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QinetiQ Space nv

PROSPECT: a Matlab tool for spacecraft propulsion comparison

Author:
Leone Maurilio MAGAZZU'

Supervisor:
Dr.Salvo MARCUCCIO,
Ir.Julien TALLINEAU

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To my family, Antonio and Giovanna,
who experiences more frustration when I
struggle and more joy when I succeed than I
could ever manage to feel myself.

"Following the light of the sun, we left the Old World. "

— Inscription on Columbus' caravels

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Abstract

Department of civil and industrial engineering
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PROSPECT: a Matlab tool for spacecraft propulsion comparison

by Leone Maurilio MAGAZZU'

The aim of this thesis is to create a methodology that estimates the characteristics and the mass breakdown of a propulsion system. This includes thrusters, propellant, propellant management, and power system requirements for end-to-end missions.

The first part of the project will concern selecting components from various propulsion systems to build a large database; It will contain data from the previous missions of QinetiQ as well as informations about the state of art technology that is currently available in the market.
Different mission requirements will be established with a GUI by the final user. These mission requirements are fundamental for the selection of the propulsion and launcher system. This selection-process doesn't take the other subsystem and the relations into account.

The second part of the project will concern the main MATLAB model, based on the requirements for the mission orbit injection, the station keeping and the disposal.

The results of the MATLAB model will show the best possible pick concerning launcher, propulsion system and a combination of S/C propulsion system and launcher, based on the user defined mission requirements.

Acknowledgements

I am very grateful to all the people I have met along the way and who have actively contributed to the development of my master thesis. In particular I would like to show my gratitude to Ir. Julien Tallineau from the QinetiQ space nv, for the time, patience and the know-how he provided me during the wonderful internship and Prof. Salvo Marcuccio from University of Pisa for the support and information given during all the fulfillment of the thesis.

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Rivolgo quindi la mia gratitudine a Paolo per tutte le sue affettuose premure e i suoi preziosi consigli; a Nicolai, amico sincero fin dai primi giorni d'università, che mi ha incoraggiato a seguire la mia strada; a Peppa una dolce presenza con sempre una parola affettuosa nei miei confronti; a Pigo, Fabio e Giuseppe grazie ai quali ho imparato a nerdeggiare con stile; a Caterina l'unica livornese la cui allegria mi ha contagiato fin dal primo momento; a Pippo che mi ha accolto affettuosamente in EUROAVIA e come compagno d'avventure; a Valentina, cara e leale amica euroaviana; a Martino, Irene, Marta e Matteone con cui ho condiviso indimenticabili momenti nella nostra biblioteca; a Quirino, "lo zi", sempre disponibile, con storie al limite dell'inverosimile e che mi ha fatto scoprire la Guinness.

Un ringraziameto speciale va poi a coloro che sono diventati la mia seconda famiglia a Pisa: a Vincenzo, detto il nonno, il più flemmatico tra tutti gli amici, a Luisa che con affetto ci ha sempre accettato anche nei nostri momenti più giocosi, a Federico trainante e determinato ma sempre sorridente, ad Angelo, Giovanni e Beniamino, che tra discorsi incomprensibili, sono sempre pronti a farti ridere, a Giusy che ammiro per la sua singolarita e la sua tenacia.

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Ps. Nico, ricordati che non è normale che i Protoss si espandano prima degli zerg.

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List of Abbreviations

AOCS	**A**ttitude and **O**rbit **C**ontrol **S**ystem
AHP	**A**nalytic **H**ierarchy **P**rocess
COG	**C**entre **Of** **G**ravity
EP	**E**lectric **P**ropulsion
ESA	**E**uropean **S**pace **A**gency
FCU	**F**low **C**ontrol **U**nit
GEO	**G**eostationary **E**arth **O**rbit
GUI	**G**raphical **U**ser **I**nterface
GTO	**G**eostationary **T**ransfer **O**rbit
ITAR	**I**nternational **T**raffic in **A**rms **R**egulations
LEO	**L**ow **E**arth **O**rbit
MS	**M**icro**s**oft
N.N.	**N**o **N**ame
PPU	**P**ower **P**rocessing **U**nit
RAAN	**R**ight **A**scension of the **A**scending **N**ode
S/C	**S**pace**c**raft
SS	**S**unl **S**ynchronous **I**nterface
TRL	**T**echnology **R**eadiness **L**evels
w/o	**W**ithout
VEGA	**V**ettore **E**uropeo di **G**enerazione **A**vanzata
VESPA	**VE**GA **S**econdary **P**ayload **A**dapter

Physical Constants

Earth radius	$R_e = 6.371\,\mathrm{km}$
Gravitational acceleration	$g = 9.8067\,\mathrm{m/s^2}$
Standard Earth gravitational parameter	$\mu = 3.986\,004\,418(9) \times 10^{14}\,\mathrm{m^3/s^2}$(exact)
Universal gravity constant	$G = 6.670 \times 10^{11}\,\mathrm{m^3/\,kgs^2}$

List of Symbols

a	semi major axis	km
C	Celsius	°C
e	eccentricity	
i	inclination	rad
I_{sp}	specific Impulse	s
K	kelvin	°K
kg	kilogram	kg
k€	kilo euro	€
m	meter	m
mm	millimeter	mm
M€	million euro	€
Pa	Pascal	Pa
t	time	s
ω	argument of periapsis	rad
Ω	RAAN	rad

Chapter 1

Introduction

1.1 Purpose of the thesis

The number of propulsion system that have been qualified with TRL 7 or more, in the past decade, have considerably increased the relevant possibility of different configuration for future generations of spacecraft.
However, the systems level designers are usually not interested in how the components works, they care about what it will provide to their spacecraft in terms of thrust and total velocity change and what it will cost in terms of mass, power, volume, risk, or money.
These values are analysed within an initial design phase in which some initial calculations are done to chose thruster, propellant storage, and (if necessary) electrical power system. When designers have to chose a specific piece for their project, they have to select it from a huge number of possibilities and configurations and, this process can be tedious; When the designer has to run some simulations with several different options for the space system in which the number of satellites and their masses may vary, changing drastically the propulsion requirements, or when some initial calculations need to be run to compare the merits of one propulsion system comparing to another one to make an initial design decision influencing the choose of the launcher or even of the launching pod; it's at this moment that my program wants to simplifies the job.
The purpose of my studies is to develop a program capable to estimate simply, quickly and reliably the mass of the propulsion system for end-to-end missions, that can reduce the time and the resources spent given to the system designer.
The general design of a spacecraft is often based on the scaling equations to estimate the mass of a system for the preliminary design purpose. The scaling equations have been long used to define the rocket lift, the orbit insertion or the propulsion system, but only a small quantity of them is applicable to a wide range of propulsion system.
The program I compiled intends to range integrated tanks, launchers, thrusters and basic feed system components available from cubesats to GEO missions. The database that has been accumulated in the course of this study has been divided into launchers, thanks, feed system components and in twelve distinct type of propulsion

thrusters, chemical and electrical, with a variety of propellant to obtain a system closer to reality.
Complementarily to the database researches have been made about: launchers for the missions, orbital maneuvers to reach the parking orbit and the related propellant consume, propellant for station keeping and the disposal maneuver.
The disposal maneuver is evaluated for all the missions due to IADC regulations [**IDAC**], which impose that controlled re-entry will become a mandatory starting from 2020, with few exceptions in which the re-entry has to be feasible within the overall scope of the mission. This necessity is caused by the common practice of abandoning the spacecrafts at the end of their missions, increasing the probability of collisions with the active spacecraft, becoming a threat to space operations and diminishing the overall controllability over this type of menace.
My thesis describes how the program has been created, including the database and the considered specifications, the equations that were developed from the database, the ones used to model the other parts of a propulsion system, few necessary assumptions, validation, and some interesting applications.

Chapter 2

Research methodology

Before the examination of the program itself, it's important to understand the literature took into account for this study, the preliminary requirements and constrains that were the guideline through all the process of research and development.

2.1 Literature review

The overview of the scientific papers, books, the websites about the different technologies and experiments have been of great help when it came to choose the technological options in order to establish the database and the mathematical equations for any possible mission scenarios.

The first document of interest is the Study made by Thomas Chiasson and Paulo Lozano [33] as they made an estimation and a mass breakout of the propulsion system providing less than 10N. Their work has been a good start to comprehend the basic request for a propulsion design.

The modeling of the chemical feed system was based on the methods used by Charles D.Brown [5] and Jason Hall [15], whose books provide a theoretical background and practical guidance to design.

In the documents [30] by J.D.Shelton there is an optimization tool that describe the use of computers to model a system propulsion. This process focuses on the development of the liquid propulsion system. From a list of launcher found online[32], it has been possible to identify all the rockets available in the market and through the data already collected in QinetiQ to build a spreadsheet with possible configurations, weight, launch base and the most accurate possible price.
Obviously all the data collected were cross checked with the factory user manuals of the launchers.
The same procedure was followed to insert the thrusters and the feed components in the database.

For the manouvers using the high thrust propulsion most of the first order equations has been taken from G.Mengali[11] , while the low-thrust orbital manouvers by J.R. Pollard[27] where it's described a calculation of the velocity increment and the trip time using simple pre-defined steering programs and an averaging technique that ignores the short-term variation of the orbital elements. This allows a rapid assessment of design options to place a satellite into the desired orbit.

The Analytic Hierarchy Process[19] by Jeff Kunz is a trade study that analyze wich criteria are more important than the others and those that should weigh heavier on the final score.

2.2 Requirements

The first requirement was to generate a easily updatable database that allows the user to integrate new technologies or even refine the data without requesting any work on the program.
This gave the possibility to have a program always updated with the most accurate data possible and the informations from state of art technology.
Related to this was the requirement for the main program to easily access the data and uniquely select the parameter needed.
The program must follow some of the requirements internal to QinetiQ: to be developed as a Matlab class code and to be integrated inside the company proprietary toolbox.
Another need was the user-friendly interface to define the characteristic parameters of the mission scenario that the propulsion system has to deal with. As output the requirement is a complete evaluation of the optimal thruster and louncher system, and all the informations about the weight of the solutions analyzed, components took into account, producers, price and overall propellant and power required.

2.3 Constraints

Due to the difficulty of the project, some constraints have been applied to simplify the calculation and the design.
First of all I based the research on a first order analysis design so I didn't take into account the problem of sloshing, the I_{sp} drop during life time, weight of feed line tube, the minor components as filters and the integration procedure time and cost.
Some of the constraints were directly defined by the producers themself due to copyright regulations, many of the limitation were related to the price of the components and particularly with the EP, information on the PPU and interaction with other subsystems.
A particular constrain was directly connected with the EP thrusters if

the time to reach the parking orbit was more than the working time, the mission would not be taken in consideration. Other constrains has been requested by the company: a light and quick program, the run time doesn't have to exceed 10 minutes, the orbit injection in LEO or GEO are perfectly circular with a possible error of 40km of altitude and 0.02°of inclination.
GTO orbit for all launcher is the same as ARIANE 5, a =24630 km, e =0.716, ω =3.10668607 and the inclination depends oh the launcher pod. The disposal phase must be calculated for LEO and GEO missions, while MEO orbits will not be considered at the moment, because the company has no project in that area of space.
For LEO mission the satellite has to reach the orbit of 150 Km of altitude to have an uncontrolled re-entry, while for the GEO missions, the procedure defines to reach the Graveyard orbit.

Chapter 3

Program Structure

In this chapter I focused on the working flow of the program, from the first interaction of the user to the results that are displayed, following the logic iteration inside the program.
Running the program, the user has to interact with a GUI (graphical unit interface) Fig. 3.1 to define the mission scenario to evaluate and other major parameters:

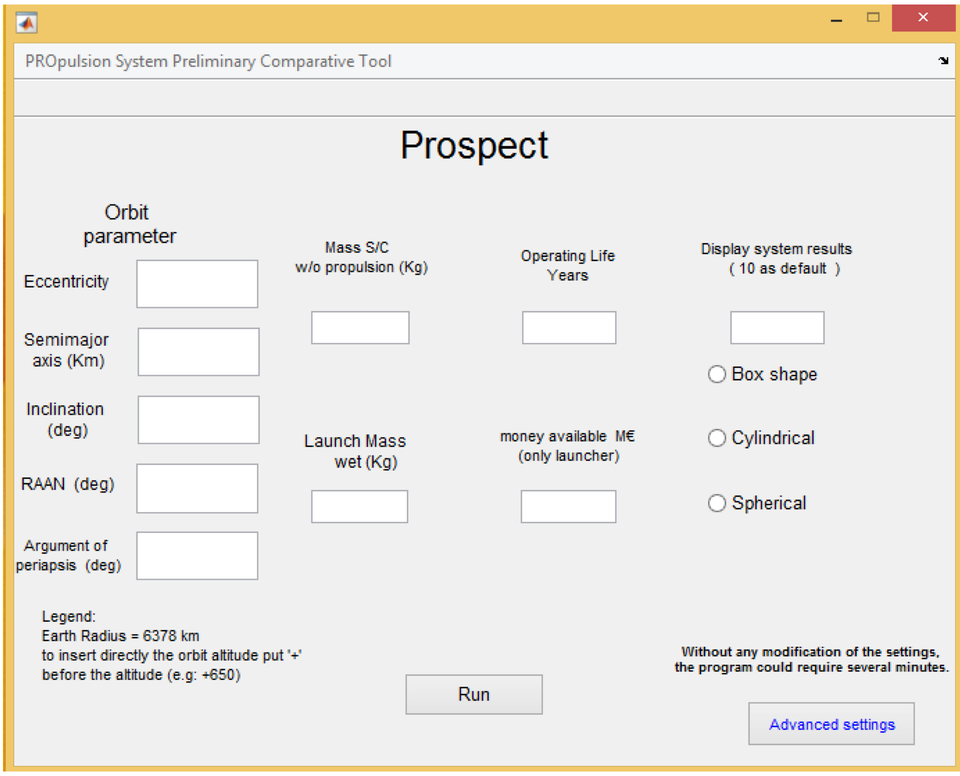


FIGURE 3.1: Prospect GUI

Final orbital parameter Eccentricity, semimajour axis, inclination, RAAN and Argument of pheriapsis are needed to define the final orbit of the S/C.
Some constrains are applied directly in the GUI: the satellite can't have an orbit that will impact with the Earth and orbits that request an orbit outside GEO generate an error that reboots the GUI itself.

Mass of the S/C The user has to insert two masses: launch wet mass of the satellite, one of the main requirements for the choice of the Launcher, and the mass of the S/C without propulsion and propellant.
The dry mass is defined as the wet launch mass minus the mass of the propellent needed for the mission.
The mass budget of the propulsion system (without propellent), according to the the general rules from the Space Mission Analysis[20], should be the 7% of the total dry mass.
After a preliminary analysis, to obtain a more realistic approach,it has been decided to use the difference between the wet launch mass and the dry mass without propellant as the mass constrain of the propulsion system comprehensive of propellant.
This influenced the design of the system and its results, but such a strict mass budget, especially for small satellite, didn't give a realistic analysis of the problem.

Operating life This parameter was introduced to establish the volume of the perturbations that the S/C has to counteract and to calculate the related propellant and propulsion for AOCS.

Price launcher budget Allows to define the cost budget for the launcher selection.

Display system results Considering the number of datas to evaluate, the program can introduce limits defined by the user of the quantity of solutions that could be displayed in the result graph chart.

S/C dimension and shape This requirement has a multiple influence on the code: the dimensions are necessary to define if the S/C could be accommodated inside the launcher and witch kind of perturbations would influence it.

All this datas are saved and accessible in every stage by the class of the program

3.1 Launcher evaluation

From the user's data, the program run its first Class, named @CLaunch. Its purpose is to compare all the launchers in the database through a working flow, Fig.3.2.

1. The data of orbit, mass, dimension and price of the launcher, are taken from the GUI;

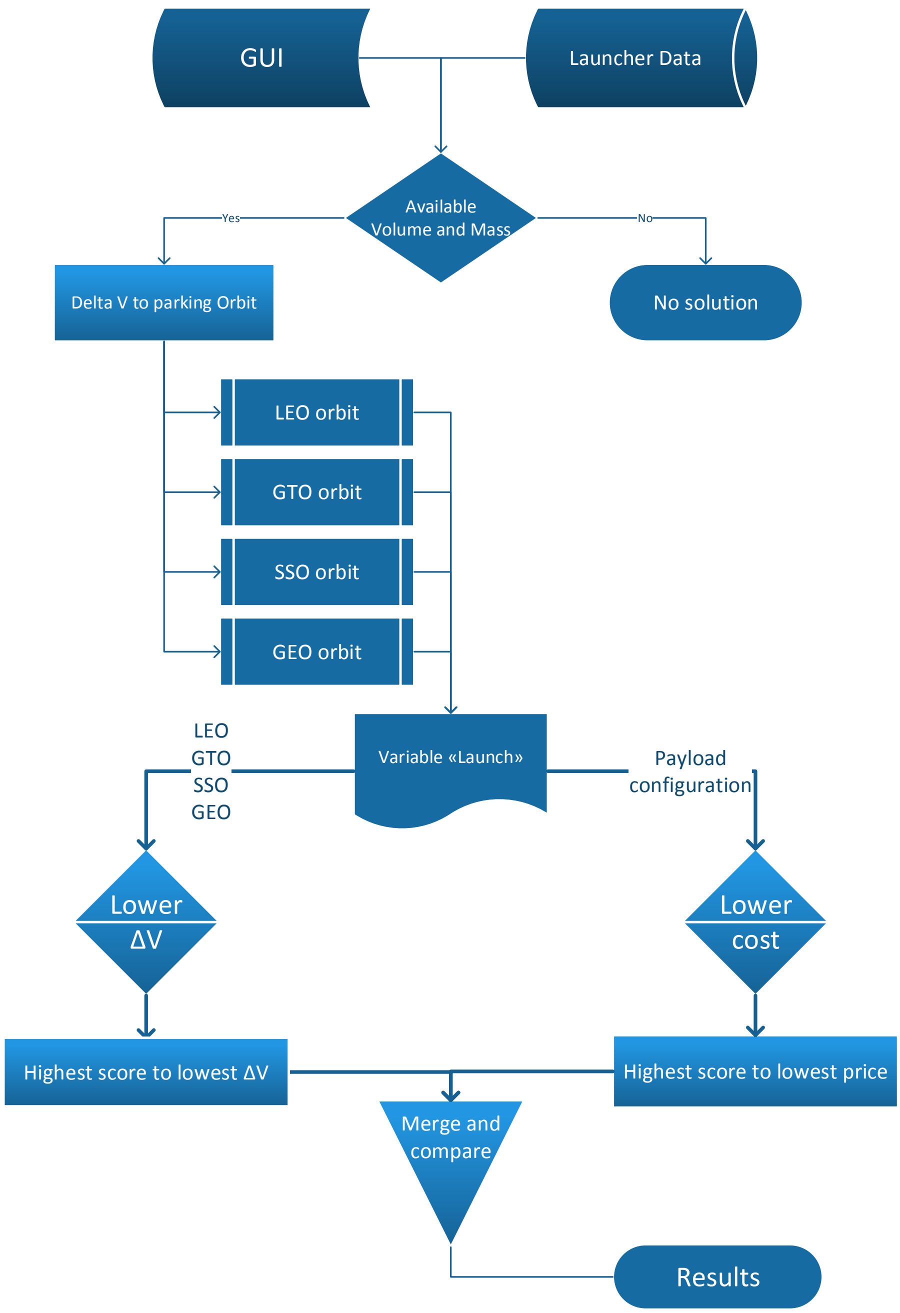


FIGURE 3.2: Flow graph of Launcher selection

2. The data of launchers are retrieve from the database. The program select among all the launchers which ones verify the costraints for the S/C dimensions. If there is no launcher available for the requirements, the program restarts for no solution found.

3. The launchers with the right dimension to carry the S/C, are then analysed to choose to ones that can carry enough mass for

the nees of the mission. There are three available possibilities :
The launcher configuration is able to transport the S/C.
The configuration can fit the S/C, but the constrain of mass don't fit, so, if another bigger configuration is available, the program analyzes and evaluates that.
The configuration can fit the S/C, but the mass is bigger that the one that can be carried and the launcher is deleted from the queue of the solutions.

4. After analyzing every launcher with the described procedure, and finding the launcher available for the request the program calculates the required ΔV, velocity needed to perform a space orbital manouver, in this particular case from the parking orbit to the working orbit.
   Usually are offered multiple orbit injections, so the program calculates all the possibilities offered by the producers.
5. The results obtained are collected in a variable that goes trough two sorting analyses: the first is based on the ΔV and the second on the cost.
6. The launchers obtain a score based on the ranking in the sorting of ΔV and cost and a higher score is related with a lower value obtained.
7. Finally the program merges the results and defines which is the best launcher depending on this two parameters.

All the data are collected and saved.
To streamline the run time, QineniQ ubtroduced a costraint to limit the launchers displayed to a mazimum of 15 to be taken into account. That's the second part of the program.

3.2 Propulsion system evaluation

The need of study the launchers before considering the thrusters is related with the fact that the propulsion system has to store the propellant to reach the working orbit.
Another value of propellant to calculate is related with the consumption for station keeping and disposal. the program calculates the ΔV needed during all the orbital period including possible perturbations and eventual adjustments for debris avoid manouvers.
The evaluations are made in m/s to easily obtein the quantity of propellant with the Tsiolkovsky's equation.
From this point the program begins the iterative analysis for the propulsion system configuration, as shown in the Fig. 3.3. The value of ΔV obtained is converted in propellant required for each of the

thrusters within the database:

Thruster	Number of Models
Cold gas	7
Solid	5
Green prop.	7
Mono prop.	11
Bi prop.	6
Resistojet	9
Arcjet	1
PPT	2
MPD	1
Ion Thrust	7
HET	12
Colloid	2
Total	70

TABLE 3.1: Thrusters included in the database.

The main characteristics of these option are summarized following:

Type	Advantages	Disadvantages
Cold Gas	Simple Low system cost Reliable Safe	Extremely low Isp Moderate Impulse capability Low density High pressure
Mono propellant	Wide trust range Modulable Proven	Low Isp (mostly) toxic fuels
Bi-Propellant (storable)	Wide thrust range Modulable Proven	Complex Costly Heavy Toxic
Solid Propulsion	Simple Reliable Low cost	Usually one Thruster per burn Total Impulse fix Currently not qualified for long-term space application
Hybrid Propulsion	Simple Modulable Low cost Reliable	Lack of suitable oxidser for long-term missions
Electric Propulsion	very high I_{sp}	Low Thrust Complex Large maneuver time Power Consumption

TABLE 3.2: Principal characteristic of Spacecraft propulsion system

The quantity and the type of propellant generate the request to size tank, there are available multiple choices that are going to be better defined in 4. The program, if the solution of tank is available in the market, generates the basic propulsion subsystem and configuration. All the data for each thrusters plus the multiple choice propulsion system per launcher are saved and stored in the database. At this point a ranking procedure is defined following multiple parameters:

- The agility of the thruster
- Mass of the complete propulsion system
- Power required for working
- Propellant required
- Price
- Interface

This multiple ranking are merged into an unified system using an Analityc Hierarchy Process that traduce the system in the most real possible. The final results are displayed in a bar chart as the best propulsion system option for the mission scenario proposed.
The last procedure is to join the launcher options with the propulsion options to finally have the results of the study, the propulsion system and launcher that best match our requirements.

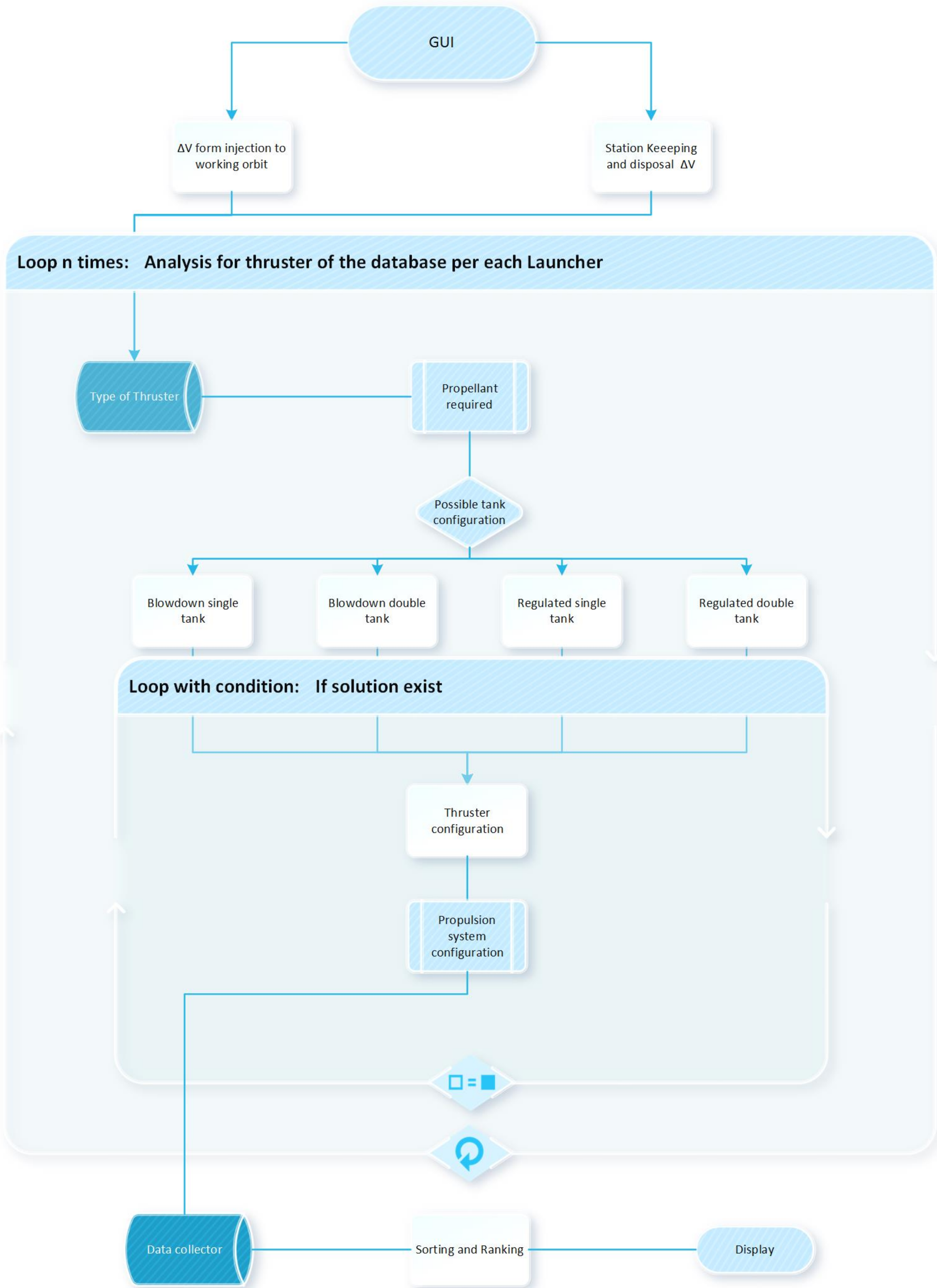


FIGURE 3.3: Flow graph of thruster selection

Chapter 4

Detailed program design

This chapter will focus on: technology consideration, assumption, constrains and decision taken in the process of program developing.

4.1 Launchers configurations

For any satellite customer, the key to the "mission success" starts with the launch phase, one of the most important and sensitive phase in the whole development chain. The launch system consists of a launch vehicle incorporating one or more stages as well as the infrastructure for ground support.
It places the payload into the desired orbit with a functional spacecraft altitude. In this chapter, the term "payload" includes all the hardware above the launch-vehicle-to-spacecraft interface, excluding the payload's protective fairing.

FIGURE 4.1: Launcher comparison by lift to LEO

The first step in the launch system selection process is to establish the mission needs and objectives. Since the manufacturers dictate the performance, the trajectory, and the family of vehicles which can operate from suitable sites, the launch platforms can be roughly subdivided as follows:

Launch platform

- Land: spaceport or fixed missile silo;
- Sea: fixed platform, mobile platform or submarine;
- Air: aircraft or balloon.

Size of rocket

- Sounding rocket : for sub-orbital spaceflight;
- Small lift : capable of lifting 2,000 kg of payload into LEO;
- Medium lift : capable of lifting between 2,000 to 20,000 kg of payload into LEO;
- Heavy lift : capable of lifting between 20,000 to 50,000 kg of payload into LEO;
- Super-heavy lift : capable of lifting more than 50,000 kg of payload into LEO.

The rockets considered for the study are only launched form the land platform. The sounding rockets have been excluded from this program due to QinetiQ is not interested in the developing of Sub-orbital S/Cs.

4.1.1 Payload configuration

The space market, in order to satisfy the needs of the customers, has developed different kinds of satellite, that can be divided by size as show in tab.4.1.

TABLE 4.1: Nomenclature for satellite

Group name	Average wet mass(kg)	Example
Large satellite	greater than 1000	-
Medium satellite	500 to 1000	GMP
Mini satellite	100 to 500	MiniSat 400
Micro satellite	10 to 100	MiniSat 100
Nano satellite	1 to 10	SNAP
Pico satellite	0.1 to 1	cubeSAT
Femto satellite	less than 100g	WikiSat

The small satellite segment of the satellite launch industry has been growing rapidly in the recent years. Development activity has

been particularly high in the 1–50 kg size range.
In the 1–50 kg range alone, fewer than 15 satellites have been launched annually in 2000 to 2005, 34 in 2006, while fewer than 30 launches annually during 2007 to 2011. This rose to 34 launched in 2012, and 92 small satellites launched in 2013[4].
Many launcher providers tried to follow the market trend introducing new launcher services:

4.1.1.1 Main Payload

The main payload in a launcher occupies all the available volume and mass provided for the S/C accommodation.

FIGURE 4.2: LISA PATHFINDER primary payload

4.1.1.2 Dual Payload

To decrease the launch cost it's possible to share the cargo bay with another mission. The two payloads have to be small and light enough to respect the transport requirements. In such case the launcher will reach the injection orbit only for one (primary configuration) of the two payloads. From there, the one in secondary configuration has to reach its working orbit. The secondary payload detaches from the launcher in order to proceed to the working orbit.
Some of the adapters taken into account are : SYLDA 5 for ARIANE (fig.4.3), ESPA for ATLAS , N.N. for PSLV and VESPA for VEGA, the mechanical drawings used are vision able in Appendix.A. Mechanical drawing has been used to define the dimensions, weigh allowed for the transport, while the average price was established using data of QinetiQ.

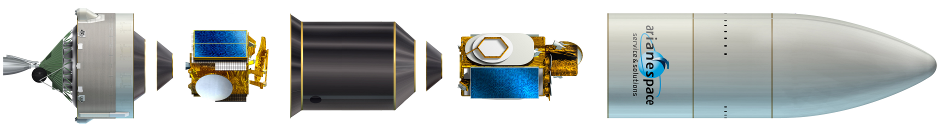


FIGURE 4.3: Dual payload with SYLDA 5

4.1.1.3 Hosted payload

Also known as piggyback payload [1],it uses the excessive launch capability of the rocket to launch small satellites.
The hosted portion of the satellite operates independently from the main spacecraft, but it shares the satellite's power supply, the transponders, and in some cases, the ground systems.
The concept of a hosted payload was developed in order to enable government organizations to use commercial satellite platforms to save costs and create a more distributed architecture for space assets. Nowadays private companies or universities profit by it.

Choosing to piggyback a hosted payload on a commercial satellite has many benefits:

- Shorter time to space. As the development of an entire satellite system is not required, a hosted payload on a commercial satellite can reach space in a fraction of the time that it would take to develop a free flyer program. Roughly 20 commercial satellites are launched to GEO orbit each year and each one presents an opportunity to add on additional capability.
- Lower cost. Placing a hosted payload on a commercial satellite costs a fraction of the amount of building, launching and operating an entire satellite. Cost reductions can result from shared integration, launch and operations with the host satellite.
- A more resilient architecture. Hosted payloads enable a more resilient space architecture by distributing assets over multiple platforms and locations. Rather than creating a single platform with multiple capabilities that could be a target for adversaries, spreading capabilities over multiple locations has the potential to contribute to a more resilient space architecture.
- Increased access to space. Roughly 20 commercial launches each year provides multiple opportunities for access to multiple orbit locations during the year.
- Operational options. Hosted payloads have multiple options to use existing satellite operations facilities with shared command and control of the hosted payload through the host satellite, or a completely dedicated and separate system operated by the hosted payload owner.

The adapter taken into account to define the constraints of weight and dimension,in A are: ASAP 5 for ARIANE 5 and Soyouz(fig.4.4),IPC or ESPA for ATLAS , ESPA for FALCON and DELTA II , N.N. for PSLV and VESPA for VEGA.
They are dividend in two classes:

[1]Data form satellite industry alliance to increase awareness of the benefits of hosted payload http://www.hostedpayloadalliance.org/

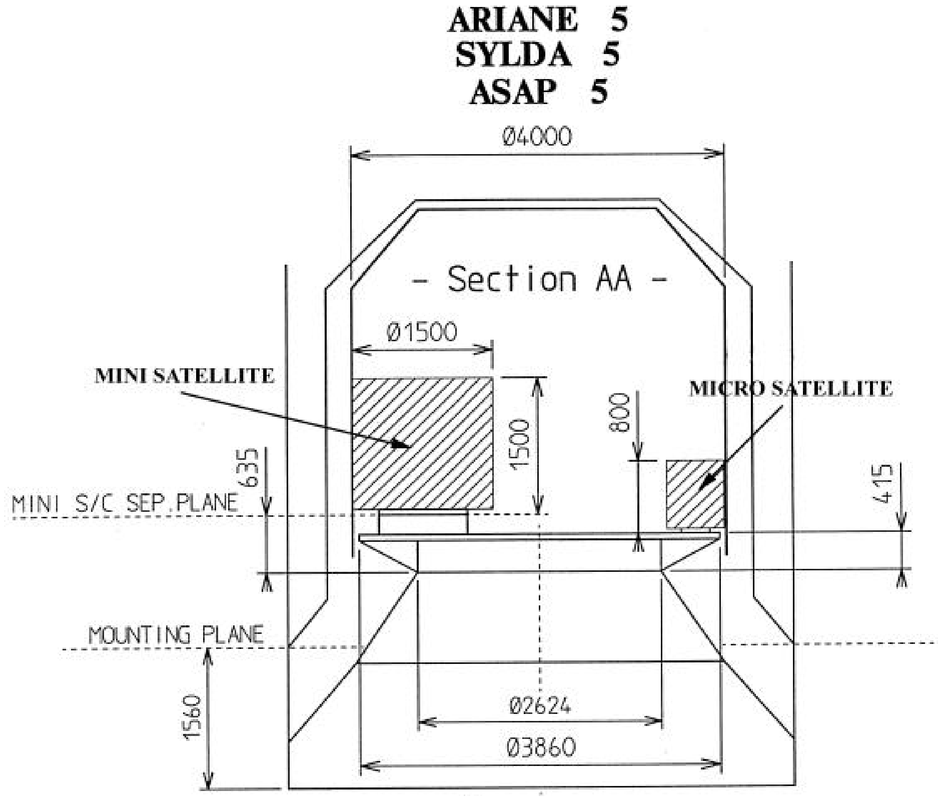


FIGURE 4.4: Piggyback configuration for ASAP 5

FIGURE 4.5: ASAP 5 assembly

Mini payload with an average mass between 100 and 500, as shown in Tab.4.1, an example can be taken from the auxiliary Payload configuration of ASAP 5 for ARIANE 5, where the mini auxiliary payload has mass range from 120 kg to 300 kg, and dimensions of 1.5 m of radius and 1.5 m of high. On the contrary the ESPA configuration available in the US market can carry up to 180 kg with dimension of 700*700*1115 mm. All the possible configurations with differences of weight and limits are saved in the database and summoned to define the correct configuration.

Micro Payload the micro satellite mass fluctuates between 10 kg, for cubesats, and 120 kg, which is the maximum allowed mass by

ARIANE 5 that's still classified as a Micro payload. The dimensions change respectively with the adapter whose a generic example could be: 600mmX600mm cross section, 700mm height on ASAP 5,adapeter for ARIANE 5, fig.4.5. Smaller satellite are adapted to be fit in this space.

4.2 Propulsion system

As mentioned before, the launcher is responsible for placing the payload in orbit, after which a first maneuver can be performed by either the spacecraft itself or any-kind of propulsion module specifically designed to modify the orbit to transfer the payload to its working orbit. Consequently, the orbit is maintained until the satellite's disposal, performing small maneuvers of station keeping.
The program first analyses which different propulsion technologies can be used for the orbit transfers defined by the user, the type and quantity of thruster included are shown in tab.4.2 .

TABLE 4.2: Thrusters included in the database.

Thruster	Number of Models
Cold gas	7
Solid	5
Green prop.	7
Mono prop.	11
Bi prop.	6
Resistojet	9
Arcjet	1
PPT	2
MPD	1
Ion Thrust	7
HET	12
Colloid	2
Total	70

All the thruster included have a TRL higher than 7. The table 4.3 contains the main characteristics of each thruster, depending on the type of propellant used.

4.2.1 Feed system

Depending on the type of thruster selected during the iterative study, the program defines a feed system design, suitable with the requirements of the type of the thruster. Tab. 4.3 shows what kind of propellant has been considered for the particular thruster and the technology of tanks related. The procedure adopted in the program is

TABLE 4.3: Propulsion system thrusters

Type	Propellant	Energy	Vacuum I_sp(sec)	Thrust range(N)
Cold Gas	N_2,NH_3, Freon, Helium	High pressure	50-75	0.05 - 200
Solid motor		Chemical	280-300	50-5x10^6
Chemical liquid:				
Monopropellant	N_2O_2,N_2;H_4	Exothermic decomposition	150-225	0.1-100
Green propelant		Chemical	150-225	0.1-100
Bipropellant	O_2 and RP-1	Chemical	350	50-5x10^6
	O_2 and H_2	Chemical	450	50-5x10^6
	N_2O_4 and MMH	Chemical	300-340	50-5x10^6
	F_2 and N_2H_4	Chemical	425	50-5x10^6
Dual Mode	N_2O_4 / N_2H_4	Chemical	330	
Electrothermal:				
Reistojet	N_2,NH_3, N_2H_4,H_2	resistive Heating	150-700	0.005-0.5
Archjet	N_2, N_2H_4,H_2	electric arc Heating	450-1500	0.05-5
Electrostatic:				
Ion	Hg/Ar/Xe/Cs	Electrostatic	2000-6000	10-6-0.1
Colloid	Glycerine	Electrostatic	1200	10-6-0.01
Hall Effect Thrusters	Xenon	Electrostatic	1500-2500	10-6-0.02
Electromagnetic:				
MPD (magnetoplasmatic)	Argon	Magnetic	2000	25-200
PPT(pulsed plasma)	Teflon	Magnetic	1500	5x10^-6-0.005
Pulsed inductive	Argon	Magnetic	4000	0.5-50

to convert the ΔV of all the missions with the Tsiolkovsky's equations, depending on the I_{sp} of the thruster selected. We obtain the overal quantity of propellant needed. The Kind of propellant is fundamental to know the mass that has to been carried by the S/C for the mission: this procedure is done converting the ΔV of all the missions with Tsiolkovsky. Knowing the quantity of propellant, the NIST Chemistry WebBook [24] gives all the average characteristics of the propellant according to the density. For the price and for the general overview, the references inside the TUDelft website [21] have been particulary usefull.

Defined the mass and volume of the propellant, supposing in a first approssimation that the ambient temperature is constant and lower than the hydrizine boiling point and lower than the helium melting point, depending on the type of thruster, the system searches the kind of tank available following the costrains of the tab.**??**..

Type	Propellant	Blowdown	Regulated	Pressurized tank
Cold Gas	N_2,Xe,, Helium	X	X	1 or 2 tanks solution
Solid motor		X	X	X
Monopropellant	N_2H_4	1 or 2 tanks solution	1 or 2 tanks solution	X
Green propelant		1 or 2 tanks solution	1 or 2 tanks solution	X
Bipropellant	$NTO\&MMH$	X	1 or 2 tanks solution	X
Reistojet	N_2	1 or 2 tanks solution	X	X
Archjet	N_2	1 or 2 tanks solution	X	X
Ion	Xenon	X	X	1 or 2 tanks solution
Colloid	Glycerine	X	X	1 or 2 tanks solution
Hall Effect Thrusters	Xenon	X	X	1 or 2 tanks solution
MPD (magnetoplasmatic)	Argon	X	X	1 or 2 tanks solution
PPT(pulsed plasma)	Teflon	X	X	1 or 2 tanks solution
Pulsed inductive	Argon	X	X	1 or 2 tanks solution

TABLE 4.4: Tank feed system to propellant system

Where X stands for: tank technology is not applicable for the type of thruster.

The program considers that the tanks are spherical made out of a single material, as seen in the AppendixB, while usually they're not.

Blowdown

This system of tanks, see Fig.4.6 on the right, contains propellant and pressurant with the same volume, which are divided by the use of surface tension or a diaphragm. The advantages of a blowdown system are:

- It's the simplest method and hence more reliable compared to others;
- It's less expensive because of fewer components.

The disadvantages are:

- Pressure, thrust, and propellant flow rate vary as a function of time;
- I_{sp} is a second-order function of chamber pressure and drops as a function of time.

Regulated

This system, see Fig.4.6 on the left, controls the pressure in the propellant tanks at a pre-defined pressure. The pressurant is stored at a high pressure, so the thrust does not vary during propellant consumption.
For bi-propellant systems, the regulation is essential in order to keep the propellant at the correct mixture ratio.
A disadvantage of this system is the increased mass of the total system is due to the increased quantity of components, which is inextricably related to an increase in cost.

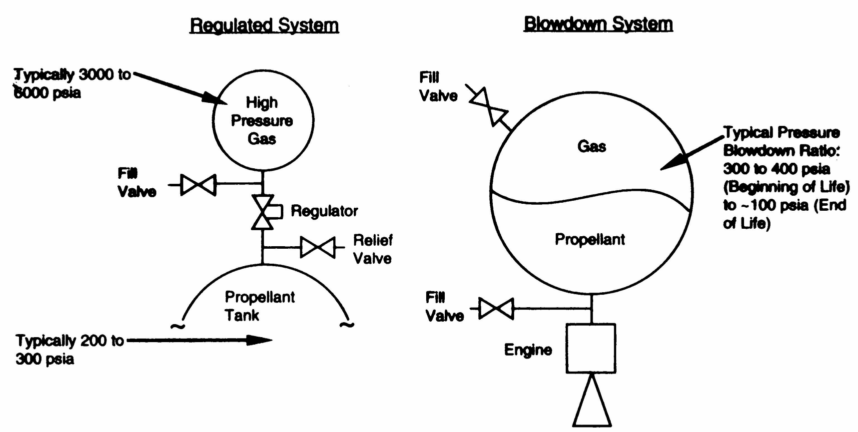


FIGURE 4.6: Pressurization systems [Larson]

Pressureized tank

This system is principally used for cold gas and EP systems. For cold gas, the analysis is made with the Nitrogen, while principally the Xenon is used for EP systems.

This analysis is iterated for all thrusters in the database per selected launcher. The equations used for the analysis are extensively explained in Appendix B.
The program doesn't indlude in the calculations any thermal shielding or heating of the tanks, before they're defined and designed. That will be part of a secondary study of the mission.

Feed components

To estimate the number of interconnections between the propulsion system and the ADPMS (Advanced Data & Power Management System), the program takes into account components as: the latch valve for each tank selected in the solution, the high pressure transducer for each tank, and the low pressure transducer or High pressure transducer for the thruster depending on the thecnology requirements. The pressure transducer for the Thruster, in case of Electric thruster, is excluded to select the appropriate PPU and FCU. Due to the lack of data about the power consumption of the PPU, its value has been established with a weighted average.

Margin philosophy

To minimize the risk of failure in the delivery of the payload to its required orbit within specified tolerances, a statistically determined performance margin needs to be reserved and controlled during the design phase
The margin philosophy adopted for evaluating the necessary mass, considers 3% of trapped propellant in the feed system and a loading error of 0.5%.
The margin philosophy related with the propellant for the procedure of station keeping is the 100% and for the disposal a 5% margin. All the margin adopted are based on requests of the QinetiQ propulsion compartment.

4.2.2 Thruster configuration

Mono system It collects all those propulsion configurations that have only one technology of thruster for the whole mission. The definition doens't restrict the number of thrusters. Since the thrusters have also to provide to the station keeping, I adopted some configurations proposed by QinetiQ: 4, 4+4, 12 and 12+12.
All the configurations are saved and analyzed selecting the optimal between weight and agility.

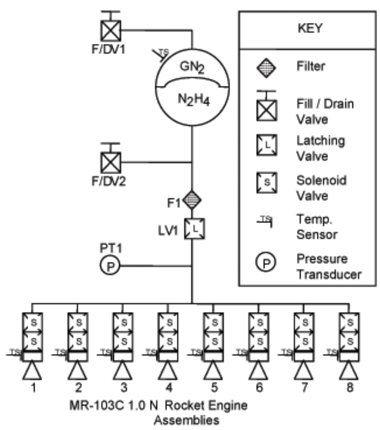


FIGURE 4.7: Typical mono-propellant configuration

the EP thrusters, due to their structure, don't follow this configuration; the details are explained in the section 4.2.3.

Hybrid system, The program assesses all the possible combinations made with an apogee motor that generates a thrust more than 20N for the orbit transfers, as well as the other thrusters for the orbit maintenance and the de-orbiting.
The value of 20N has been established inside QinetiQ, due also the lack of knowledge to use thruster more than 20 N for operation of station keeping. Apogee thrusters are: mono-propellant, green propellant and bi-propellant, instead the secondary propulsion for station keeping are all the possible combination available in the market:

1. Cold gas, the ΔV required for station keeping and de-orbit is provided by a cold gas system, with 4, 8, 12 or 24 thrusters;
2. Own propellant, Fig.4.8, the ΔV required for station keeping and de-orbit is generated with the same propellant used for the transfer orbit. This solution have a main apogee motor and thrusters for station keeping with the same technology.
3. Electric Propulsion, this particular configuration requires all of the components present in section 4.2.3 when using 1, 2, 4 or 8 electric thrusters.

4.2.3 Electric propulsion design

For Low thrust propulsion, the code recognizing the request exclude all tanks with regulated system for propulsion using particular blow down tanks that have been already develop for Xenon. Two main feed components are introduced : PPU and FCU.
These two components are fundamental for EP, yet they will also increase the maximum mass an power required from the Propulsion subsystem.
The PPU, is a power regulator for Electric Thrusters, which provides the high and low voltage supplies as well as the associated current and voltage telemetry.
The FCU, this technology provides precise control of the propellant flow rates to electric thrusters.
Both PPU and FCU are scaled from QinetiQ and Sitael products, due to the lack of information of the other producers. A detailed design of the Electric propulsion system is momentarily put aside until a more specific request is inquired.

4.3 Orbital mechanics

The primary objective of the propulsion system is to provide the necessary thrust in order to reach the chosen orbit. These requirements

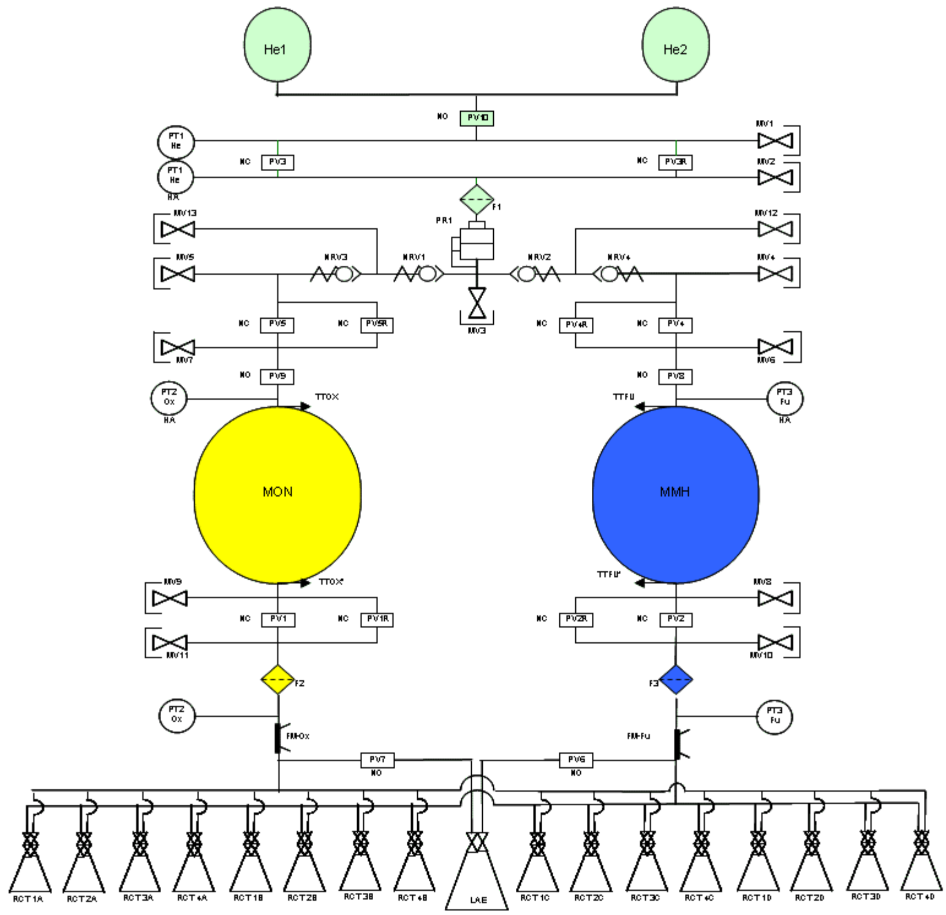


FIGURE 4.8: alphabus propulsion schematic configuration

are strictly influenced by the thrusters typology or configuration, as seen in the previous section.

4.3.0.1 Orbit injection

For every selected launcher in the system, the code analyses the transfer needed from : LEO, SSO, GTO and GEO in order to reach the chosen orbit, Fig.4.10.

The Class that provides this verification is cLaunch, which starts from the **function *deltaToEnd*** and where all of the possible transfers are evaluated in order to define the respective ΔV required. The system will analyze different trajectories depending on the type of propellant used.

4.3.0.2 Transfer method

This preliminary study considers multiple impulse transfers. The developed function starts from a simple Homman transfer and, if the orbits are co-planar, it will go to multiple impulse variations of

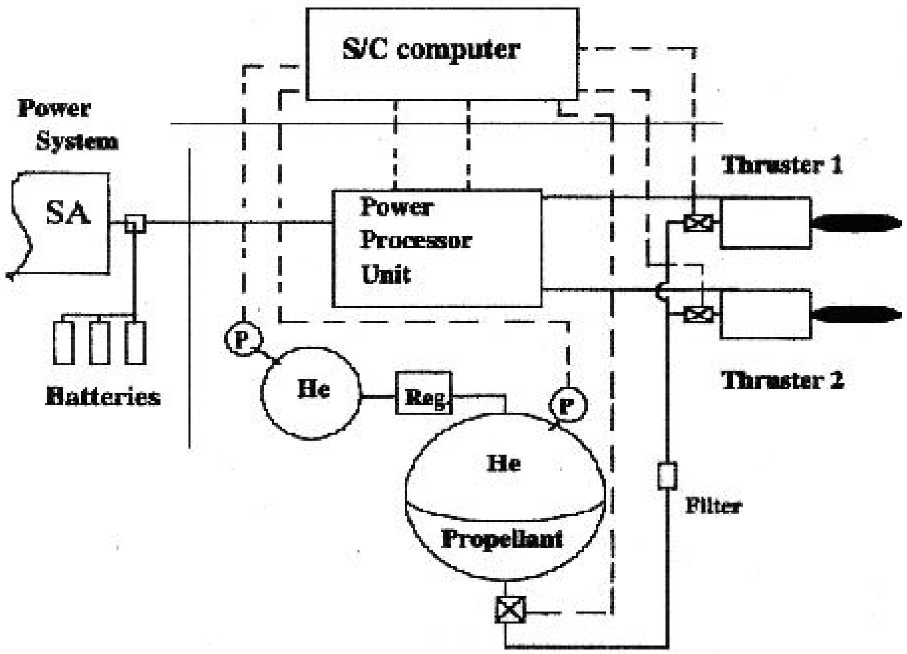


FIGURE 4.9: Schematic of electric propulsion system with separate power source/system [26]

the orbit.
It will first reach the required altitude and eccentricity, subsequently the inclination will change, which is followed by an adjustment of the RAAN and the ω.
A single impulse reduces the chance of losses and the time to achieve the result.
Though for a wide range of solutions, it is easier to analyze every single impulse required by the transfer.

Chemical This part of the code uses the classical equation of space-flight mechanics to calculate the ΔV required to reach the final orbit. One of the most common transfers used are the Hohmann manoeuvres, Fig.4.11, due to the assumption of circular injection of the launcher for LEO, SSO and GEO orbit. Other minor adjustment equations[11] are evaluated and calculated, excluding possible perturbation errors due to misalignments of the thrusters and COG is considered to be perfectly aligned.

Electrical The transfer with electrical thrusters takes another type of analysis. This kind of propulsion system requires more time due to the low generated thrust. This results in a spiral manoeuvre which is equivalent to the Hohmann transfer for the chemical thrusters.
Assuming that a spiral transfer is used to transfer a satellite from LEO to GEO, the ΔV and the transfer time will need to be computed.

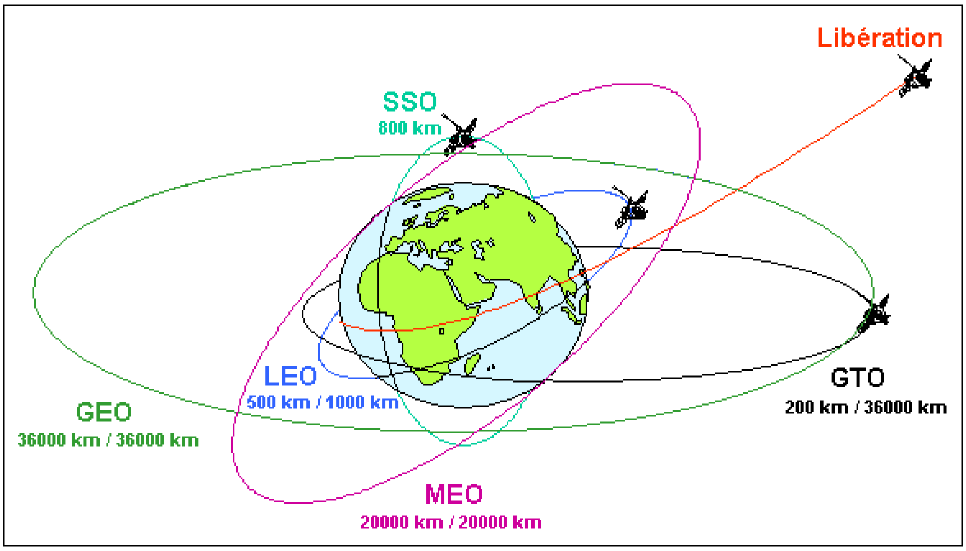


FIGURE 4.10: Typology of trajectory,CNES, Direction des lanceurs, Christophe Bonnal

If the initial and final orbits are at different inclinations, the value β and the out-of-plane thrust angle need to be determined, in order to reach at the desired orbit with the correct inclination.
The initial out-of-plane angle is called β_0 and can be calculated as:

$$\tan(\beta_0) = \frac{\sin(\frac{\pi}{2}\Delta i)}{\frac{V_0}{V_f} - cos(\frac{\pi}{2}\Delta i)} \tag{4.1}$$

In this case, β_0 is the thrust vector of the yaw angle, Δi, the orbital inclination change and V_0 is the initial circular velocity, which is calculated with:

$$V_0 = \sqrt{\frac{\mu}{r_0}} \tag{4.2}$$

Where r_0 is the radius of the initial orbit, and V_f the final orbit velocity. Now the ΔV can be calculated as follows:

$$\Delta V = V_0 cos(\beta_0) - \frac{V_0 * sin(\beta_0)}{\tan(\frac{\pi}{2}\Delta i + \beta_0)} \tag{4.3}$$

It is also possible to define the velocity during the manoeuvre itself, as well as the inclination change over time, though this depends on the vehicle acceleration. However, for this first approximation, the ΔV required was enough.
Simultaneous eccentricity and inclination changes with EP are best accomplished by having an in-plane acceleration perpendicular to the major axis in order to obtain:

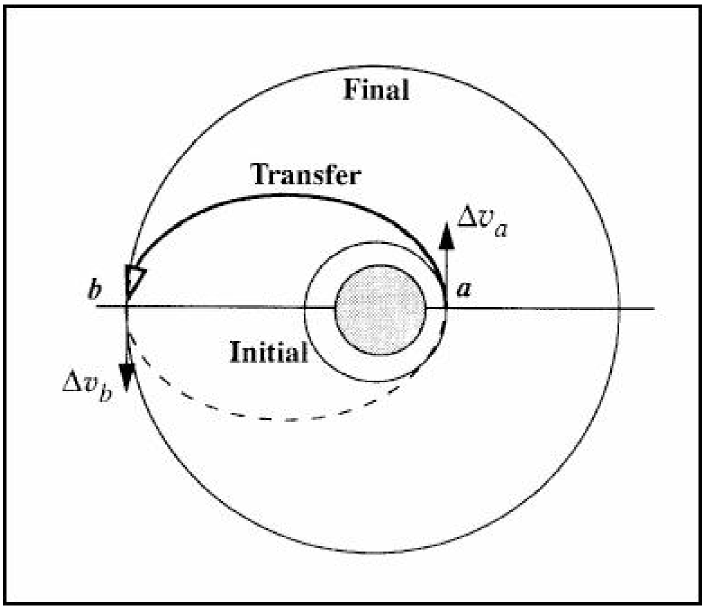


FIGURE 4.11: Hohmann transfer orbit

$$\left|\frac{de}{dt}\right| = \frac{f\cos|\beta|}{\pi}\sqrt{\frac{a(1-e^2)}{\mu}}(3a+\cos\alpha\sin\alpha) \tag{4.4}$$

Where α is the arc of discontinuous thrusting, β is the out-of-plane thrust angle and f is the acceleration of the S/C. When the perigee and the apogee are centred around burns on each revolution, the ΔV obtained is :

$$\frac{d\Delta V}{dt} = \frac{2\alpha f}{\pi} \tag{4.5}$$

This is relevant to a maximum burn arc α= 90°. An integration of the Eq. 4.4 and 4.7 gives the following velocity increment:

$$\Delta V = \frac{2\alpha f\delta t}{\pi} = \sqrt{\frac{\mu}{a}}\frac{2\alpha|\arcsin e_1 - \arcsin e_2|}{cos|\beta|(3\alpha+\cos\alpha\sin\alpha)} \tag{4.6}$$

The limiting cases of impulsive small thrust α and continuous thrust, α = 90°, can be compared using the Eq.4.6, whose result is:

$$\Delta V_{cont} = 4/3\Delta V_{imp} \tag{4.7}$$

Hence, the low-thrust eccentricity change needs a 33% greater ΔV than the corresponding impulsive manoeuvre. The inclination change that occurs simultaneously with the change of eccentricity, assuming that β is the inclination, reverses at the minor axis crossings. The secular rate of change of *i* is:

$$\left|\frac{di}{dt}\right| = \frac{f\sin|\beta|}{\pi}\sqrt{\frac{a}{\mu}}|\cos\omega|\sin\alpha\frac{2(1+e^2)}{\sqrt{1-e^2}} \tag{4.8}$$

When combining Eq.4.4 and 4.8, it is possible to obtain the change rate of *i* with respect to *e*:

$$\left|\frac{di}{de}\right| = \frac{2\tan|\beta||\cos\omega|\sin\alpha}{3\alpha+\cos\alpha\sin\alpha}\frac{1+e^2}{1-e^2} \tag{4.9}$$

Also evaluated with EP, is the adjustment of the argument of perigee:

$$\Delta V = \sqrt{\frac{\mu}{a}}\frac{e}{\sqrt{1-e^2}}\frac{2\alpha|\Delta\omega|}{3\alpha-\cos\alpha\sin\alpha} \tag{4.10}$$

As can be seen, ΔV is independent from *f* and the acceleration of spacecraft. From the comparison of the limiting cases of impulsive thrust and continuous thrust for adjusting ω, the following relation was found :

$$\Delta V_{cont} = \frac{2}{3}\Delta V_{imp} \tag{4.11}$$

In this particular situation, the ΔV required for the low-thrust manoeuvre is smaller than for an impulsive manoeuvre.
From this result, the previous evaluation, the type of propulsion system and type of thrusters, the program derives the value of the propellant needed to reach the operational orbit.

4.3.1 Transfer time

A preliminary general evaluation of the Transfer time can be made using the Tsiolkovsky equation and Newton's second law . Now the result is integrated :

$$\Delta t = \frac{m_0}{F/I_{sp}g}\left(\exp\left(\frac{\Delta V}{I_{sp}g}\right)-1\right) \tag{4.12}$$

Where F stand for Thrust, g is the gravitational acceleration , m_0 the initial mass of the spacecraft, ΔV the required velocity variation for the manoeuvre and I_{sp} the specific impulse of the thruster. This allows for a rough first approximation of the required time. The Eq.4.12 is used in the table 4.5, where a comparison of transfer orbit from LEO to GEO with different thrusters is shown:

TABLE 4.5: Comparison of low-thrust and chemical propulsion transfer[12]

Transfer type	**Δ V (km/s)**	**Transfer time**	**Assumed I_{sp}(s)**	**Assumed Initial mass(kg)**	**Propellant mass(kg)**	**Total mass (kg)**
Low thrust	4.651	5.4 to 53.8 days	2000	500	133.8	633.8
Hohmann transfer	3.893	5.27 hours	420	500	786.1	1286.1

Chemical This kind of propulsion requires less flight time, yet comes at the expense of additional propellant mass that needs to be carried into space, which means less available mass for the payload.

Electrical The electric propulsion has significantly fewer propellant requirements, however they go hand-in-hand with a transfer time in the order of days instead of hours. This latter aspect is very important as it increases the difficulty of calculating the time-related effects on the transfer orbit.

4.3.2 Station keeping

For many Earth satellites the effects of the non-Keplerian forces, i.e. the deviations of the gravitational force of the Earth from that of a homogeneous sphere, gravitational forces from Sun/Moon, solar radiation pressure and air-drag, must be counteracted with a propulsion system and the results of this study are analyzed in this chapter.

4.3.2.1 Perturbations explanation

Once in their mission orbits, some missions may require corrective manoeuvres when perturbing forces change their trajectory in such a way that affects the payload functionality.
The effects of the non-Keplerian forces, like the Earth's oblateness gravity field, sun-moon gravitational forces, solar radiation pressure and aerodynamic drag, must be counteracted.

The Earth's gravity field, combined with the effect of the Sun/Moon, usually generates a perturbation in the orbital plan. The magnitude of this influence depends on the typology of the mission:

- For SSO, the precession of the orbital plane due to the Earth's oblateness is considerable. This could be incorporated as part of the mission design, though in general, the inclination change caused by Sun-Moon effects is always undesirable.
- For GEO, the main disturbing force is the Sun/Moon's gravitational influence which needs to be counteracted in order to avoid any inclination variations.
- For LEO, due to the low altitude and hence the higher density of atmospheric particles, the most relevant effect that needs to be compensated is the atmospheric drag, as this influences all the orbital parameters and instigates orbital decay.

The solar radiation pressure mostly affects the orbit's eccentricity. This and other occurring perturbations will be analysed in the subsequent chapters of this report. How the program calculates the required ΔV in order to compensate for these perturbations will be explained as well.

4.3.2.2 Atmospheric drag

The Aerodynamic drag is generally a prime source of disturbing torques and it generates a velocity reduction for the S/C in low Earth orbits. Drag forces can be estimated from the following classic relationship:

$$D = \frac{1}{2}\rho V^2 C_d A \tag{4.13}$$

Where D is the drag force in the opposite direction of the velocity, ρ is the atmosphere density, V the S/C velocity, C_d the drag coefficient and A is the area normal to the velocity vector. In this first approximation, the torque generated by the atmospheric drag is equal to zero due to the fact that the COG is equivalent to the center of pressure.
The effect of drag significantly influences all the orbits below 1000km. For the orbits below this level, the code utilizes the "COSPAR International Reference Atmosphere 2012", also know as CIRA2012. It contains four semi-empirical models of the Earth's atmosphere, whose code specifically utilizes the JB2008 Model, which currently reflects the best understanding of the atmospheric density.
The effect of the atmospheric drag reduces the satellite's orbital velocity, which causes the decrease of the orbital altitude. As the orbit shrinks, the orbital period decreases as well, causing the ground track to seemingly shift eastward. If the drift of the semi-major axis exceeds 5km, or if the ground track shifts more than 30 minutes w.r.t. the required position, the code will proceed to calculate the periodically manoeuvre required in order to maintain the desired orbit.
The effect of the atmospheric drag on the orbital velocity can be considered as a positive effect for the end-of-life phase of low orbit satellites as it decelerates the satellite and hence decreases the required ΔV to force re-entry.

4.3.2.3 Magnetic perturbation

The Earth's magnetic field interacts with any residual magnetic moment within the satellite, which generates a disturbing torque.
The Earth's magnetic field presides about the earth's axis but is very weak. This field is continuously fluctuating in direction and intensity because of magnetic storms and other influences. Since the field strength decreases with $1/r^3$, where r is the radial distance w.r.t. the Earth's centre and hence it is directly defined by the orbital altitude, it is acceptable to neglect the magnetic field forces in this preliminary design.

4.3.2.4 Solar radiation

This factor dominates at high altitudes (above 800 km) and is due to the impingement of solar photons on the satellite surfaces.
The solar radiation pressure causes periodic variations in all the orbital elements, providing an acceleration defined as:

$$a_r = -4.5e - 6(1 - r)\frac{A}{m} \tag{4.14}$$

Where A, in m^2, is the satellite's cross-section exposed to the Sun, m is the S/C mass in kg, and r is a reflection factor which has different values for different conditions :

- r = 0 for absorption;
- r = 1 for specular reflection;
- r = 0.4 for diffuse reflection.

Below 800 km altitude, the atmospheric drag perturbations are greater than those from the solar radiation pressure.

4.3.2.5 Secular perturbation

The gravitational torque in the spacecraft results from a variation in the gravitational force on the distributed mass of a S/C. The determination of this torque requires knowledge about the gravitational field and the distribution of the spacecraft mass.
This torque decreases as a function of the orbital radius and increases with the offset distances of masses within the spacecraft (including booms and appendages). These perturbations induce a rotation of the perigee.
The satellite's acceleration can be found by taking the gradient of the gravitational potential function Φ:

$$\Phi = \left(\frac{\mu}{r}\right)\left[1 - \sum_{n=2}^{\infty} J_n \left(\frac{R_e}{r}\right)^n P_n(\sin L)\right] \tag{4.15}$$

Where μ is the standard Earth gravitational parameter, R_e is the Earth's radius, P_n the Legendre polynomials, L is the geocentric latitude and J_n are the dimensionless geopotential coefficients whose the first are:

J_2 = 0.00108263
J_3 = -0.00000254
J_4 = -0.00000161

By using the Goddard Earth Model 10B, known as GEM10B, the dominant effects can be evaluated. These are secular variations in the right ascension of the ascending node and the argument of the perigee due to the Earth's oblateness, as represented by J_2:

$$\dot{\Omega}_{J2} = -1.5nJ_2 \left(\frac{R_e}{a}\right)^2 (\cos i)(1-e^2)^{-}2 \tag{4.16}$$

$$\dot{\omega}_{J2} = -0.75nJ_2 \left(\frac{R_e}{a}\right)^2 (4-5\sin^2 i)(1-e^2)^{-}2 \tag{4.17}$$

4.3.2.6 Debris avoidance

The collision avoidance manoeuvre or the Debris Avoidance Manoeuvre (DAM) are conducted by a spacecraft in order to avoid colliding with another object in orbit. The most commonly used programs in order to calculate the required ΔV to avoid all pieces of space debris, are MASTER and DRAMA. These are both programs developed by ESA[2].
MASTER (Meteoroid and Space Debris Terrestrial Environment Reference) is able to assess the debris or meteoroid flux imparted on a spacecraft operating in any arbitrary Earth orbit. The program also provides the necessary reference data for the DRAMA-program.
DRAMA (Debris Risk Assessment and Mitigation Analysis) is a comprehensive tool that performs a compliance analysis of a space mission with the space debris mitigation standards.

After comparing the results of the different systems and programs, they seemed to be quite comparable with a difference margin of 5% respect the DAM obtained with the ESA program. Whether it is necessary to directly call DRAMA as a root program for the final evaluation of ΔV for debris avoidance, it is dictated by a matter of running time.

4.3.3 End of life

Because of the ever-increasing presence of the orbital debris, which consists of decommissioned satellites and associated parts, it is quite essential that the new satellite designs should plan for de-orbiting at the end-of-life phase of a mission. Or alternatively, it should be removed from areas such as the geostationary ring, where they pose a serious threat to other spacecraft on any low-Earth orbit constellation.

4.3.3.1 Normative

In order to ensure a corporate approach on the space debris mitigation, the Agency's policy is that the ECSS-U-AS-10C is established as the ESA standard ("the standard") for technical requirements on space debris mitigation for Agency projects.

[2] https://sdup.esoc.esa.int/web/csdtf/home

4.3.3.2 Reentry

The code always chooses to de-orbit the LEO satellites. The required ΔV for de-orbiting a satellite in a circular orbit at initial altitude, H_i, and initial velocity, V_i is:

$$\Delta V deorbit = V_i \left[1 - \sqrt{\frac{2(R_e + H_e)}{2R_e + H_e + H_i}} \right] \tag{4.18}$$

R_e is the radius of Earth and H_e is the perigee altitude at the end of the burn. This latter one settles at 50km, though it can be adjusted depending on the requests of the user.

4.3.3.3 Graveyard orbit

Also called as junk orbit or disposal orbit, this is the orbit where the spacecrafts are intentionally placed at the end of their operational life.

For the satellites in geostationary orbit and geosynchronous orbits, the graveyard orbit is a few hundred kilometres above the operational orbit. A reliable attitude control is very important during this transfer manoeuvre.

According to the Inter-Agency Space Debris Coordination Committee [**IDAC**], the minimum perigee altitude, ΔH, above the geostationary orbit is:

$$\Delta H = 235 \text{ km} + \left(1000 C_R \frac{A}{m}\right) \text{ km} \tag{4.19}$$

Where C_R is the solar radiation pressure coefficient (typically between 1.2 and 1.5) and $\frac{A}{m}$ is the ratio of the aspect area A in $[m]$, over satellite mass in [kg].

4.3.4 Advanced modifier

After running a beta version of the program, some of the first bugs were fixed and new strong key point have been introduced on the structure of the program itself. During the research of a possible solution, with a simple click on the GUI 'Advanced button', the program opens a window shown in fig.4.12

The main function of this process is to modify the iteration made by the program and to follow the user preferences for the propulsion. The parameters to modify are:

ITAR. It controls the export and import of the defence-related articles and the services on the United States Munitions List (USML).

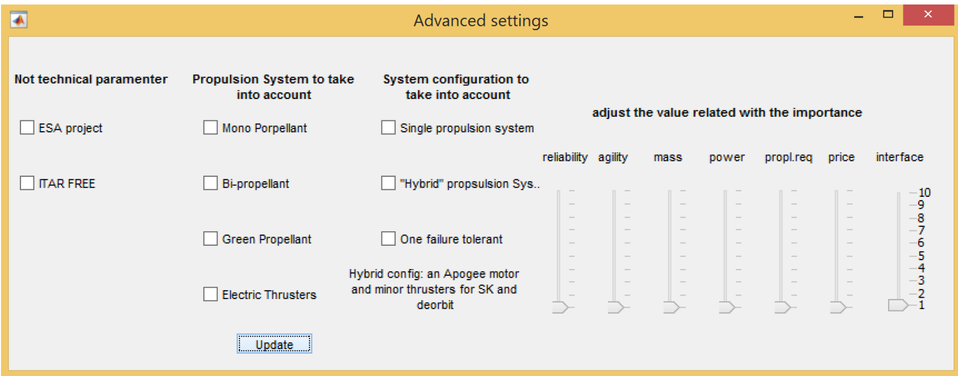


FIGURE 4.12: Advanced setting in GUI

This means that informations and materials pertaining to defence and military-related technologies (items listed on the U.S. Munitions List) may only be shared with the U.S. citizens unless an authorization from the Department of State is received or a special exemption is ushered. This has two implications in real life:

- Time and cost increase to obtain any U.S. technology due to the bureaucratic review;
- The Launcher selection is influenced as mentioned in Part-126 [17] : "foreign launch vehicle or for use on a foreign launch vehicle or satellite that is to be launched from a foreign country shall be considered a permanent export."

This implies that there is no possibility for state-use of a launcher with the current weapons embargo of the U.S. which results in the exclusion from the selection for the launchers from People's Republic of China (PRC), Ukraine and Republic of India. Selecting ITAR free the program automatically deselects from the study all the launchers, thrusters and components subject to ITAR.

ESA Project. When an industry works on a ESA project, the launchers built within the UE are prefered, so there is the possibility to give them a higher score. [3],

- maintain the competitiveness and affordability of Ariane and Vega launchers;
- maintain the ground infrastructure necessary for launches;
- foster the creation of a European institutional market for Ariane and Vega;

[3]Launcher strategy: `http://www.esa.int/Our_Activities/Launchers/Launcher_strategy`

- ensure that Europe can respond to evolving market demands by continuing the improvement of Ariane and Vega and facilitating the use of Europe's Spaceport by the Russian Soyuz rocket;
- support European industry, technology and research capabilities by improving industrial competitiveness, promoting innovation and creating employment;
- develop the next generation of launchers as well as the related ground infrastructures to better serve the institutional and commercial market;
- encourage international cooperation and play a leading role in future developments.

Propulsion system to take into account The selection of technology that the program is going to evaluate during whole the process of analysis, this choice is introduced to eliminate the study of all the system if the designer ha a particular Technology orientation due to obtain the preliminary study in the field required.

system configuration to take into account The selection show the possibility of choosing only one typology of propulsion or a mix between different technology usyng the idea of Hybrid trusther before definer Meanwhile the box one failure tolerant enable inside the program a complete hardware redundancy with all the consequence that metter with the problems of weight .

Reliability Accoriding to its definition [14], it describes the ability of a system or a component to function under state conditions for a specified moment or within a certain time interval. It is theoretically defined as the probability of success. If the propulsion system is not able to work as intended or perform its function within the requested period of time, the system experiences a failure.
The propulsion system has a large impact on the overall reliability of the entire vehicle. Its failure could jeopardize the success of the entire mission.
Different propulsion systems have different levels of reliability, due to the different complexity of each system. Because of the lack of data regarding the probability of an engine failure per hour, the code is adapted to utilize the thrusters and components in order define its reliability. For example, if the system has 8 thrusters instead of 4, it gains more points for its reliability score, because then the system is regarded as a 4 + 4 thruster system, which makes it fully redundant should one entire thruster branch fail.

Agility We define agility as the time needed by the system in order to provide a ΔV of 1 m/s. The worst-case scenario is when the satellite has more mass, m, hence more thrust is required to change its moment of inertia. This can be done with the known I_{sp} and maximum nominal thrust, T, generated by the thrusters. From the rocket equation related to the steady state mass flow:

$$\dot{m} = \frac{Thr}{I_{sp} g_0} \tag{4.20}$$

The agility can be obtained as:

$$agility = \frac{m * g * I_{sp}}{T} \left(1 - \frac{1}{\exp \frac{1}{g * I_{sp}}} \right) \tag{4.21}$$

For a more realistic result the response of the system takes into account the time that passes between ushering the firing command and the actual firing itself. Because of this delay, it is possible to subdivide the thrusters into three categories:

- **Cold start**, the thrusters do not require any pre-heating in order to obtain full or partial efficiency.
- **Warm start**, Arcjet, Resistojet and also LPM-103S green propellant thrusters could not work if they are not partially heated.
- **Hot start**, a system of this typology fully requires to be heated before use.

Mass Mass is a fundamental parameter in space, since the satellite's mass is strictly related to cost. The mass you can carry when acquiring a launcher is limited, hence it is important to look for the lightest possible propulsion system which results in larger available mass margins for other uses, e.g. the payload.
Reducing the mass of the whole propulsion system can be done by either reducing the propellant mass or by selecting a lighter feed System. In order to reduce the propellant mass, EP are usually taken into consideration. However, this kind of propulsion is not always the best. The main disadvantages are :

- there are more difficulties on constructing the feed system that needs to provide an ad hoc solution as PPU and as feed regulator;
- this also implies more mass added to the feed system;
- necessary manoeuvring time is obviously longer compared to other propulsion systems.

Reducing the feed system can be done by using new carbon fiber tanks instead of titanium tanks. The heritage of these new fiber tanks is much smaller compared to the one of titanium tanks.
And so, the mass will always be a compromise of several factors which needs to be optimized in order to find the best solution for the mission.

Power The power expresses the Watts needs of the propulsion system. The collected data represent the power peaks that can be generated by all the different propulsion systems during thrusting.
The power can be considered as a requirement that influences the choice of the propulsion system. An example is shown in table4.6; EP usually has a higher I_{sp}, in the order of thousands instead of hundreds, that allows for a reduction in the required quantity of propellant, though this comes at a cost of higher power requirements.
With higher power requirements comes the need for more power production on-board. This need can be satisfied by using more solar panels, batteries and cabling, which will increase the total mass of the power subsystem. In some cases, the increased mass of the power system, in order to satisfy the higher power requirements, summed with the lesser amount of propellant mass, could render a total mass that exceeds the one of a chemical propulsion system.

TABLE 4.6: Power/Thrust comparison for 0.5 N thruster

Thruster name	**Type or propulsion**	I_{sp}**(s)**	**Power(W)**
HPGP	green propellant	225	8
BHT-8000	HET	1880	4000

Propellant required The mass-aspect of propellant is of relevant importance for the entire system. The possibility to select the importance of the quantity of propellant loaded into the S/C, directly influences the choice of the propulsion system.

Price Though the price always needs to be considered when choosing a propulsion system, there is a general rule that the thrusters with a higher flight-heritage have a cheaper price-tag compared to newer types of propulsion. The prices are always represented in the order of dozens of k€. New prototypes or small high-tech products like EP tend boast prices up to the order of hundreds of k€.
This study only analyzes the cost concerning the propellant, the feed system and the thrusters selected as final solution. In reality, these

components only constitute a small part that needs to be incorporated in the overall Cost Analysis.

Interface Each component is related with electrically powered unit, connectors and cabling. The more cables a system has, the more time and difficulties will go into the integration and verification of the system. Hence the interface can be seen as a reference for how easily the system can be assembled in the last phase of the production process. This term should not be confused with the level of complexity of the entire propulsion system, e.g. a mono-propellant system can have more cabling than an EP, though this does not mean the latter one is a less complex system.

4.3.5 Hierarchy Process

The Analytic Hierarchy Process (AHP)[29] is an effective tool to deal with complex decisions, and it may aid the engineer to set up priorities and make the best decision. By reducing complex decisions to a series of pairwise comparisons, and subsequently synthesizing the results, the AHP helps to capture both subjective and objective aspects of a decision.
In addition, the AHP incorporates a useful technique for checking the consistency of the engineer's evaluations, hence reducing the bias in the decision making process.

The AHP considers a set of evaluation criteria, as well as a set of alternative options which contains a certain best decision. It is important to note that, since some of the criteria could be counteracting one another, it is not generally true that the best option is the one which optimizes each single criterion. The best option is rather a compromise which achieves the most suitable trade-off among the different criteria.
The AHP generates a weight for each evaluation criterion according to the engineer's pairwise comparisons of the criteria. The higher the weight, the more important the corresponding criterion is. Next, for a fixed criterion, the AHP assigns a score to each option according to the engineer's pairwise comparisons of the options based on that criterion.
The higher the score, the higher the performance of a particular option corresponding to the considered criterion. Finally, the AHP combines the criteria weights and the options scores, thus determining a global score for each option, resulting in a final ranking for all the options. The global score for a given option is a weighted sum of the scores it obtained w.r.t. all the criteria.

The AHP is a very flexible and powerful tool because of the scoring system and the final ranking, obtained on the basis of the pairwise relative evaluations of both the criteria and the options provided by the user. The computations made by the AHP are always guided by the engineer's experience. The AHP can thus be considered as a tool able to translate the evaluations (both qualitative and quantitative) made by the engineer into a multi-criteria ranking.

On one hand, the AHP is simple because there is no need to build a complex expert system with the engineer's knowledge embedded in it.

On the other hand, the AHP may require a large number of evaluations by the user, especially for problems with many criteria and options. Although every single evaluation is quite simple, since it only requires the engineer to express how two options or criteria compare to one another, the load of the entire evaluation task may become unreasonable.

In fact the number of pairwise comparisons grows quadratically with the number of criteria and options. For instance, when comparing 10 alternatives on 4 criteria, $4 \cdot 3/2=6$ comparisons are requested to build the weight vector, and $4 \cdot (10 \cdot 9/2)=180$ pairwise comparisons are needed to build the score matrix.

However, in order to reduce the engineer's workload the AHP can be completely or partially automated by specifying suitable thresholds to decide some pairwise comparisons autonomously.

Fundamentally, the AHP works by developing weights, also called priorities, for alternatives and the criteria are used in terms of their importance in order to achieve the goal. These priorities are derived from pair-wise assessments using judgements, or ratios of measurements from a certain scale, if one exists.

Hence a weighted trade-off using AHP is performed in three steps:

1. Develop the weights (or priorities) for the criteria
2. Develop the scores (or ratings) for each decision alternative
3. Calculate the weighted average score (or rating) for each decision alternative

Chapter 5

Validation and reliability

After defining the structure and the results attended by the program it is necessary to define the level of reliability of its results.
A complete mass breakout of the propulsion system can be difficult to find and even if when these data are found, they're taken from systems that violate the assumptions used in the program.
Many propulsion systems are composed by multiple thrusters even if a single thruster would be able to provide all the necessary thrust; the additional thrusters are added for redundancy purposes or for attitude control.
Some spacecrafts end their missions without using all their propellant, but such a choice is done for security purpose.
All these conditions, by schedule, are indicative to the fact that there are many factors in a propulsion system design that this program doesn't take into account.
Quite simply, the program predicts what the system could be made of, but not necessarily what it will be in reality.
All this issues have been taken into account and followed by a long validation procedure, trough all the different classes of the program.

5.1 Missions comparison

The validation procedure is done comparing the results displayed by the program with the parameters of the missions already developed.

SAOCOM-CS This system compares a mission for which QinetiQ space NV is competing for the construction-process.
SAOCOM-CS is the result of the collaboration of the Argentinian Space Agency and the ESA. Its name is the acronym of **SAR Observation & Communications Satellite - Companion Satellite**.
It is defined as a Companion Satellite because it has to be launched and fly together with the Argentinian SAOCOM 1B[7] satellite. SAOCOM-CS is a receive-only satellite. The two satellites would fly in formation, with SAOCOM acting as an illuminator, enabling the first time single-pass SAR interferometry at L-band.
Key parameters of the architecture for the SAOCOM-CS, which were

defined at QinetiQ, are represented in Table 5.1:

TABLE 5.1: QinetiQ's SAOCOM-CS spacecraft characteristic

SAOCOM-CS system	Spacecraft characteristic
Spacecraft launch mass	Dry mass: 374kg
	Wet Mass: 398kg
Spacecraft orbit	Sun-synchronous
	Inclination: 97.89 deg
	Altitude: 620 km
	LST: 6:12
Spacecraft dimension	974 x 1362 x 1325 mm^3 (stowed)
Propulsion subsystem	Architecture: 4+4 1N Thrusters
	30L tank
	Hydrazine based propulsion subsystem
	Dry mass: 17.08 kg
	Peak power 66.3 W
	Wires 131

Taking this data as a baseline and running the programm, the result of the launcher system is shown in Fig.5.1.
Every bar represent a different launcher, with a click on them an auxiliary window appears showing the data of the rocket that correspond in the evaluation with the bar.
An example is displayed in Fig.5.2, where the bar with the highest ranking is selected. This should be the best result regarding the launcher:

The SAOCOM-CS mounts the Falcon 9 as launcher but this program shows why another one would be more efficient. the results show that the Strela launcher, from Bakour, has a better cost-effectiveness. The main reasons are:

- Small dimension of the S/C, possibility to be the primary payload with all the additional advantages for a reasonable price;
- Injection with the upper stage to the final orbit, minimum request of propellant in order to reach the final orbit;
- The price of this launcher is around 15 M€while the Falcon 9 is around 60 M€.

TABLE 5.2: First launcher option result

Launcher name	Strela
Manufactur	Khrunichev(Russia)
Launch Base	Baikonur(Russia)
Payload configuration	primary payload, fairing SHS-1
Injection	SSO
Delta V required	1 m/s
Launcher cost	10.5 M€
Scoring	9,44

This difference between the theorical solution and the real solution can be caused by the geo-political decision of the launching site, the schedule availability for the launch or even by the increased reliability of one rocket compared to others.

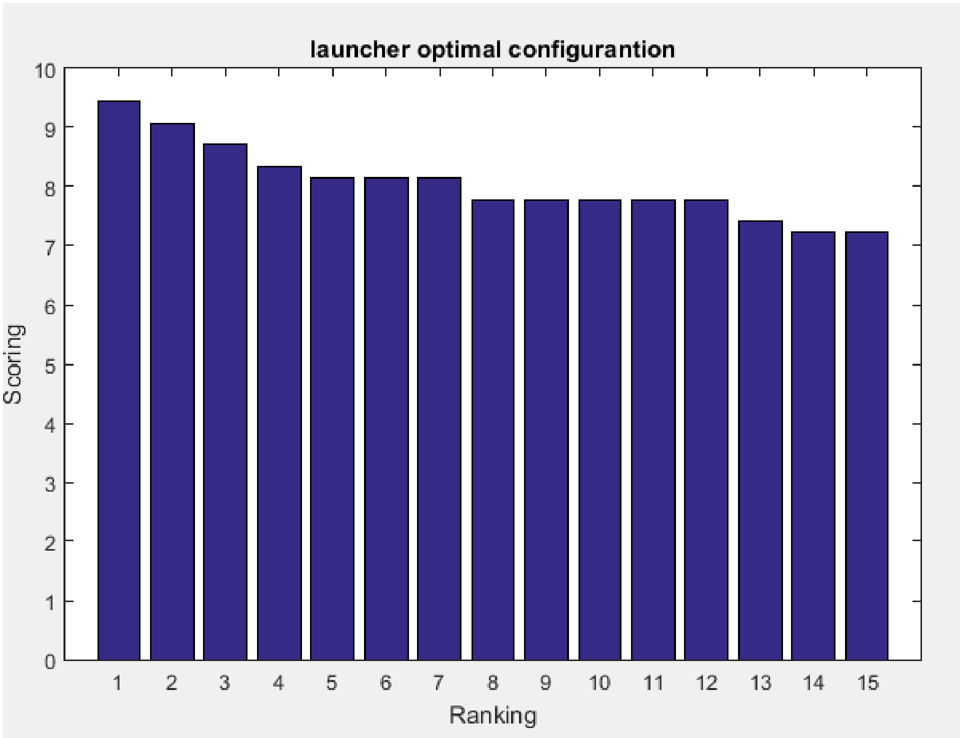


FIGURE 5.1: Prospect launcher results based on SAOCOM-CS w/o costrains

Regarding the propulsion design, selecting the first 10 solutions,

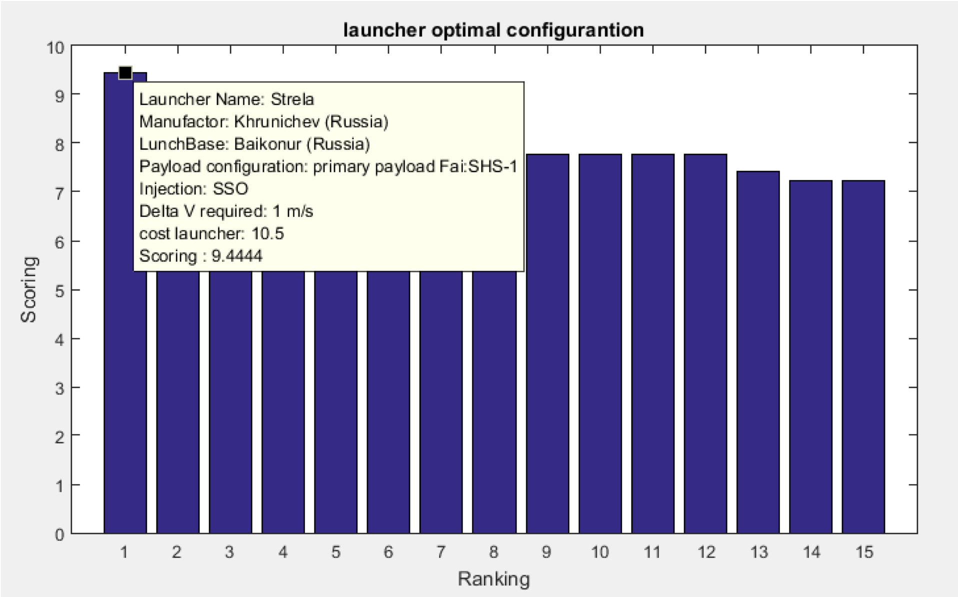


FIGURE 5.2: Prospect launcher results first selection

it's possible to see that the outputs are quite different from the reality, as shown in the following descending Table 5.3. The program can define more solutions as well, though only the first 10 are displayed in order to facilitate the interpretation of the results:

TABLE 5.3: Prospect propulsion system config. for SAOCOM-CS w/o costrains

Single Thruster type	**Name**	**Secondary Thruster type**	**Name**	**Dry mass system(kg)**	**Wet mass system(kg)**	**Power peak(W)**	**Thruster configuration**	**Agility (h t)**	**Tank configuration**	**Score**
Ion thruster	BIT-3	X	X	17.65	20.82	235.9	1	78.97	blowdown single	8.028
Ion thruster	BIT-1	X	X	17.5	22.66	185.9	1	1105.53	blowdown single	7.893
Ion thruster	Rit µX	X	X	17.89	21.59	225.9	1	221.11	blowdown single	7.887
Ion thruster	BIT-3	X	X	17.7	20.91	235.9	1	78.97	blowdown single	7.887
Ion thruster	BIT-1	X	X	17.60	22.71	185.9	1+1	1105.53	blowdown single	7.848
Bipropellant	10N	X	X	8.65	33.91	141.8	4	0.01	regulated double	7.816
Ion thruster	BIT-1	X	X	17.66	22.82	347.5	2+2	884.2	blowdown single	7.790
Ion thruster	Rit µX	X	X	18.33	22.03	225.9	1+1	221.11	blowdown single	7.775
Bipropellant	10N	X	X	10.05	35.31	181.8	8	0.01	regulated double	7.57

Most of the solutions are based on Ion Thrusters, because the propellant mass is significantly less than the one required for chemical and bi-propellant propulsion.
The reasons of the solution are: the price is not considered as a parameter due lack of data, the program doesn't consider the difficulty of assembling and the cost of research for the system.

If the solution doesn't satisfy the requests it's possible to reboot the evaluation and modify the advanced settings, excluding the EP in the solution or the relevance of the power consumption can be increased by using the slider tool.
If EP is deselected from our solution, the table 5.4 will give an overview of the system . Bi-propellant is evaluated as one of the best solutions for the nominal case without EP, while the Green propellant system is found in second position and the 7th position is occupied by mono-propellant.
The main difference between the solution not related to price and TRL is that the green propellant has a higher I_{sp} and that's why the program prefers one solution over the others.
If a system is more oriented to Hydrazine, the green propellant option can be removed as well as the bi-propellant option with the optional commands or even the slider tool can be used to simply increase the values of agility and reliability.
Increasing these slider values, the program will give the priority to the configurations that provide higher agility and reliability.
This shows up in Table 5.5. The Bi-propellant, and the green propellant have a higher I_{sp} but if the required parameter is an higher reliability the Mono propellant is the optimal choice. The bi-propellant solution is better because the I_{sp} generated is higher and this means less propellant consumed for the mission and green propellant too. Then it's also possible to see it related with the mass of the propellant required.
The reliability of the Mono propellant, due to the cheap and well know construction procedure, allows the Mono propellant 1N of the company Airbus to be selected by the program as the optimal solution for us.
This shows the flexibility of the program depending on the general requests or the needs during the phase of design.

TABLE 5.4: Prospect propulsion system config. for SAOCOM-CS w/o EP

Single Thruster type	**Name**	**Secondary Thruster type**	**Name**	**Dry mass system(kg)**	**Wet mass system(kg)**	**Power peak(W)**	**Thruster configuration**	**Agility (h)**	**Tank configuration**	**Score**
Bipropellant	10N	X	X	18.65	40.91	141.8	4	0.01	regulated single	8.73
Green Propellant	HPGP$(5N)$	X	X	6.16	38.09	161	4	0.02	regulated single	8.23
Bipropellant	10N	X	X	20.05	41.31	181.8	4+4	0.01	regulated single	8.09
Green Propellant	HPGP$(5N)$	X	X	18.85	41.00	161	4	0.02	regulated single	7.88
Green Propellant	HPGP$(1N)$	X	X	18.77	43.27	133.8	4	0.11	regulated single	7.45
Green Propellant	HPGP$(5N)$	X	X	17.76	39.69	221	4+4	0.02	regulated single	7.34
Mono Propellant	1N	X	X	1 8.41	44.69	133.8	4	0.1	regulated single	7.23
Green Propellant	HPGP$(0.5N)$	X	X	17.97	43.92	133.6	4	0.22	regulated single	7.16
Arcjet	MR-510	X	X	17.38	25.15	195	4	2	blowdown single	7.03
Green Propellant	HPGP$(5N)$	X	X	20.45	42.60	221.8	4+4	0.02	double tank	7.01

TABLE 5.5: Prospect propulsion system config. for SAOCOM-CS w/o EP and high reliability

Single Thruster type	**Name**	**Secondary Thruster type**	**Name**	**Dry mass system(kg)**	**Wet mass system(kg)**	**Power peak(W)**	**Thruster configuration**	**Agility (h)**	**Tank configuration**	**Score**
Mono Propellant	MR-111C	X	X	15.62	43.72	132.9	4	0.1	blowdown single	8.83
Bipropellant	10N	X	X	18.65	35.91	141.8	4	0.01	regulated single	8.79
Green Propellant	HPGP$(5N)$	X	X	16.16	39.09	161	4	0.02	regulated single	8.69
Mono Propellant	1N	X	X	19.57	44.84	165.8	4+4	0.1	blowdown single	8.44
Mono Propellant	1N	X	X	18.41	44.69	133.8	4	0.1	regulated single	8.23
Bipropellant	10N	X	X	18.65	33.91	141.8	4	0.01	regulated double	8.13
Green Propellant	HPGP$(5N)$	X	X	6.16	38.09	161	4	0.02	regulated single	8.11
Bipropellant	10N	X	X	10.05	35.31	181.8	4+4	0.01	regulated double	8.09
Green Propellant	HPGP$(5N)$	X	X	18.85	41.00	161	4	0.02	regulated double	7.88
Green Propellant	HPGP$(1N)$	X	X	18.77	43.27	133.8	4	0.11	regulated double	7.45

Proba V The Proba satellites are part of ESA's in-orbit technology demonstration program. These are missions dedicated to the demonstration of innovative technologies with lower budget expense respect the past.
The PROBA-V [1] (Vegetation) mission definition is an attempt, spearheaded by ESA and CNES, to accommodate an improved smaller version of the large VGT (Vegetation) optical instrument of SPOT-4 and SPOT-5 mission heritage on a smaller satellite bus, such as the one of PROBA-2.
Vegetation principally addresses key observations in the following application domains:

- General land use in relation to vegetation cover and its changes;
- Vegetation behavior to strong meteorological events (severe droughts) and climate changes (long-term behavior of the vegetation cover);
- Disaster management (detection of fires and surface water bodies);
- Biophysical parameters for model input devoted to water budgets and primary productivity (agriculture, ecosystem vulnerability, etc.).

Proba-V [2] was launched from ELA-1 at Guiana Space Centre on board the second launch of the VEGA rocket on 7^{th} May 2013 together with the Vietnamese VNREDSat 1A satellite, and Estonia's first satellite, ESTCube-1.
The launch marked the first test of the new VESPA dual-payload adapter. PROBA-V rode in the upper position of the VESPA adapter, and VNREDSat 1A will sit in the lower position.
Given that Proba-V has no on-board propulsion, the natural drift of this LTDN depends on the satellites' in-orbit injection accuracy. Based on the injection accuracy specifications of the VEGA launcher, a usable lifetime between 2.5 and 5 years was predicted.

The mission, Table 5.6, doesn't request a specific propulsion system, even if the program can analyze the best launcher in order to accomplish the orbit injection. If no value is inserted in the text-box 'S/C wet mass' in the GUI, the program will show a warning, Fig.5.3. If a solution w/o propulsion system is selected, the program will use the satellite dimensions in order to select the configuration which is the most competitive for the launch and then the program will rank the configurations by comparing the differences of velocity, required to reach the parking orbit.

[1] Eoportal data `https://directory.eoportal.org/web/eoportal/satellite-missions/p/proba-v`

[2] Eoportal data `https://directory.eoportal.org/web/eoportal/satellite-missions/p/proba-v`

TABLE 5.6: QinetiQ's Proba V spacecraft characteristic

Proba V system	Spacecraft characteristic
Spacecraft launch mass	140kg
Spacecraft orbit	Sun-synchronous Inclination: 98.8 deg Altitude: 820 km LST: 10:30
Spacecraft dimension	800 x 800 x 100 mm^3 (stowed)
Propulsion subsystem	Architecture: None

The ΔV in this case is a representation of the difference of altitude in order to reach the orbit: if the result displayed is just 1 m/s, it means that the upper stage can provide an almost perfect injection to reach the working orbit of the satellite. The results of the iteration are displayed in Table 5.7. Only the first 10 solutions are selected to display and the non-technical parameters are not taken into account.
Not selecting non-technical parameters implies that all the launcher possibilities are considered, without considering ITAR restrictions and ESA preferences. The solutions are a perfect match of the Launcher selected for the mission, proving that VEGA was the best solution for such a small satellite in the primary position for the launch.

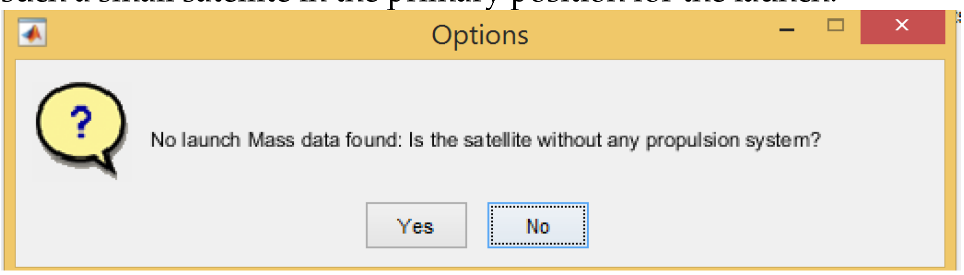


FIGURE 5.3: Warning option if S/C wet mass is not inserted

TABLE 5.7: Proba V possible launcher solutions

Launcher Name	Manufactor	Launch Base	Payload Configuration	Injection	ΔV required(m/s)	Price(M€)
VEGA-VESPA	ESA/ASI(Europe)	Korou(French Guiana)	dual payload	LEO	Null	10
Strela	Khrunichev(Russia)	Baikour(Russia)	Primary payload	LEO	Null	10.5
VEGA-VESPA	ESA/ASI(Europe)	Korou(French Guiana)	primary payload	LEO	Null	23.5
Rokot	Khrunichev(Russia)	Baikour(Russia)	Primary payload	LEO	Null	27
Soyuz 2.1a	TsSKP-Progress(Russia)	Baikour(Russia)	Mini payload in ASAP 5	SSO	31.80	5
Arianne 5 ECA	Airbus Space & Defense(Europe)	Korou(French Guiana)	Mini payload in ASAP 5	SSO	36.20	5
Athena 2c	ATK/Lockheed Martin(US)	Spaceport Florida (US)	Mini payload in ESPA	LEO	153.02	5
Minotaur	Orbital(US)	Vandenberg (US)	Primary payload	SSO	139.12	13
H−IIA 202	Mitsubishi Heavy Industries(Japan)	LA-Y, Tanegashima(Japan)	Mini payload in ESPA	SSO	305.46	5
CZ-2C/SD	CALT(China)	Taiyuan Satellite Launch Center(China)	Primary payload	SSO	138.31	20

Low Orbit mission This paragraph analyzes the possible propulsion system configurations for the Low LEO orbit, instead of comparing different options for an already developed mission.
The occurrence of these particular kinds of missions within the satellite market are increased for several reasons:

- Development of miniaturized technology becomes more widely available;
- Less expensive launch costs;
- Less propellant required for disposal, mandatory from 2020;
- Easier and more profitable study of the Earth's surface and magnetosphere.

An example that can be taken into consideration is GOCE, Gravity Field and Steady-State Ocean Circulation Explorer, whose mission was described as:
The mission objectives are to determinate the stationary gravity field - geoid and gravity anomalies with high accuracy (1 cm of geoid heights) at spatial grid resolutions of 100 km or less over the Earth's surface. GOCE mission flew at an extremely low altitude, just 260 km from the Earth surface. Mixing all this data makes possible to define the mission TAB.5.8

TABLE 5.8: Low orbit spacecraft characteristic

Low Orbit system	Spacecraft characteristic
Spacecraft launch mass	Dry mass: 300kg
	Wet Mass: 400kg
Spacecraft orbit	Sun-synchronous
	Inclination: 98 deg
	Altitude: 300 km
	LST: 10:30
Spacecraft dimension	100 x 1000 x 1400 mm^3 (stowed)
Propulsion subsystem	Architecture: TBD
Operating Life	6/7 years
Launcher cost limit	60 M€

This possible mission is analyzed and verified with the program. First, the solution for the launchers will be displayed and explained, then the solution for the thrusters will be similarly evaluated and finally a brief call for the merged solution (launcher/thrusters) will be discussed.

Launcher solutions: the request for this particular mission is a satellite with a lower price that requires less propellant for the injection.
Taking into account any possible configuration that was obtained, without ITAR restriction or any particular preferences of the customer, Table 5.9 presents an overview.
The code indicates the Strela launcher as the best solution, due to the fact that a perfect injection in the parking orbit is obtained with the upper stage only and it has a relatively cheap price-tag.
If the satellite is to be commissioned from a US company, the customer will probably prefer a US launcher. The code shows as a 3^{rd} solution the Athena 1c/bis that has a cheap price but requests a ΔV of 67.08 m/s in order to reach the parking orbit.
The disadvantage of increasing the ΔV to reach the parking orbit and this influences all S/C is an increase of the propellant used for the injection which means less propellant for the rest of the mission or a loss of available payload mass.
For a European Customer as ESA, the best solution displayed is the Arianne 5G, dual configuration inside the SYLDA_5 adapter. The overall price of an Arianne 5 is more than a VEGA, but the propulsive power of the Arianne's upper stage allows for perfect injection in the parking orbit that cannot be reached with the VEGA rocket, and the launch cost could be split with other mission in secondary position or piggy back.
Propulsion system configuration solutions:
The Table 5.10 shows the thruster configurations for the system. A first noticeable point is that all the solutions are based on EP.
Due to the really low orbit, the propulsion system has to principally balance the drag effect, which is possible with a semi-continuous thrust generated by an Electric Thruster.
It is confirmed that EP is an excellent choice regarding this parameter, because the reference mission of GOCE used a ion thruster as well, specifically the T5 of QinetitQ UK.
The solution displayed uses two possible technologies named ION and HET, which are respectively the power requirement and the propellant required. Ion thrusters with a higher I_{sp} require less propellant, but this has to be balanced with more power which implies a larger power system.
It could be interesting also to consider if the components are ITAR FREE. If this parameter is selected in the advanced settings of the GUI, the results' best match will always be an ION thruster, but, instead of being the BIT-3, it will be the T5 as already used for the GOCE mission.

TABLE 5.9: Launchers solutions for Low Orbit satellite

Launcher Name	**Manufactor**	**Launch Base**	**Payload Configuration**	**Injection**	**ΔV required(m/s)**	**Price(M€)**
Strela	Khrunichev(Russia)	Baikour(Russia)	Primary payload	SSO	Null	10.5
Moniya-M	TsSKP-Progress(Russia)	Baikour(Russia)	primary payload	LEO	Null	15
Athena 1c/bis	ATK/Lockheed Martin(US)	Spaceport Florida (US)	primary payload	SSO	67,08	16
Soyuz-U	TsSKP-Progress(Russia)	Baikour(Russia)	primary payload	SSO	Null	25
CZ-2C/SD	CALT(China)	Taiyuan Satellite Launch Center(China)	Primary payload	SSO	78.23	20
Soyuz-2,1b	TsSKP-Progress(Russia)	Baikour(Russia)	primary payload	SSO	Null	40
Athena 1c	ATK/Lockheed Martin(US)	Spaceport Florida (US)	primary payload	SSO	246.68	16
Rokot	Khrunichev(Russia)	Baikour(Russia)	Primary payload	LEO	212.77	27
Arianne 5 G(2)	Airbus Space(Europe) & Defense	Korou(French Guiana)	dual payload in Sylda_5	SSO	Null	60

TABLE 5.10: Thrusters solutions for Low Orbit satellite

Single Thruster type	**Name**	**Secondary Thruster type**	**Name**	**Dry mass system(kg)**	**Wet mass system(kg)**	**Power peak(W)**	**Thruster configuration**	**Agility (h)**	**Tank configuration**	**Score**
Ion Thruster	BiT-3	X	X	17.65	32.85	235.9	1	79.36	blowdown single	8.49
Ion Thruster	BiT-3	X	X	18.25	33.45	415	2+2	64.43	blowdown single	7.90
Ion Thruster	BiT-3	X	X	19.05	34.25	655	4+4	50.44	blowdown single	7.24
Ion Thruster	T5	X	X	25.5	40.32	850.9	1	5.56	blowdown single	6.88
Ion Thruster	T5	X	X	27.88	43.086	850.9	1	5.56	blowdown double	6.72
HET	BHT-200		X	X 18.92	56.49	836.61	1	8.55	blowdown single	6.44
HET	BHT-200		X	X 21.68	59.69	836.61	1	8.55	blowdown double	6.2
Ion Thruster	RIT-10	X	X	28.45	45.04	1050.9	1	4.44	blowdown single	6.19
HET	H-100		X	X 18.38	56.65	986.61	1	7.41	blowdown single	6.09
Ion Thruster	RIT-10	X	X	31.18	47.74	1050.9	1	4.44	blowdown double	6.19

Chapter 6

Conclusion

The program most useful application is within larger simulations to define where mass and propellant fractions of hypothetical spacecraft must be generated for a range of scenarios. When the mass of the satellite was brought into the nanosatellite range (1-10 kg), the program showed that monopropellant thrusters were the most feasible option, due to the excessive weight of the PPU and FCU. This research has as result a useful tool for the spacecraft design and simulation.

Although it has limitations that are important to understand, it also carries the unique capability of providing a quick preliminary design and it estimates that it would be much more time consuming. In addition, the database itself can be useful if looking for a particular thruster mode or launcher.

The validity of the program lies in its use of historical data. Assumptions have been made, but a lot of these are constants that can easily be adjusted.

There are several improvements that could be made to the program that are still unexplored.

The first is the calculation of power system mass. Instead of using a single constant for power density, different constants could be used for different types of propulsion to represent the diverse power conditioning requirements for each type of propulsion. Scaling equations instead of constants could be used as well, reflecting the difficulty of scaling the power systems into the low mass regime. If the data on power processing units and power system design are readily available, this would be very easy to add into the program.

Some thrusters, such as PPTs, FEEPs, are commonly integrated with their propellant and power systems. Although the program calculates thrusters, propellant storage, and power separately, exceptions could be made for these thrusters if the necessary data was gathered.

Information on the the cost, reliability, or flight status of thrusters could be added and filters could be implemented to only return data on, for example, commercially available flight qualified thrusters. Ultimately, this direction leads away from simulation and prediction and towards choosing an actual thruster model for the designer. It is worth noting, however, that data on efficiency and minimum impulse but have already been gathered for many thrusters in the database and could easily be incorporated into any modifications. The real strength in the program, as said, lies in how quickly it can compare so many different options. As a spacecraft moves from its preliminary design phase, more extensive and specific calculations are required. For assistance in comparing propulsion options or estimating mass fractions, however, this research fills a much needed role.

Appendix A

Adapters mechanical drawings

This appendix is going to collect all the mechanical drawings took as references for defining the available dimensions for the S/C, more in detail information of the interface needed to connect to the satellite and requirements are available on the users manual.

A.1 ASAP

In order to provide launch opportunities to micro Auxiliary Payload (Mass $\leq$ 120 kg) and mini Auxiliary Payload (120 kg$\leq$ Mass $\leq$ 300 kg), ARIANESPACE has developed a structure called ASAP 5 (ARIANE 5 structure for Auxiliary Payload) to carry and deploy small and medium satellites on LEO, SSO, MEO or GTO orbits.

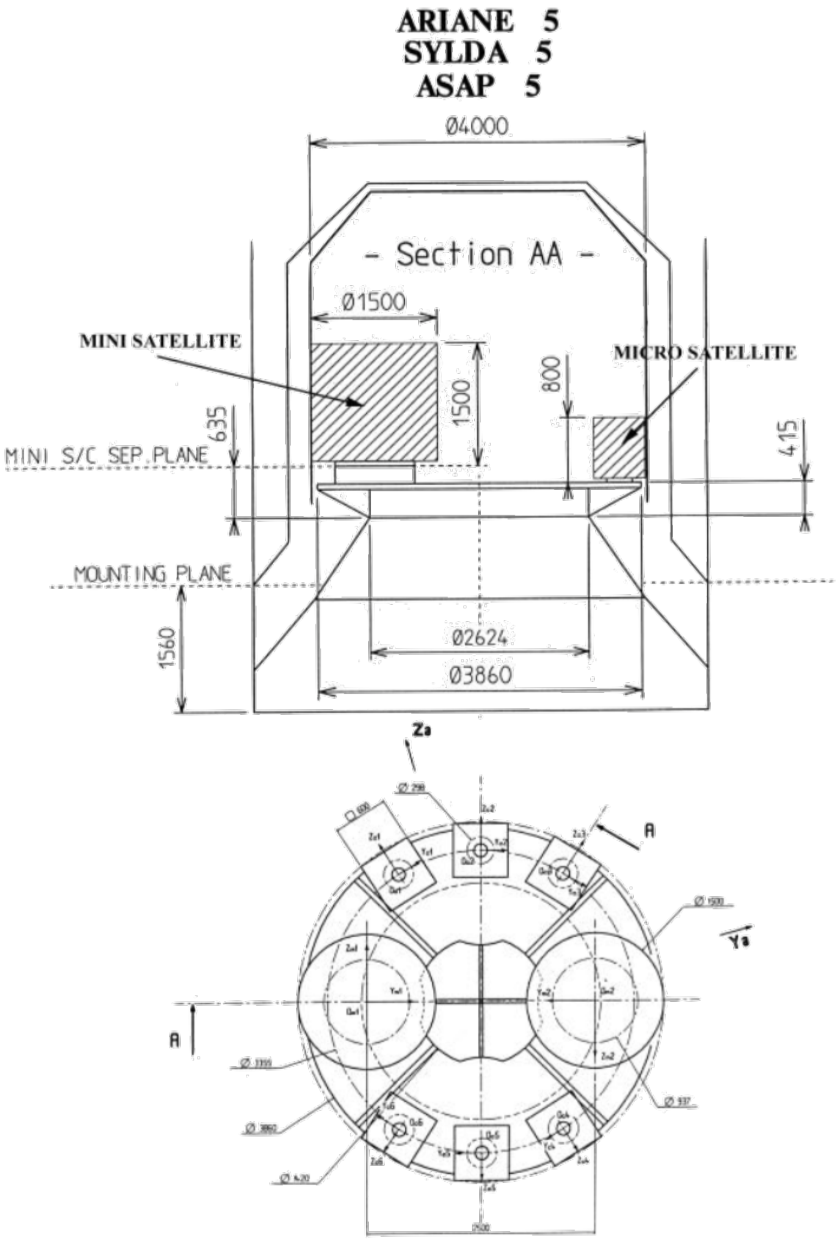


FIGURE A.1: Mixed configuration of 2 mini and 6 micro Auxiliary Payloads,copyright ArianeSpace

A.2 VESPA

The VESPA (VEga Secondary Payload Adapter) has been developed to offer launch opportunities to mini satellites on VEGA Launcher. It has successfully flown on the second Vega flight in 2013. The VESPA carrying structure allows to embark:

- passenger in upper position (1000 kg max);
- passenger(s) inside the VESPA cavity (600 kg max in total).

The VESPA consists of a load bearing carbon structure, comprising a cylindrical part enclosing the lower passenger(s) with their adapter, and an upper conical shell supporting the main passenger. The separation of the VESPA upper part is achieved by means of a clamp band and the distancing is ensured by a series of springs, more in detail information are available on the user manual of VEGA launcher.

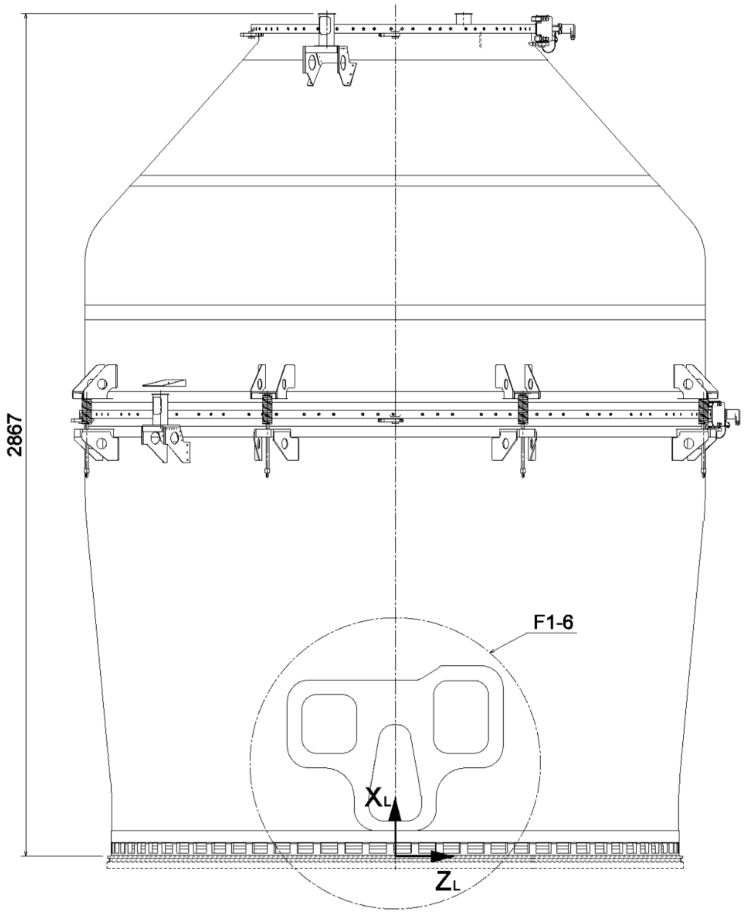
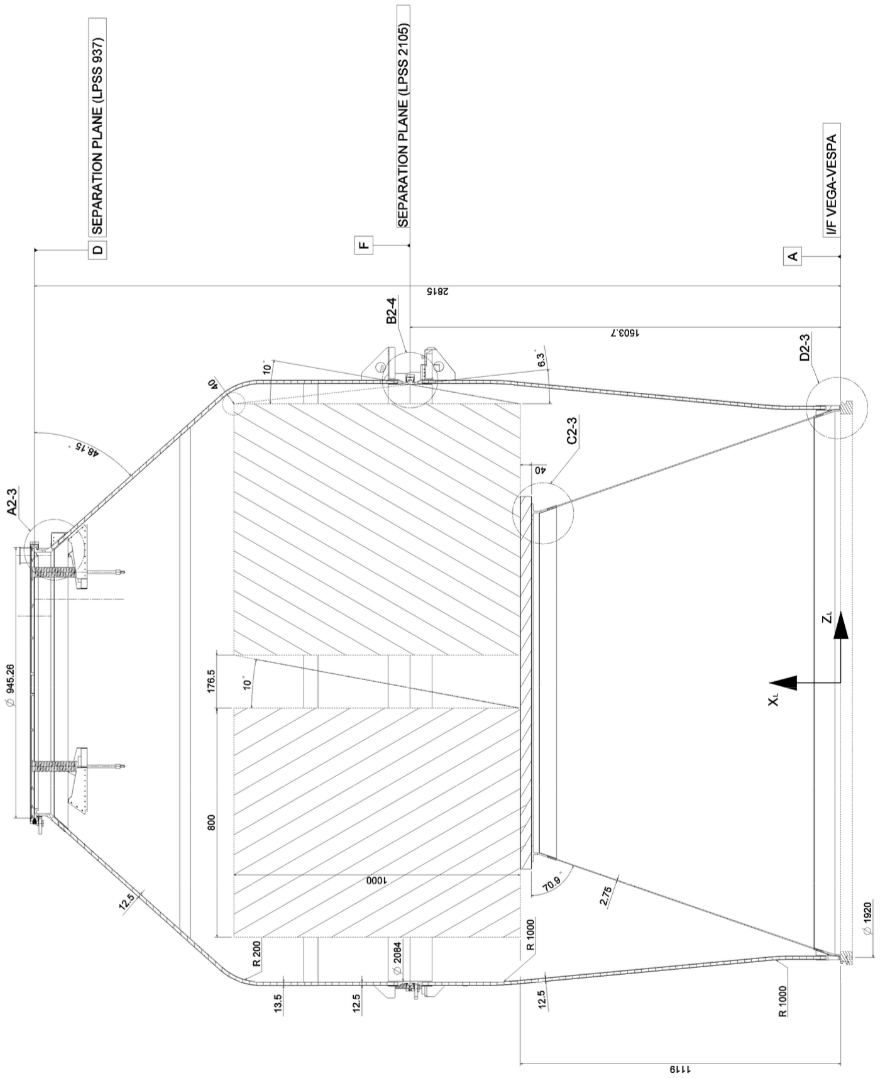


FIGURE A.2: VESPA Front view, copyright,ArianeSpace



GENERAL SECTION

FIGURE A.3: Mixed configuration mini micro Auxiliary Payloads on VESPA Adapter, copyright ArianeSpace

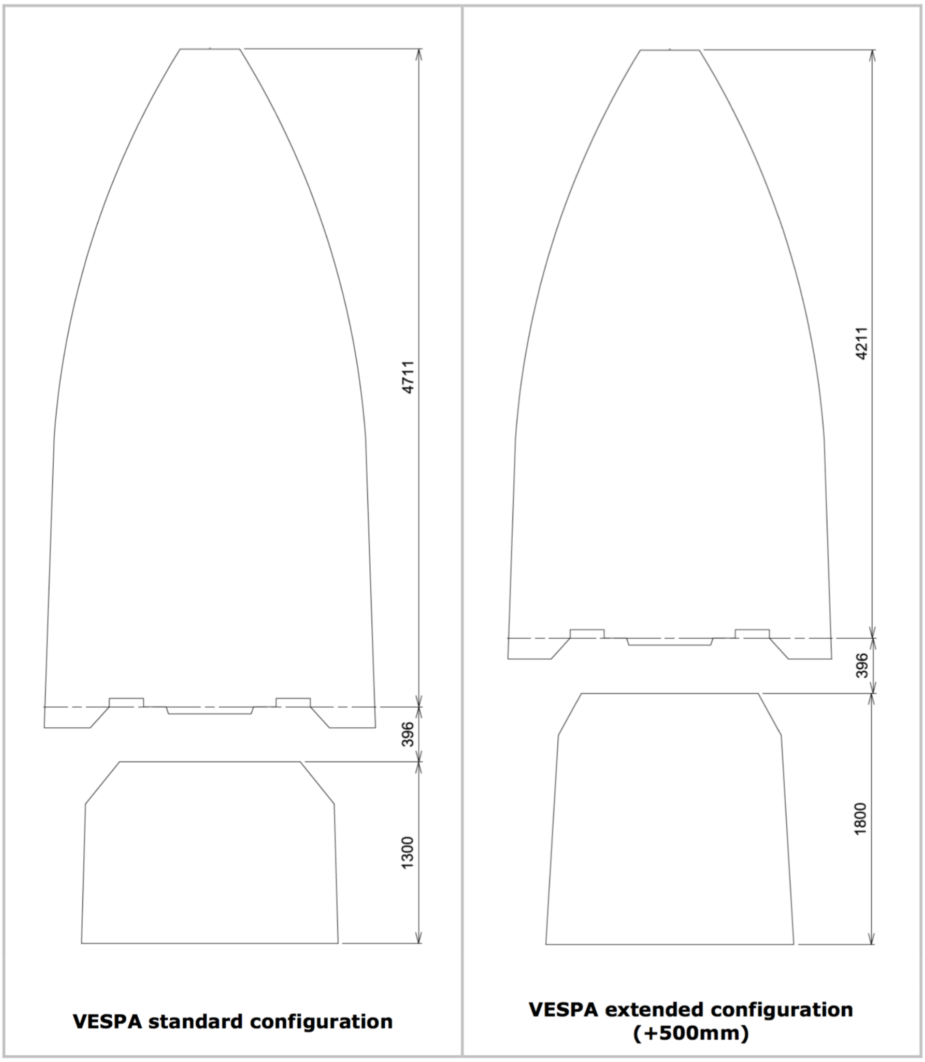


FIGURE A.4: VESPA possible configuration, copyright ArianeSpace

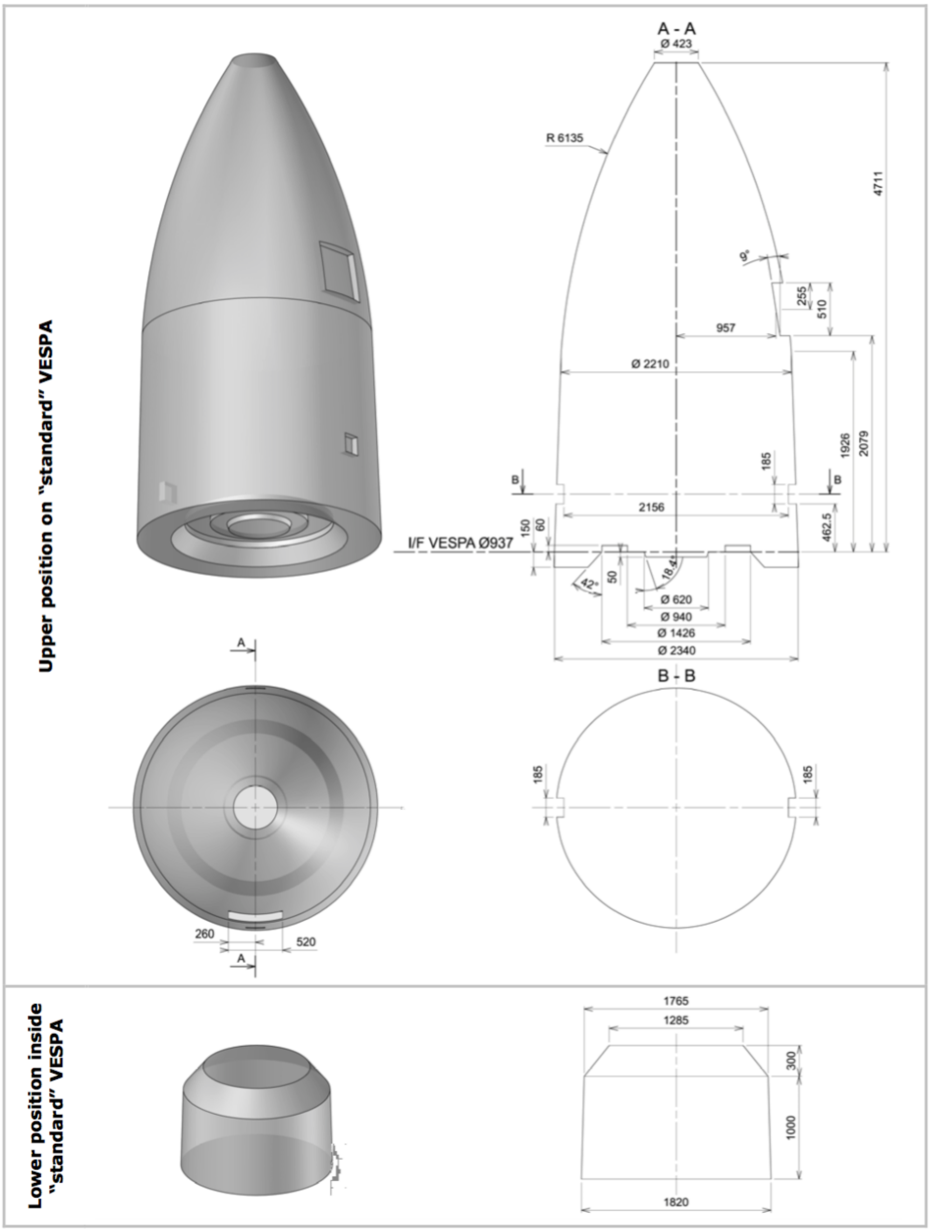


FIGURE A.5: VESPA standard configuration, copyright ArianeSpace

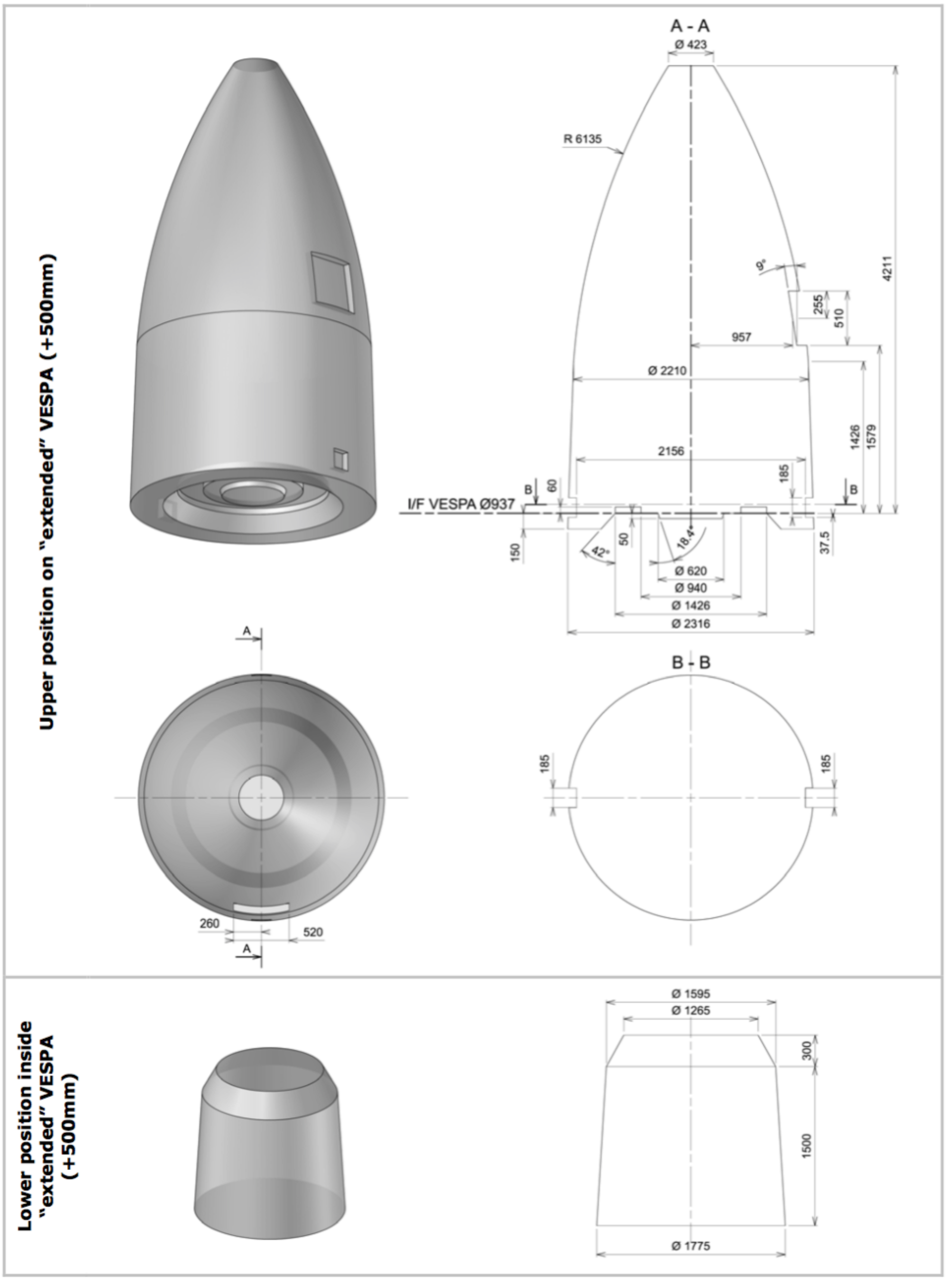


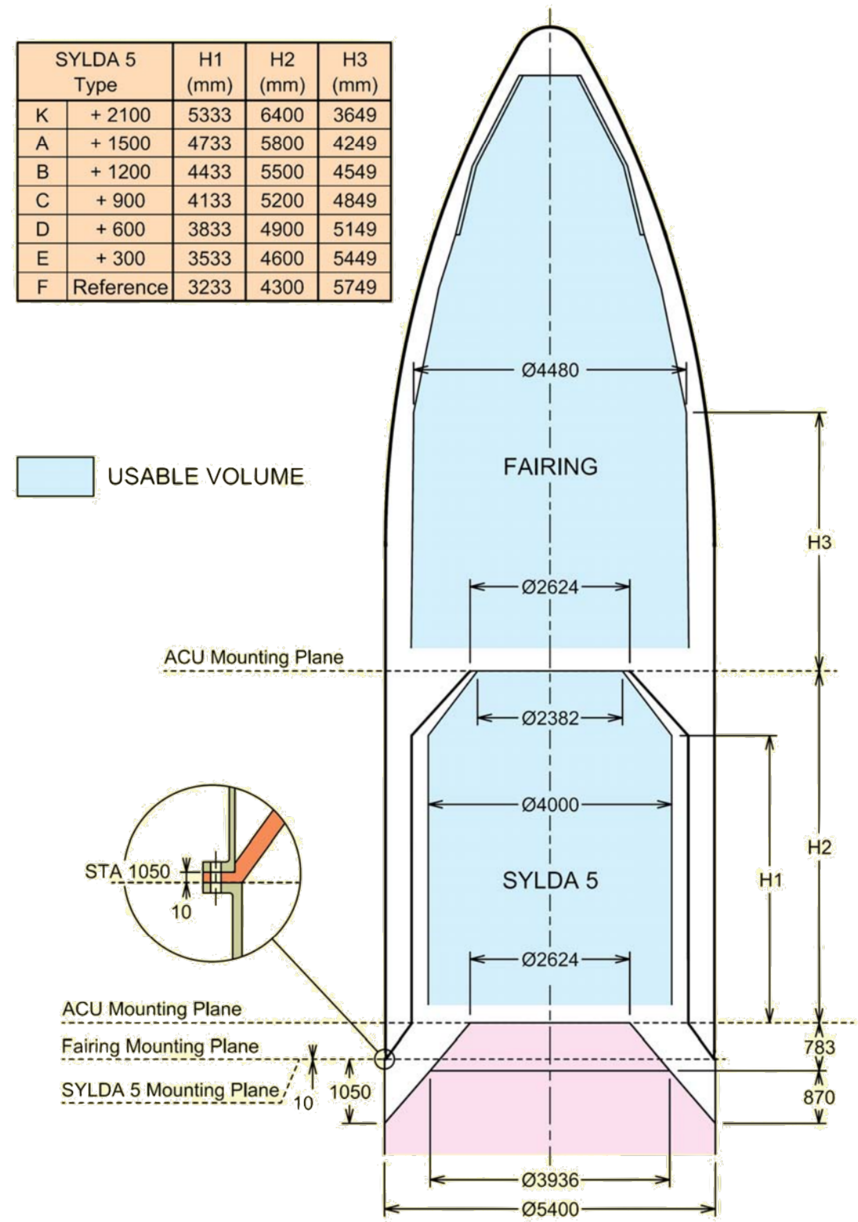
FIGURE A.6: VESPA long configuration,copyright ArianeSpace

A.3 Sylda 5

The Sylda 5 is housed inside the fairing. At the bottom it has a 50 cm cone which interfaces with the vehicle equipment bay. The Sylda 5 is smaller than the Speltra; the cylinder is 4.5 m in diameter and 3.2 m high, while the cone above is 1 m high and ends in the 2.6 m diameter of the standard payload interface. When empty the Sylda 5 weighs 440 kg.

During a standard dual-launch mission, the upper satellite is released first. Then, the Speltra or Sylda-5 is jettisoned in order to release the second satellite. Both systems have pyrotechnic separation systems at their base and push-off springs.

Separation is triggered by two detonators which sever the steel attachment between the vehicle equipment bay and the Speltra or Sylda. The dual payload structure is then pushed away by means of eight special steel springs.



SYLDA 5 Type		H1 (mm)	H2 (mm)	H3 (mm)
K	+ 2100	5333	6400	3649
A	+ 1500	4733	5800	4249
B	+ 1200	4433	5500	4549
C	+ 900	4133	5200	4849
D	+ 600	3833	4900	5149
E	+ 300	3533	4600	5449
F	Reference	3233	4300	5749

FIGURE A.7: SYLDA 5 configuration ,copyright ArianeSpace

A.4 ESPA

ESPA is a cylindrical aluminum structure that duplicates the EELV Standard Interface Plane (SIP) for the primary payload, and provides up to six slots for deployment of secondary satellites (or payloads), the following photos use imperial units.

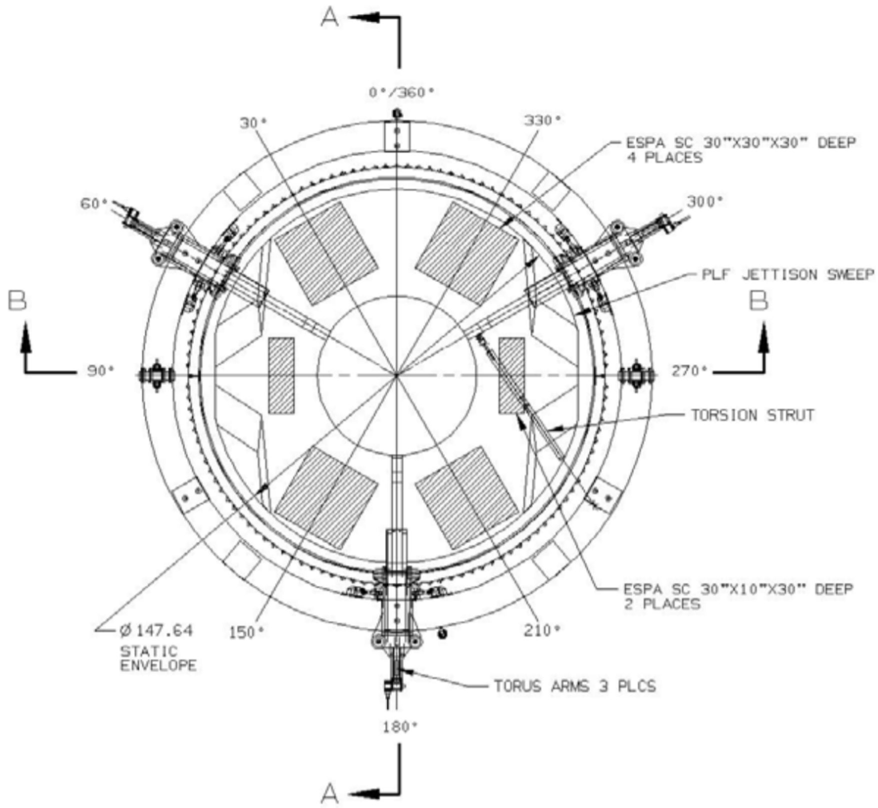


FIGURE A.8: Top View of ESPA Volumes for Atlas-V 400 Series LV, copyright Moog

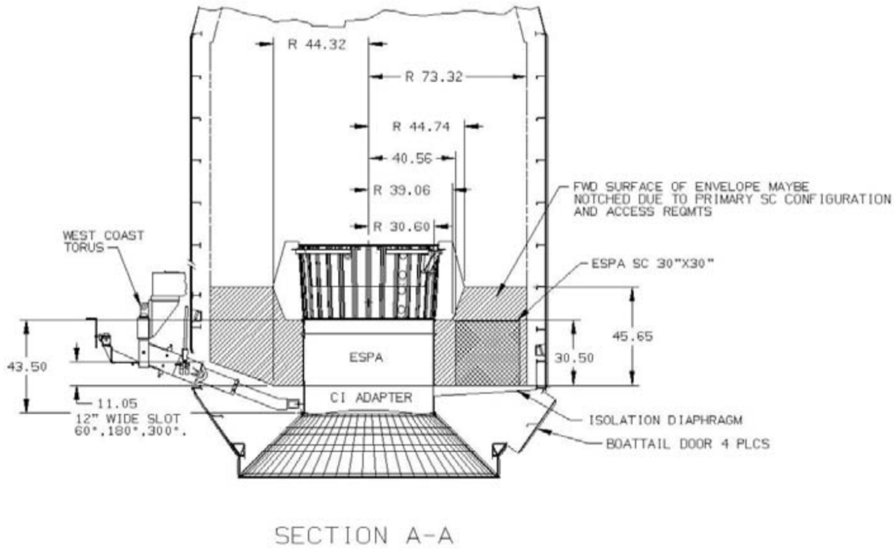


FIGURE A.9: Side View for Standard ESPA Satellite Volume on Atlas-V 400 Series LV, copyright Moog

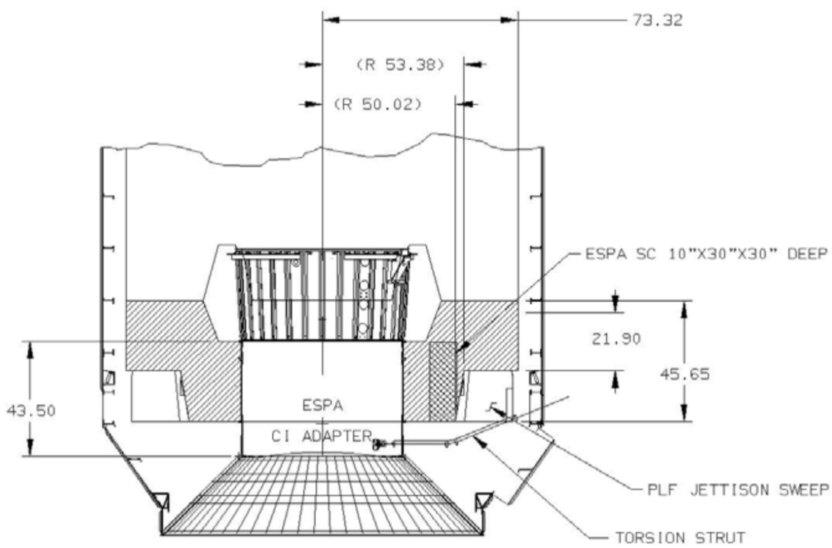


FIGURE A.10: Side View for Limited-Sized ESPA Satellite Volume on Atlas-V 400 Series LV, copyright Moog

Appendix B

Propellant Tank Design

For a blowdown system the procedure for design is :
- Calculate the propellant required: add margin;
- The propellant control device;
- Decide dual-vs single-propellant tanks;
- Decide propellant tank type: sphere, barrel, conosphere;
- Select the pressurant: helium if mass is critical,otherwise nitrogen;
- Select pressurization system type and set the performance parameters, max tank pressure and blowdown ratio;
- Design the propellant tank;
- Design the engine modules and general arrangement;
- Calculate system mass;
- Conduct trade studies of system alternatives; repeat the process.

We made some simplification obtaining. The mass of propellant loaded with all the margins , m_{pl} ; the propellant tank volume should be calculated based on propellant density at maximum temperature, ρ:

$$V_{pl} = \frac{m_p}{\rho} \tag{B.1}$$

is possible to define the volume of propellant usable, m_{pu} using:

$$V_{pu} = \frac{m_p}{\rho} \tag{B.2}$$

The initial ullage volume is, taking into account a nominal blowdown ratio of 4:

$$V_u = \frac{V_{pu}}{3} \tag{B.3}$$

The preliminary volume summing Eq.B.1 + Eq.B.5:

$$V_{t1} = V_{pl} + V_u \tag{B.4}$$

For tanks that have a diaphragm inside must be added a 1% volume for each tank, knowing this data is possible to derive e build the perfect tank for a mission, we took the current already developed tanks. The initial ullage volume is so:

$$V_u = V_{pl} + V_u \tag{B.5}$$

In the development of a perfect tank this would be the solution of our ullage, in a more realistic analysis the tank selected is going to be the one with a volume closer to V_u; the helium mass loaded is derived from the equation of state (specific constant of Helium is 2078.5) :

$$W_g = \frac{PV}{RT} \tag{B.6}$$

where the temperature of working is usual 290 K and the pressure is 3619750 Pa.

For a bipropellant system the first basic information is the mixture ratio, oxidizer and fuel:

$$\frac{Mr = u_o}{u_f} \tag{B.7}$$

where u_o s the oxidizer: weight flow rate and u_f the fuel weight flow rate, rearranging the equation the we can derive the usable propellant into the fuel and oxidizer:

$$W_f = \frac{W_u}{1 + Mr} \tag{B.8}$$

and

$$W_o = W_u - W_f \tag{B.9}$$

where W_u is the total usable propellant weight, W_f is the total fuel load and W_o the total usable oxidizer load, all in Kg.
For the bipropellant system the margin philosophy is: 3% of trapped propellant , 1% of outage and 0.5% of loading error, as pressurat either Helium or Nitrogen can be used, for our system we just considered Helium, stored in high pressure tanks between 21000 and 34500 kPa, for bipropellant systems each propellant tank ullage must be isolated to prevent vapor mixing.
For constant temperature condition the volume or the pressurant sphere is

$$V_s = \frac{P_r V_u}{P_1 - P_2} \tag{B.10}$$

where V_s is the volume of the pressurant sphere, P_r the regulated propellant tank pressure, V_u the volume of the usable propellant,, P_1 the initial pressurant sphere pressure and P_2 the final pressurant sphere pressure.
The weight of the initial gas load from the equation of state, B.6, the same for the pressurant loaded, knowing the parameters requests by

the missions and the manufactures the final match is our weight and volume required.

Appendix C

Database datasheet

This appendix resume all the data collected as base for the program, the table are shown in the following order:

1. Launchers
2. Thrusters
3. Tubing and small other components
4. Tanks
5. Pressure Vassels

Name	Manufactor	launch base	Mass to Leo (kg)	Leo Altitude (km)	Leo Inclination(deg)	GTO (geosynchronous transfer orbit kg)	GTO inclination(deg)	GTO velocity (Km/s)	GSO (geostazionary orbit kg)	GEO (kg)	SSO (Sun sincronous Orbit kg)	SSO Circular Altitude (km)	SSO inclination (deg)	Escape velocity LEO (3,150 Km/s) (kg)	Fairing	Satellite diameter (mm)	satellite height (mm)	dual configuartion	dual satellite config diameter (mm)	dual satellite config height (mm)	Price (M$)	Value Year	Piggyback Adapeter	Mini satellite diameter (mm)	Mini satellite height (mm)	Micro satellite diameter (mm^3)	Micro satellite height (mm^3)	secondary cost (M$)	Piggy-back Mini cost (M$)	Piggy-back micro (M$)	upper stage propellant/4 (Kg, last manouve	upper stage mass (Kg)	upper stage ISP (s)
Angara 1.2	Khrunichev (Russia)	Plesetsk (Northern Russia)	3500	200	63					8000											25	2003											
Angara 3	Khrunichev (Russia)	Plesetsk (Northern Russia)	14000	200	63	2400	25		1000																						538	3180	463
Angara A5	Khrunichev (Russia)	Plesetsk (Northern Russia)	24000	200	63	7500	25		4500																						980	3930	463
Antares 230	Orbital(United States)	Wallops (United States)	6400	200	51,6						3600	200	99			3505	4197																
Antares 231	Orbital(United States)	Wallops (United States)	5900	200	51,6											3505	4197																
Antares 232	Orbital(United States)	Wallops (United States)				1900	28,5									3505	4197																
Arianne 5G(1)	Airbus Space & Defence	Kourou(French Guiana)	18000	550	5,23	6640	7	1,5			9500	800	98.6		Short	4570	5770				125	2004									745	4540	446
Arianne 5G(2)	Airbus Space & Defence	Kourou(French Guiana)	18000	550	5,23	6640	7	1,5			9500	800	98.6		Short			SYLDA_5	4570	5600	125	2004						60			745	4540	446
Arianne 5G(1)	Airbus Space & Defence	Kourou(French Guiana)	18000	550	5,23	6640	7	1,5			9500	800	98.6		Medium	4570	6850				140	2004									745	4540	446
Arianne 5G(2)	Airbus Space & Defence	Kourou(French Guiana)	18000	550	5,23	6640	7	1,5			9500	800	98.6		Medium			SYLDA_5	4560	6400	140	2004						60			745	4540	446
Arianne 5G(1)	Airbus Space & Defence	Kourou(French Guiana)	18000	550	5,23	6640	7	1,5			9500	800	98.6		Long	4570	10040				155	2004									745	4540	446
Arianne 5G(2)	Airbus Space & Defence	Kourou(French Guiana)	18000	550	5,23	6640	7	1,5			9500	800	98.6		Long			SYLDA_5	4560	6400	155	2004						60			745	4540	446
Arianne 5 ES(V)	Airbus Space & Defence	Kourou(French Guiana)	19300	500	5,23	7575	7	1,5			15700	800	98.6	3500							120	2000									745	4540	446
Arianne 5 ECA	Airbus Space & Defence	Kourou(French Guiana)	10000	500	5,23	9600	7	1,5			10000	800	98.6	4300	Short	4570					150	2002	ASAP-5	1500	1500	600	800	60	5	2,5	745	4540	446
Arianne 5 ECA	Airbus Space & Defence	Kourou(French Guiana)	10000	500	5,23	9600	7	1,5			10000	800	98.6	4300	Short	4570					150	2002	ASAP-5	1500	1500	600	800	60	5	2,5	745	4540	446
Arianne 5 ECA(1)	Airbus Space & Defence	Kourou(French Guiana)	10000	500	5,23	9600	7	1,5			10000	800	98.6	4300	Medium	4570	6660	SYLDA_5	4000	3833	150	2002	ASAP-5	1500	1500	600	800	60	5	2,5	745	4540	446
Arianne 5 ECA(2)	Airbus Space & Defence	Kourou(French Guiana)	10000	500	5,23	9600	7	1,5			10000	800	98.6	4300	Medium	4570	6660	SYLDA_5	4000	4133	150	2002	ASAP-5	1500	1500	600	800	60	5	2,5	745	4540	446
Arianne 5 ECA(1)	Airbus Space & Defence	Kourou(French Guiana)	10000	500	5,23	9600	7	1,5			10000	800	98.6	4300	Long	4570	10042	SYLDA_5	4000	4433	150	2002	ASAP-5	1500	1500	600	800	60	5	2,5	745	4540	446
Arianne 5 ECA(2)	Airbus Space & Defence	Kourou(French Guiana)	10000	500	5,23	9600	7	1,5			10000	800	98.6	4300	Long	4570	10042	SYLDA_5	4000	4733	150	2002	ASAP-5	1500	1500	600	800	60	5	2,5	745	4540	446
Arianne 5 ECA(3)	Airbus Space & Defence	Kourou(French Guiana)	10000	500	5,23	9600	7	1,5			10000	800	98.6	4300	Long	4570	10042	SYLDA_5	4000	5333	150	2002	ASAP-5	1500	1500	600	800	60	5	2,5	745	4540	446
Athena 1c	ATK/Lockheed Martin (United States)	Kodiak, MARS, Spaceport Florida, Vanderberg	795	200	28,7	593	28,5				350	700	98			2057	2294				16	1994											
Athena 1c/bis	ATK/Lockheed Martin (United States)	Kodiak, MARS, Spaceport Florida, Vanderberg	470	200	90	593	28,5				450	370	98			2057	2294				16	1994											
Athena 2c	ATK/Lockheed Martin (United States)	Kodiak, MARS, Spaceport Florida, Vanderberg	1885	500	28,5	605	28,5									2743	4221				25	1998	ESPA	1500	1500	600	800		5	2,5			

Athena 2c	ATK/Lockheed Martin (United States)	Kodiak, MARS, Spaceport Florida, Vanderberg	1295	700	98	605	28,5									2743	4221				25	1998	ESPA	1500	1500	600	800		5	2,5			
Atlas V 401	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	12500	185	28,5	3765	28,5	1,5							Short	3750	4010				75	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 401	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	12500	185	28,5	3765	28,5	1,5							Medium	3750	4925	knownNam	3632	2460	75	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 401	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	12500	185	28,5	3765	28,5	1,5							Long	3750	5839	knownNam	3632	2460	75	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 411	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)				4535	28,5	1,5							Short	3750	4010				75	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 411	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)				4535	28,5	1,5							Medium	3750	4925	knownNam	3632	2460	75	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 411	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)				4535	28,5	1,5							Long	3750	5839	knownNam	3632	2460	75	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 421	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)				5255	28,5	1,5							Short	3750	4010				75	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 421	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)				5255	28,5	1,5							Medium	3750	4925	knownNam	3632	2460	75	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 421	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)				5255	28,5	1,5							Long	3750	5839	knownNam	3632	2460	75	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 431	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)				5885	28,5	1,5							Short	3750	4010				90	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 431	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)				5885	28,5	1,5							Medium	3750	4925	knownNam	3632	2460	90	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 431	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)				5885	28,5	1,5							Long	3750	5839	knownNam	3632	2460	90	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 501	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	10300	185	28,5	3000	28,5	1,5							Short	4570	4888				85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 501	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	10300	185	28,5	3000	28,5	1,5							Medium	4570	7631	knownNam	3632	2460	85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 501	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	10300	185	28,5	3000	28,5	1,5							Long	4570	10679	knownNam	3632	2460	85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 521	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	15080	185	28,5	4930	28,5	1,5							Short	4570	4888				85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 521	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	15080	185	28,5	4930	28,5	1,5							Medium	4570	7631	knownNam	3632	2460	85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 521	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	15080	185	28,5	4930	28,5	1,5							Long	4570	10679	knownNam	3632	2460	85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 531	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	17250	185	28,5	5645	28,5	1,5							Short	4570	4888				85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 531	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	17250	185	28,5	5645	28,5	1,5							Medium	4570	7631	knownNam	3632	2460	85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 531	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	17250	185	28,5	5645	28,5	1,5							Long	4570	10679	knownNam	3632	2460	85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 541	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	18960	185	28,5	6280	28,5	1,5							Short	4570	4888				85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 541	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	18960	185	28,5	6280	28,5	1,5							Medium	4570	7631	knownNam	3632	2460	85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 541	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	18960	185	28,5	6280	28,5	1,5							Long	4570	10679	knownNam	3632	2460	85	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5

Atlas V 551	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	20520	185	28,5	6695	28,5	1,5							Short	4570	4888				110	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 551	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	20520	185	28,5	6695	28,5	1,5							Medium	4570	7631	nknowNan	3632	2460	110	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
Atlas V 551	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	20520	185	28,5	6695	28,5	1,5							Long	4570	10679	nknowNan	3632	2460	110	2001	IPC/ESPA	1500	1500	600	965	60	5	2,5	833,2	2243	448,5
CZ-2C/SD	CALT(CHINA)	Taiyuan Satellite Launch Center(China)	2800	200	28,7	1250	28				1900	600	99		Type B	3070	2000				20	1990											
CZ-2C/SD	CALT(CHINA)	Taiyuan Satellite Launch Center(China)	750	200	98	1250	28				1900	600	99		Type Sd	3070	4000				20	1990											
CZ-3A	CALT(CHINA)	Taiyuan Satellite Launch Center(China)	6000	200	28,5	2600	28,5														55	2003											
CZ-3BE	CALT(CHINA)	Taiyuan Satellite Launch Center(China)	11500	200	28,5	5500	28,5				3100	600	99	2600		3650	4610				60	1992											
CZ-4B	CALT(CHINA)	Taiyuan Satellite Launch Center(China)	4200	200	60	1419	28,5				2800	900	99			3000	3900				30	1998											
Delta II 7320	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	2800	185	28,7						1652	883	98.7		Static Envelo 2,9	2184	3814				45	1995	ESPA			600	965			2,5			
Delta II 7321	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	2800	185	28,7						1652	883	98.7		Static evelop 3,0	2743	4163				45	1995	ESPA			600	965			2,5			
Delta II 7322	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	2070	185	90						1652	883	98.7		Stretched Composite	2743	5117				45	1995	ESPA			600	965			2,5			
Delta II 7920	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	5100	185	28,7						3186	883	98.7		Static Envelo 2,9	2184	3814				55	1995	ESPA			600	965			2,5			
Delta II 7921	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	5100	185	28,7						3186	883	98.7		Static evelop 3,0	2743	4163				55	1995	ESPA			600	965			2,5			
Delta II 7922	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	3830	185	90						3186	883	98.7		Stretched Composite	2743	5117				55	1995	ESPA			600	965			2,5			
Delta IV-M	Boing(United States)	Cape Canaveral, Vanderberg (United States)	9190	200	28,7	4210	28,5	1,8			7350	500	90			3750	6893				70	2001											
Delta IV-M	Boing(United States)	Cape Canaveral, Vanderberg (United States)	7690	200	90	4210	28,5	1,8			7350	500	90			3750	6893				70	2001											
Delta IV-M+(4,2)	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	12900	200	28,7	6160	28,5	1,8			10250	500	90			3750	6893				75	2001											
Delta IV-M+(4,2)	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	10530	200	90	6160	28,5	1,8			10250	500	90			3750	6893				75	2001											
Delta IV-M+(5,2)	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	11060	200	28,7	5080	28,5	1,8			8650	500	90			4572	8527				80	2001											
Delta IV-M+(5,2)	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	9610	200	90	5080	28,5	1,8			8650	500	90			4572	8527				80	2001											
Delta IV-M+(5,4)	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	13730	200	28,7	6890	28,5	1,8			11700	500	90			4572	8527				85	2001											
Delta IV-M+(5,4)	United Launch Alliance (United States)	Cape Canaveral, Vanderberg (United States)	11600	200	90	6890	28,5	1,8			11700	500	90			4572	8527				85	2001											

Dnepr	Yuzhmash (Ukraine)	Yasny(Russia)	3200	200	65											965	2560				20	2003									477,5	450	320
Epsilon	IHI Aerospace (Japan)	Uchinoura (Japan)	120	500	30						600	500	99																		25	90	215
Falcon 9 v1.1	Space X (United States)	Cape Canaveral, Vanderberg (United States)	10454	200	28,5	4850	27	1,8			8351	200	96.3	2900		1402	2011				61,2	2015	ESPA	1500	1500	600	965		5	2,5			
GLSV Mk.II	ISRO(India)	Satish Dhawan(India)	5000	200	45	2000	19,5				5000	407	51.6			3050	2771				35	2004											
GLSV Mk.III	ISRO(India)	Satish Dhawan(India)	8000	200	45	4000	19,5									3050	2771				45	2004											
H-IIA 202	Mitsubishi Heavy Industries (Japan)	LA-Y, Tenegashima (Japan)	10000	250	30	3800	30	1,8			3800	300	99	2500		3700	5800	unknow Name	3700	3800			ESPA	1500	1500	600	965		5	2,5			
H-IIB 304	Mitsubishi Heavy Industries (Japan)	LA-Y, Tenegashima (Japan)	16500	250	51,6	8000	51.6	1,8																									
Minotaur	Orbital(United States)	Vanderberg, MARS(United States)	640	185	28						335	740	98		Standard 50'	1194	1216				13	2001											
Minotaur	Orbital(United States)	Vanderberg, MARS(United States)	640	185	28						335	740	98		Optional 61'	1389	2088				13	2001											
PSLV-C	ISRO(India)	Satish Dhawan(India)	3700	200	49,5	1100	49.5				1600	647	99					unknow Name	2500	941	20	2001	unknow Name			600	800			2,5	500	920	308
PSLV-C	ISRO(India)	Satish Dhawan(India)	1600	800	98	1100	49.5				1600	647	99					unknow Name	2500	484	20	2001	unknow Name			600	800			2,5	500	920	308
PSLV-CA	ISRO(India)	Satish Dhawan(India)	1100	800	98						1100	647	99																		630	920	308
PSLV-XL	ISRO(India)	Satish Dhawan(India)	1800	800	98	1300	49,5				1750	647	99																		500	920	308
Pegasus	Orbital(United States)	Air launch to orbit	360	400	28,5											117	193				10	1996											
Proton-M	Khrunichev (Russia)	Baikonur (Russia)	20610	175	64,8	6150	25			2920				4800		4350	6000				85	2001									500	920	328
Rokot	Khrunichev (Russia)	Baikonur (Russia)	1950	200	63						1200	800	90			2100	3711				27	2013									650	1320	326
Strela	Khrunichev (Russia)	Baikonur (Russia)	1500	200	63						900	230	96.5		SHS-1	2200	2000				10,5	2003									181,25	300	200
Strela	Khrunichev (Russia)	Baikonur (Russia)	1500	200	63						900	230	96.5		SHS-2	1550	4000				10,5	2003											
Shavit	IAE(Israel)	Palmachim Airbase (Israel)	160	200	143																13	1995											
Soyuz-U	TsSKB-Progress (Russia)	Baikonur/Plesetsk (Russia)	7000	200	51,6						5800	200	97			3720	5370				25	2002	ASAP-5	1500	1400	800	1000		5	2,5	663,8	902	332
Soyuz-U/Fregat	TsSKB-Progress (Russia)	Baikonur/Plesetsk (Russia)	4500	200	97									1600		3600	5330	FREGAT	3600	4050	25	2002	ASAP-5	1500	1400	800	1000	60	5	2,5	663,8	902	332
Molniya-M	TsSKB-Progress (Russia)	Baikonur/Plesetsk (Russia)	2500	200	97											2300	1650				15	1996	ASAP-5	1500	1400	800	1000		5	2,5	663,8	902	332
Soyuz-FG	TsSKB-Progress (Russia)	Baikonur/Plesetsk (Russia)	7200	200	51,6	1100	7				3100	400	97			3600	5330	FREGAT	3600	4050	40	2001	ASAP-5	1500	1400	800	1000	60	5	2,5	663,8	902	332

Soyuz-2,1a	TsSKB-Progress (Russia)	Baikonur/Plesetsk (Russia)	4500	200	97	2780	7	1,5		1300	2780	820	98.7		Type S	3480	2254				40	2004	ASAP-5	1500	1400	800	1000		5	2,5	663,8	902	332
Soyuz-2,1a	TsSKB-Progress (Russia)	Baikonur/Plesetsk (Russia)	4500	200	97	2780	7	1,5		1300	2780	820	98.7		TYPE ST	3800	5070				40	2004	ASAP-5	1500	1400	800	1000		5	2,5	663,8	902	332
Soyuz-2,1b	TsSKB-Progress (Russia)	Baikonur/Plesetsk (Russia)	8500	200	51,6	3060	7	1,5		1480	3060	820	98.7		Type S	3480	2254				40	2004	ASAP-5	1500	1400	800	1000		5	2,5	663,8	902	332
Soyuz-2,1b	TsSKB-Progress (Russia)	Baikonur/Plesetsk (Russia)	8500	200	51,6	3060	7	1,5		1480	3060	820	98.7		TYPE ST	3800	5070				40	2004	ASAP-5	1500	1400	800	1000		5	2,5	663,8	902	332
Soyuz-2,1v	TsSKB-Progress (Russia)	Baikonur/Plesetsk (Russia)	1400	835	98,7																40	2004									663,8	902	332
Vega	ESA/ASI (Europe)	Kourou (French Guiana)	2300	300	0											2200	3515				23,5										137,5	147	315,5
Vega /bis	ESA/ASI (Europe)	Kourou (French Guiana)	1500	700	90											2200	3515				23,5										137,5	147	315,5
Vega/tris	ESA/ASI (Europe)	Kourou (French Guiana)	1100	1500	90											2200	3515				23,5										137,5	147	315,5
VEGA-VESPA	ESA/ASI (Europe)	Kourou (French Guiana)	1000	300	0											2000	1926	VESPA	2000	1000			VESPA	800	800	800	800	10	5	2,5	137,5	147	315,5
Zenit-2N	Yuzhnoye (Ukraine)	Baikonour (Russia)	13920	200	51,6											3400	6440				35	1993											
Zenit-3F	Yuzhnoye (Ukraine)	Baikonour (Russia)				3300	51,6	1477		1740						3400	6440				35	1993											
Zenit-3SL	Yuzhmash (Ukraine)	Baikonour (Russia)				6066	51,6	1477						4000		3400	6440				35	1993											
Zenit-3SLB	Yuzhmash (Ukraine)	Baikonour (Russia)	5000	200	51,6									3780		3400	6440				35	1993											

Type	Product name	Supplier	Inlet Max Pressure Range (bar)	Inlet Min Pressure Range (bar)	Thrust Max (N)	Thrust Min (N)	Maximum temperature of work (celsius)	Minimum temperature of work (celsius)	Max ISP (sec)	Min ISP (sec)	Minimum impulse bit (Ns)	Mass (Kg)	Preheating Power Nominal (W)	Power firing (W)	Pulses	Longhest continius fairing (s)	TRL	Propellant	Propellant operating temperature (celsius)	Cost w/o propellant (k€)	time pre-heating (m)	possible cold start	ITAR
Cold gas	2lbf cold gas	VACCO	17,9		8,89		65,5	-20	75			0,38		20	150000		9	ALL		25	0	YES	YES
Cold gas	3axis cold gas	VACCO	17,9		8,89		65,5	-20	75			1,41		20	150000		9	ALL		25	0	YES	YES
Cold gas	Low power cold gas	VACCO	20,6	0	5	0,5	50	17	42	4				30				ALL		20	0	YES	YES
Cold gas	SV14	MAROTTA	2,5		0,04	0,01	65	-35	70			0,075		3,5	2000000		9	GN2, Xe		15	0	YES	NO
Cold gas	Proportional Micro Thruster	BRADFORD	2,5	0	0,002	0	70	-45	65			0,175		4,5				Ghe,GN2,Xe, Dry air		15	0	YES	NO
Cold gas	MEMS micropopulsion	NANOSPACE	9	4	0,001	0,00001	50	0	100			0,115		2			9	Ghe,GN2,Xe, Dry air		10	0	YES	NO
Cold gas	P/N TG1	Omnidea-RTG	9	5,2	1	0,01			65,00			0,089		6	250000			ALL		10	0	YES	NO
Solid	Star 3	ATK			2050				266			1,16		1		0,62	9	TP-H-3498			0	YES	YES
Solid	Star 3-A	ATK			800				241,2			0,89		1		0,44	9	TP-H-3498			0	YES	YES
Solid	Star 5C	ATK			2023				268,1			4,47		1		2,8	9	TP-H-3062			0	YES	YES
Solid	Star 5CB	ATK			2188				262			4,5		1		2,67	9	TP-H-3237A			0	YES	YES
Solid	Star 8	ATK			4448				272			17,43		1		4,33	9	TP-H-3062			0	YES	YES
Green Propellant	HPGP (0,5N)	ECAPS	22	5,5	0,5	0,12	50	10	225	200	0,007	0,18		8	25000	30	5	LMP-103S	min 10 \max 50	60	30	NO	NO
Green Propellant	HPGP (1N)	ECAPS	22	5,5	1	0,25	50	10	235	204	0,07	0,38		8	60000	1,5h	8	LMP-103S	min 10 \max 50	67,5	30	NO	NO
Green Propellant	HPGP (5N)	ECAPS	24	5,5	5,5	1,5	50	10	253	239	0,25	0,4		15	10000	60	5	LMP-103S	min 10 \max 50	60	30	NO	NO

Green Propellant	HPGP (22N)	ECAPS	24	5,5	23	6	50	10	255	243	1	0,75		25	2600	120	5	LMP-103S	min 10 \max 50	81	30	NO	NO
Green Propellant	HPGP (50N)	ECAPS	26	5	50	12,5	50	10	255	243	2,5	1,5		50	160	10	3	LMP-103S	min 10 \max 50	120	30	NO	NO
Green Propellant	GR-1	Aerojet Rocketdyne	27,6	6,8	1,1	0,4	50	10	245		0,08	0,5		8	23000		8	LCH-240		250	30	NO	YES
Green Propellant	GR-22	Aerojet Rocketdyne	27,6	6,8	25	8	50	10	255		0,116	1		15	74000		8	LCH-240		320	30	NO	YES
Mono propellant	MONARC-1	Moog-Isp	22	4,8	1,42	0,38	40	5	227,5		0,35	0,35	34,9	54,9	357000	26,5	9	N2H4	min 5\max 40	50	40	YES	YES
Mono propellant	1N	Airbus	22	5,5	1,1	0,32			223	200	0,01	0,29		8	135000	12h	9	N2H4	min 5\max 40	45	40	YES	NO
Mono propellant	20N	Airbus	24	5,5	24,6	7,9			230	222	0,238	0,65		15	517000	10,5h	9	N2H4	min 5\max 40	60	40	YES	NO
Mono propellant	400N	Airbus	26	5,5	420	120			220	212	9	3,8		40	3900	450	9	N2H4	min 5\max 40	100	40	YES	NO
Mono propellant	MR-103D	Aerojet Rocketdyne	27,6	6,2	1,02	0,22			224	209	0,027	0,33		8	275028	111,4h	9	N2H4	min 5\max 40	145	40	YES	YES
Mono propellant	MR-111C	Aerojet Rocketdyne	27,6	5,5	5,3	1,3			229	215	0,08	0,33	1,54	8	420000	5000	9	N2H4	min 5\max 40	145	40	YES	YES
Mono propellant	MR-111E	Aerojet Rocketdyne	25,5	4,1	2,2	0,5			224	213	0,02	0,33	1,54	15	420000	15,5h	9	N2H4	min 5\max 40	160	40	YES	YES
Mono propellant	MR-106E	Aerojet Rocketdyne	24,1	6,9	30,7	11,6			235	229	0,46	0,635	3,27	6,36	12405	2000	9	N2H4	min 5\max 40	200	40	YES	YES
Mono propellant	MR-107S	Aerojet Rocketdyne	35	7	360	85			233	225		1,01		34,8	30300	41	8	N2H4	min 5\max 40	200	40	YES	YES
Mono propellant	MR-107t	Aerojet Rocketdyne	37	7	125	54			228	222		1,01		34,8	30275	100	7	N2H4	min 5\max 40	200	40	YES	YES
Mono propellant	MR-104A/C	Aerojet Rocketdyne	28,9	6,9	572,5	204,6			239	223	8,23	1,86		30	1742	2000	9	N2H4	min 5\max 40	250	40	YES	YES
Bipropellant	10N	Airbus	23	10	12,5	6			292			0,35		10		8h	9	N2H4 / Mon-1, Mon-3	min 5\max 40		0	No	NO
Bipropellant	200N	Airbus	17	7	270	180			270			1,9		40	320000	7600	9	N2H4 / Mon-1, Mon-3	min 5\max 40		0	No	NO

Bipropellant	400N	Airbus	18,5	12,5	450	340			321			4,3		80			9	N2H4 / Mon-1, Mon-3	min 5\max 40		0	No	NO
Bipropellant	R-6D	Aerojet Rocketdyne	27,96	6,9	22				294		0,0086	0,454		5	336331		9	MMH/MON-3			15		YES
Bipropellant	R-1E	Aerojet Rocketdyne	27,6	6,9	111				280		0,89	2		36	330000		9	MMH/MON-3					YES
Bipropellant	HIPAT	Aerojet Rocketdyne	27,6	6,9	445				320		35,6	5,44		46	500		9	MMH/MON-3					YES
Resistojet	XR-50	SITAEL			0,100				55			0,050		50			5	Xenon			15	YES	NO
Resistojet	XR-50	SITAEL			0,100				85			0,050		50			5	Argon			15	YES	NO
Resistojet	XR-100	SITAEL			0,125				63			0,150		80			5	Xenon			15	YES	NO
Resistojet	XR-100	SITAEL			0,125				105			0,150		80			5	Argon			15	YES	NO
Resistojet	XR-150	SITAEL			0,250	0,100			65	58		0,220		95			6	Xenon			15	YES	NO
Resistojet	XR-150	SITAEL			0,250	0,100			110	90		0,220		95			6	Argon			15	YES	NO
Resistojet	RS422	SST			0,050	0,020			42					95			9	Xenon			15	YES	NO
Resistojet	RS422	SST			0,050	0,020			100					95			9	Nitrogen			15	YES	NO
Resistojet	MR-502A	Aerojet	26,5	6,2	0,800	0,360			303	294	N/a	0,870		95		2h	9	N2H4			15	YES	YES
Arcjet	MR-510	AEROJET	18,6	13,8		0,222			615	585		1,58		95	1730	20h	9	N2H4			15	YES	YES
PPT	PPT	MARS SPACE					65	-20	1300		0,125	5	65	200				Teflon (solid bar)					NO
PPT	PRS -101	Aerojet							1350			4,78		100		3000	9	Teflon (solid bar)					YES
MPD	100-kW MPD	SITAEL			5	0,5			3000			18	20000	25000				Argon					NO

Ion Thruster	T5	QinetiQ	125	5	0,020	0,001			3500	500		1,7		600	8500		9	Xenon		910			NO
Ion Thruster	RIT µX	Airbus			0,0005	0,00005	55	-40	3000	300		0,44		50				Xenon		700			NO
Ion Thruster	RIT 10	Airbus			0,025	0,005	140	-75	3200	1900		1,8		760				Xenon		800			NO
Ion Thruster	RIT 2X	Airbus			0,200	0,080	190	-50	4300	3400		8,8		5785				Xenon		800			NO
Ion Thruster	BiT-1	Busek			0,0001				2150			0,053		10				Xenon		1200			YES
Ion Thruster	BiT-3	Busek			0,001				3500			0,2		60				Xenon		1200			YES
Ion Thruster	BiT-7	Busek			0,011				3500					360				Xenon		1200			YES
HET	HT-100	SITAEL		2,5	0,015	0,005			1350	1000		0,436	120	350			6	Xenon		700			NO
HET	HT-400	SITAEL		2,5	0,050	0,020			1850			900	250	800			5	Xenon		900			NO
HET	HT-5K	SITAEL		2,5	0,350	0,150			2800	1700		12,2	2500	7500				Xenon		1200			NO
HET	BPT-2000	Aerojet		2,5	0,123				1765			5,2		2200		6000h	7	Xenon					YES
HET	BPT-4000	Aerojet		2,5	0,290	0,132			1790	1676		12,3	2000	4500		10000h	8	Xenon					YES
HET	TMA (ALL integrated 2+2)	SECMA(SAFRAN)		2.5		0,083	58	-40	1500					1350			8	Xenon					NO
HET	BHT-200	Busek			0,013				1376			0,97		200				Xenon /Iodine					YES
HET	BHT-600	Busek			0,039				1585			2,2	300	800				Xenon /Iodine					YES
HET	BHT-1000	Busek			0,058				1750			3,5	500	1500				Xenon /Iodine					YES
HET	BHT-1500	Busek			0,101				1670				1000	3000				Xenon /Iodine					YES

HET	BHT-8000	Busek			0,507				1880				4000	12000				Xenon /Iodine					YES
HET	BHT-20k	Busek			0,807				2320				5000	20000				Xenon /Iodine					YES
Colloid thruster	1mN th	Busek			0,0001				1600			1,15		15			5						YES
Colloid thruster	LISA path FEEP	Busek			1E-06				4000			0,32		5				ion liquid			45		YES

Name	Supplier	Propellant	Volume(l)	Volume (m^3)	Pressure (bar)	Material	Type	Mass (Kg)	Price k€
PEPT-420	RAFAEL	M2H4	30	0,03	24	Titanium	Regulated System	3,5	77
PEPT-330	RAFAEL	M2H4	15,4	0,0154		Titanium	Regulated System	2,95	30
PEPT-260	RAFAEL	M2H4	6,9	0,0069		Titanium	Regulated System	1,7	15
270mm Xenon Tank (XTA)	RAFAEL	XENON	18,2	0,0182	150	Titanium	Regulated System	2,5	60
80389-1	ATK	M2H4	22,53	0,02253	21,33	Titanium	blowdown System	3,5429	415
80468-1	ATK	M2H4	22,53	0,02253	21,33	Titanium	blowdown System	3,72	415
80271-3	ATK	M2H4	24,91	0,02491	20	Titanium	blowdown System	5,17	416
80275-1	ATK	M2H4	32,04	0,03204	29	Titanium	blowdown System	5,76	418
80303-1	ATK	M2H4	32,17	0,03217	22,67	Titanium	blowdown System	5,9	418
80337-1	ATK	M2H4	28,27	0,02827	29	Titanium	blowdown System	6,35	417
80358-1	ATK	M2H4	32,2	0,0322	26,4	Titanium	blowdown System	5,9	418
80384-1	ATK	M2H4	32,2	0,0322	23,33	Titanium	blowdown System	5,9	418
80397-1	ATK	M2H4	30,15	0,03015	21,33	Titanium	blowdown System	4,54	417
80401-1	ATK	M2H4	33,58	0,03358	23,33	Titanium	blowdown System	6,99	418
80222-1	ATK	M2H4	4,7	0,0047	27,6	Titanium	blowdown System	1,3	410
80278-1	ATK	M2H4	4,7	0,0047	6,9	Titanium	blowdown System	1,5	410
80342-1	ATK	M2H4	14,5	0,0145	33,1	Titanium	blowdown System	2,7	413
80444-1	ATK	M2H4	10	0,01	23,4	Titanium	blowdown System	2,7	411
80216-1	ATK	M2H4	12,5	0,0125	27,3	Titanium	blowdown System	2,7	412

80290-1	ATK	M2H4	12,5	0,0125	27,3	Titanium	blowdown System	2,5	412
80225-1	ATK	M2H4	22,5	0,0225	22,1	Titanium	blowdown System	3,7	415
84486-1	ATK	M2H4	45,1	0,0451	26	Titanium	blowdown System	13,25	422
80273-7	ATK	M2H4	55,2	0,0552	29,6	Titanium	blowdown System	12,2	425
80259-1	ATK	M2H4	68,3	0,0683	25,9	Titanium	blowdown System	6,4	428
80298-1	ATK	M2H4	77,9	0,0779	33,4	Titanium	blowdown System	9,5	431
80562-1	ATK	M2H4	113,4	0,1134	38	Titanium	blowdown System	14,3	441
80514-1	ATK	M2H4	245,9	0,2459	28,4	Titanium	blowdown System	62	480
80386-1	ATK	XENON	32,1	0,0321	222	Titanium/polimer	Regulated System	6,4	500
80412-1	ATK	XENON	50	0,05	188	Titanium/polimer	Regulated System	7	540
8045-101	ATK	XENON	119,7	0,1197	285	Titanium/polimer	Regulated System	19,1	480
80263-101	ATK	M2H4	461,4	0,4614	23,3	Titanium	blowdown System	34,5	543
80263-201	ATK	M2H4	461,4	0,4614	24	Titanium	blowdown System	34,5	543
80318-1	ATK	M2H4	461,4	0,4614	23,3	Titanium	blowdown System	34,5	543
80370-1	ATK	M2H4	461,4	0,4614	23,3	Titanium	blowdown System	34,5	543
80376-1	ATK	M2H4	461,4	0,4614	23,3	Titanium	blowdown System	34,5	543
80382-1	ATK	M2H4	461,4	0,4614	21,7	Titanium	blowdown System	34,5	543
80407-1	ATK	M2H4	461,4	0,4614	21,7	Titanium	blowdown System	34,5	543
80463-1	ATK	M2H4	461,4	0,4614	23,3	Titanium	blowdown System	34,5	543
80485-1	ATK	M2H4	461,4	0,4614	23,3	Titanium	blowdown System	34,5	543

80487-1	ATK	M2H4	461,4	0,4614	23,3	Titanium	blowdown System	34,5	543
80515-1	ATK	M2H4	461,4	0,4614	23,3	Titanium	blowdown System	34,5	543
80516-1	ATK	M2H4	461,4	0,4614	23,3	Titanium	blowdown System	34,5	543
80557-1	ATK	M2H4	481	0,481	24,1	Titanium	blowdown System	33,6	548
80315-1	ATK	M2H4	600,4	0,6004	27,6	Titanium	blowdown System	44,5	583
80451-1	ATK	M2H4	623	0,623	28,4	Titanium	blowdown System	49,9	590
80523-1	ATK	M2H4	623	0,623	28,4	Titanium	blowdown System	49,9	590
6,4 inch RMD Tank	MOOG	M2H4	1,3	0,0013	48,26	Aluminium	blowdown System	0,73	15
9,6 inch RMD Tank	MOOG	M2H4	6,5	0,0065	45,85	Aluminium	blowdown System	1,50	30
12 inch RMD Tank	MOOG	M2H4	13,8	0,0138	27,58	Aluminium	blowdown System	3,22	60
18,2 inch RMD Tank	MOOG	M2H4	33,0	0,0330	22,41	Aluminium	blowdown System	7,67	80
31 inc	MOOG	M2H4	219,6	0,2196	26,06	Aluminium	blowdown System	27,26	90
BT 01/0	AIRBUS	M2H4	39,0	0,039	39	Titanium	Regulated System	8,50	40
T 11/0	AIRBUS	MMH/MON	174,4	0,1744	27,5	Titanium	blowdown System	11	140
OST 31/0	AIRBUS	N2H4	78	0,078	36,9	Titanium	blowdown System	6,4	125
OST 33/0	AIRBUS	N2H4	140	0,14	36,9	Titanium	blowdown System	13,5	130
OST 21/0	AIRBUS	MMH/MON	188	0,188	33	Titanium	blowdown System	16	145

Name	Supplier	Propellant	Volume(l)	Volume (m^3)	Pressure (bar)	Material	Type	Mass (Kg)	Price k€
80119-105	ATK	Helium	7,5	0,00749	41,4	Titanium	Pressure vassel	0,8	340,0
80186-1	ATK	Helium	28,7	0,02868	250	Titanium	Pressure vassel	10,6	361,4
80194-1	ATK	Helium	15,7	0,01565	248	Titanium	Pressure vassel	5,4	348,3
80195-1	ATK	Helium	9,4	0,00937	184	Titanium	Pressure vassel	5,4	341,9
80198-1	ATK	Helium	18,8	0,01880	250	Titanium	Pressure vassel	7,7	351,4
80202-1	ATK	Helium	14,5	0,01452	310	Titanium	Pressure vassel	7,2	347,1
80218-1	ATK	Helium	120,7	0,12069	234	Titanium	Pressure vassel	35,8	454,3
80221-1	ATK	Helium	88,2	0,08819	207	Titanium	Pressure vassel	24,9	421,5
80295-1	ATK	Helium	1,6	0,00164	552	Titanium	Pressure vassel	1,5	334,0
80314-1	ATK	Helium	36,1	0,03605	248	Titanium	Pressure vassel	15,9	368,9
80314-201	ATK	Helium	36,1	0,03605	248	Titanium	Pressure vassel	16	368,9
80326-1	ATK	Helium	3,9	0,00385	248	Titanium	Pressure vassel	1,5	336,4
80333-1	ATK	Helium	105,7	0,10570	280	Titanium	Pressure vassel	35,8	439,2
80345-1	ATK	Helium	6,6	0,00655	310	Titanium	Pressure vassel	3,4	339,1
80383-1	ATK	Helium	36,1	0,03605	248	Titanium	Pressure vassel	16	368,9

80448-1	ATK	Helium	31,2	0,03120	28,6	Titanium	Pressure vassel	2,2	363,9
80499-1	ATK	Helium	3,7	0,00374	248	Titanium	Pressure vassel	1,5	336,2
80509-1	ATK	Helium	116,4	0,11635	336	Titanium	Pressure vassel	23,1	450,0
80541-1	ATK	Helium	4	0,00385	248	Titanium	Pressure vassel	1,5	450
80386-101	ATK	Helium	32,1	0,032118511	222	composite	Pressure vassel	6,4	430,0
80400-1	ATK	Helium	67,3	0,067268616	388	composite	Pressure vassel	10	516,5
80402-1	ATK	Helium	67,3	0,067268616	388	composite	Pressure vassel	10	516,5
80412-1	ATK	Helium	50	0,05001311	188	composite	Pressure vassel	7	412,3
80436-1	ATK	Helium	81,4	0,081394206	414	composite	Pressure vassel	12,7	530,0
80445-1	ATK	Helium	67,3	0,067268616	388	composite	Pressure vassel	10,5	516,5
80446-1	ATK	Helium	67,3	0,067268616	388	composite	Pressure vassel	10,7	516,5
80458-1	ATK	Helium	132,8	0,132734662	248	composite	Pressure vassel	20,4	443,5
80458-101	ATK	Helium	119,7	0,119625066	248	composite	Pressure vassel	19,1	443,5
80458-201	ATK	Helium	54,1	0,054077084	248	composite	Pressure vassel	12,2	443,5
80459-1	ATK	Helium	67,3	0,067268616	388	composite	Pressure vassel	10,7	516,5
80465-1	ATK	Helium	81,4	0,081394206	414	composite	Pressure vassel	12,7	530,0

80475-1	ATK	Helium	87	0,086998558	388	composite	Pressure vassel	16,8	516,5
80492-1	ATK	Helium	67,3	0,067268616	388	composite	Pressure vassel	10,7	516,5
80496-1	ATK	Helium	81,4	0,081394206	414	composite	Pressure vassel	12,7	530,0
80525-1	ATK	Helium	67,3	0,067268616	388	composite	Pressure vassel	10,7	516,5
80530-1	ATK	Helium	81,4	0,081394206	414	composite	Pressure vassel	12,7	530,0
80536-1	ATK	Helium	67,3	0,067268616	388	composite	Pressure vassel	10,7	516,5
80548-1	ATK	Helium	51,4	0,051406004	387	composite	Pressure vassel	12,5	515,9
80XXX	ATK	Helium	35	0,034986235	388	composite	Pressure vassel	8	516,5

PRESSURE TRANSDUCERS												
Name	Supplier	Compatibility	Dimension d*h (mm)	Max pressure (bar)	Min pressure (bar)	Mass (kg)	Max operating temperature (°C)	Min operating temperature (°C)	Power consuption (W)	Otput signal (V)	Price (K€)	interface (cable)
Mini Press. Transducer	Moog	N2H4,MON,MMH,GP,HE,Xe	43*84,6	320	0	0,125	70	-20	0,3	5	15	4
PRESSURE TRANSDUCERS	Moog	N2H4,MON,MMH,GP,HE,Xe	43,5*126	320	0	0,23	70	-20	0,3	5	15	4
P2911	TABER		27*89	1000	0		76	-34		10	15	4
PYROVALVE												
Name	Supplier	Compatibility	Dimension d*h (mm)	Response time (s)		Mass (Kg)	Max operating temperature (°C)	Min operating temperature (°C)	Pressure qualification (bar)			4
Pyrovalve	Airbus	N2H4,MON,MMH,GP,HE,Xe		0,007		0,160	100	-90	310			4
Pyrotechnic valve	Dassault		90*104	0,010		0,400	70	-20				4
Pyrovalve	Cobham											4
FILL AND DRAIN VALVE												
Name	Supplier	Compatibility	Dimension d*h (mm)	Max operating temperature (°C)	Min operating temperature (°C)	Mass (s)	Operating cycles	Pressure qualification (bar)	TRL			
Fill & Drain valve	Airbus	Propellant and Pressurants	6,4*109			0,09		33				
Fuel Fill Valve	Airbus	MMH	6,4*108	80	-30	0,09	40		9			
Fuel Vent Valve	Airbus	MMH	6,4*107	80	-30	0,09	40		9			
High pressure Helium Valve	Airbus	He	6,4*94,5	80	-30	0,06	40		9			
Low Pressure Helium Valve	Airbus	He	6,4*94,5	80	-30	0,06	40		9			
Oxidiser Fill valve	Airbus	MON	6,4*108	80	-30	0,09	40		9			
Oxidiser Vent valve	Airbus	MON	6,4*107	80	-30	0,09	40		9			
High pressure Xenon Valve	Airbus	XENON	6,4*115	80	-30	0,06	40		9			
1/2'' Low Pressure	VACCO	N2H4	41,3*118	32	7	0,68	150	25	9			
1/4'' Low pressure	VACCO	GN2,N2H4	28*127	32	7	0,275	150	25	9			
1/2'' Low Pressure II	VACCO	MMH,N2O4,IPA,H20	28*101	55	-7	0,113	100	17,24	9			
1/4'' High pressure	VACCO	GHe, GN2	28*986	55	-35	0,113	100	310	9			
1/8'' High pressure micro	VACCO	GHe, GN2, Argon	14,27*70	60	-34	0,030	50	690	9			
3/8'' High pressure micro	VACCO	Ghe,GH2,GO2, NTO,MMH	11,18*35,6	30	10	0,010	50	460	9			
Latch Valves												
Name	Supplier	Compatibility	Tube interface (inch)	Response time(s)	Power consuption (W)	Mass (kg)	Operating cycles	Max operating temperature (°C)	Min operating temperature (°C)		20	7
Latch Valves	Airbus	ALL	1/4	0,03	50	0,545	500	50	9		20	7

SALV	Moog	MMH, MON, H2N4, IPA, GN2, GHe	1/4	0,03	50	0,750		65	-5		20	7
400N APOGEE ENGINE FLOW CONTROL VALVE	Airbus	N2H4		0,03	38,4		5000	115	0		40	7
REGULATOR FEED SYSTEM												
Name	Supplier	Compatibility	Max Pressure (bar)	Min Pressure (bar)	Outlet Pressure(bar)	Mass (Kg)	Dimesion (mm)	Max operating temperature (°C)	Min operating temperature (°C)	Power consuption (W)		
XRFS	Airbus	XENON	120	3	2,65	5,9	443,5*278*228	45	27	0		
B4035-1	Cobham	N2O4, N2H4	275	27,5	16	0,98	127*89*74	-45,6	48,9	0		
Flow control unit												
Name	Supplier	Compatibility	Max Pressure (bar)	Min Pressure (bar)	Outlet Pressure(bar)	Mass (Kg)	Dimesion (mm)	Max operating temperature (°C)	Min operating temperature (°C)	Power consuption (W)		
Flow control unit	Moog	XENON, Argon	2,85	2,5	0,300	1,16	184*144*76	65	-30	0		
Xenon flow control module	VACCO	Xenon	206	0,69	3,45	1,4	191*69*47	70	0	0		
Power Processing unit												
Name	Supplier			Max operating temeperature (C°)	Max operating temeperature (°C)	Mass (kg)	Dimension (mm)	Peak transient consuption (W)	Power output (W)	Power consuption (W)		
CAMELIA Power Processing Unit	RAFAEL					7	200*150*220	125				
CAMELIA Propellant Management Assembly	RAFAEL					1						
T5 Reduced PPU	CRISA					12		15				
T5 IPCU (GOCE)	CRISA			50	-20	12	300*250*200	15	700	150	250	
Power Processing Unit (PPU) for 5kW Hall Effect Thrusters	Sitael					13	350*250*200		5000	1071	500	
Power Processing Unit (PPU) for Hall Effect Thrusters	Thales			70	-25	10,4	390*190*186					
Mr-510 Arcjet Power conditioning unit	Aerojet					15,8	632*361*109					
4,5 KW HALL Thruster power process	Aerojet					12,5	430*400*110		4000			
Power Processing Unit for Electric Propulsion	Airbus			50	-20	17,5	380*270*205		770	150	500	
Empty row	to use for assign	chemical	0	0	0	0	0	0	0	0	0	0

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