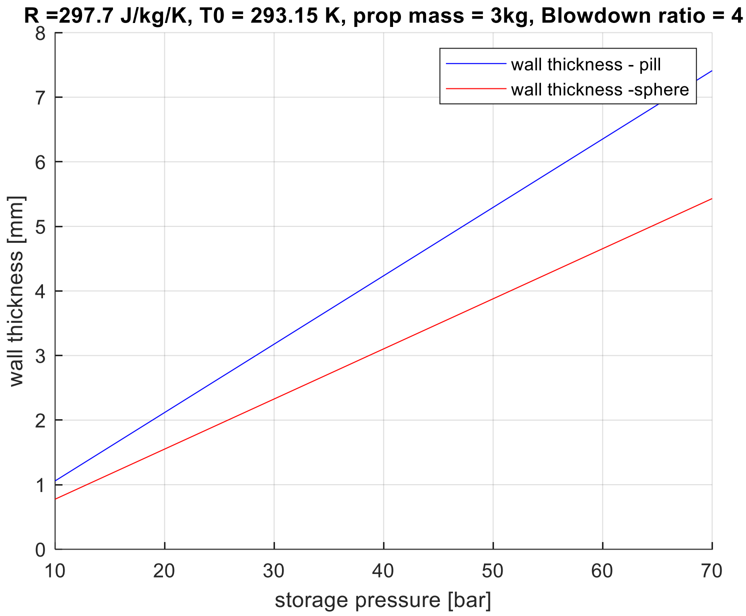
The results of variating the storage pressure are total analogous to those presented in Annex III - 1Annex III – Propellant tank. At given propellant mass, thus propellant volume, at fixed blowdown ratio, increasing the pressure op storage means higher stresses on the metal of the tank, thus higher wall thickness with consequently increase in mass, volume and geometrical dimensions.



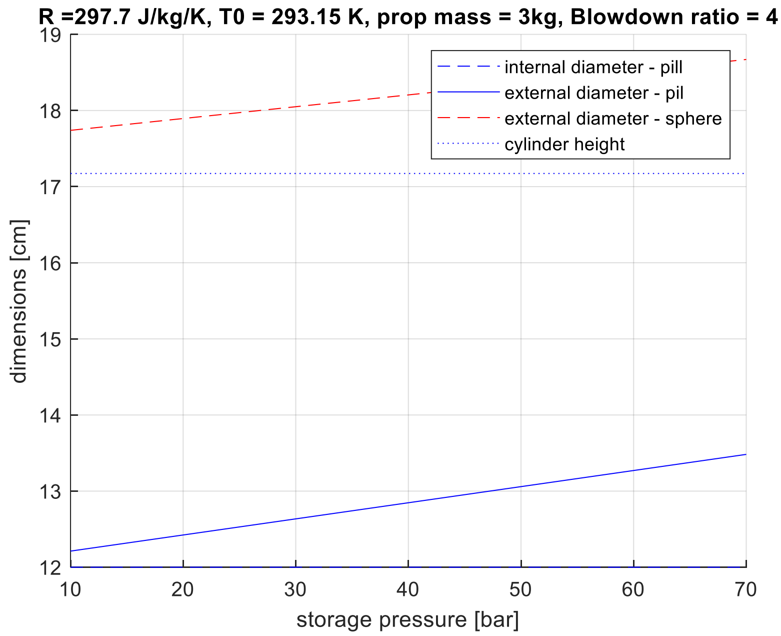
Annex V - 9 Distribution of volumes as a function of the storage pressure



Annex V - 10 Distribution of masses as a function of the storage pressure



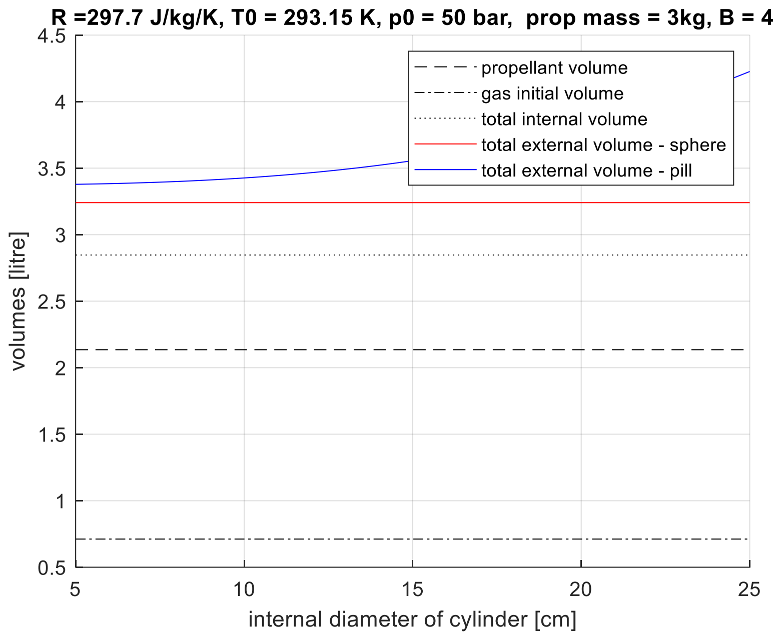
Annex V - 12 Wall thickness as a function of the pressure of storage



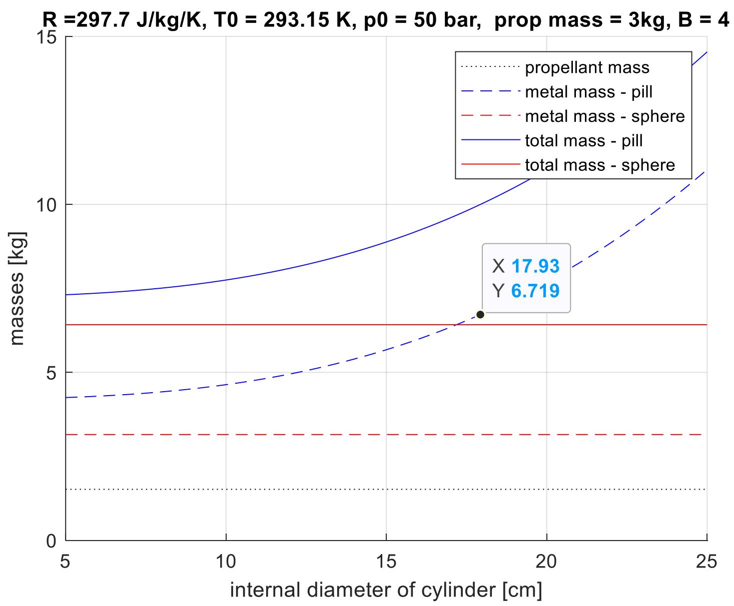
Annex V - 11 Geometrical dimensions as a function of the pressure of storage

- **Variation of the diameter of the pill**

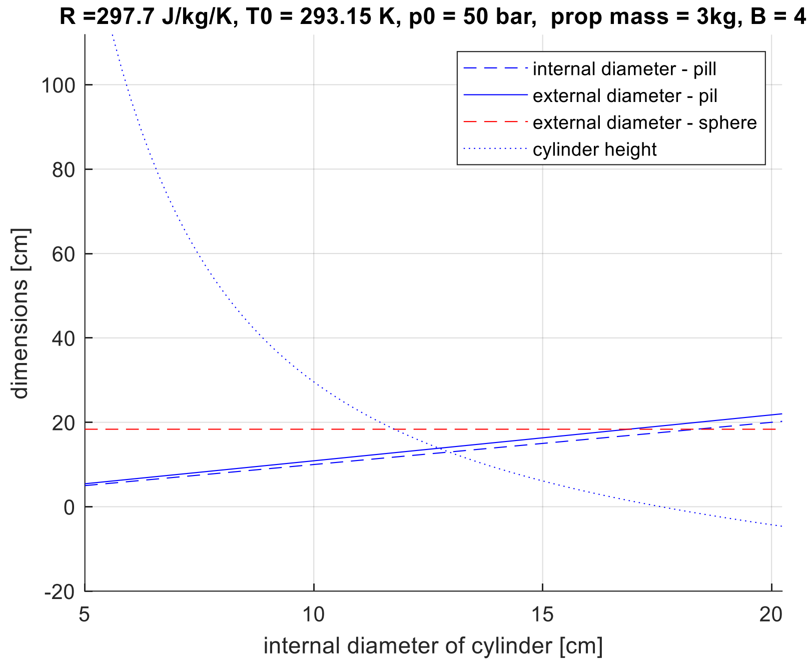
Again, these results are analogous to those in Annex III – Propellant tank in the proper section. Fixing the propellant mass and thus its volume and fixing the blowdown ratio, the total internal volume required is set. Also setting the internal diameter of the cylinder leaves a degree of freedom on its height. If the internal diameter exceeds the diameter of the sphere, then the resulting volume would be higher than required, so the height would become negative. This is not acceptable, so the only data to be considered are those which give a positive height. The limit case is when the diameter of the cylinder equals that of the sphere: the pill degenerates into a sphere.



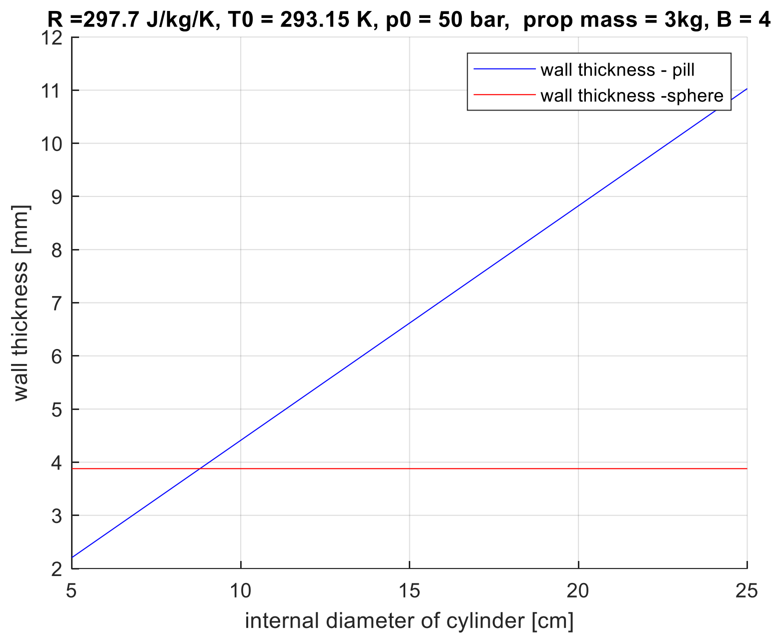
Annex V - 14 Distribution of volumes as a function of the internal diameter of the cylinder



Annex V - 13 Distribution of masses as a function of the internal diameter of the cylinder



Annex V - 15 Geometrical dimensions as a function of the internal diameter of the cylinder



Annex V - 16 Wall thickness as a function of the internal diameter of the cylinder

The only material considered here is steel, but considering another material would mean to modify the wall thickness because of the yield strength and the masses and volumes. As an example, Titanium would reduce masses and encumbrances, at given conditions.

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