Master Thesis
Conceptual Lay-out of a Small Launcher

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Abstract

The objective of this diploma thesis is to perform a conceptual lay-out of a small launcher. Requirements have been defined in order to realize this first preliminary study and design of a small launcher. In that frame, a MATLAB code has been written in order to simulate the rocket trajectories. An optimization program on launcher staging has been written as well. To validate this code, the VEGA and Ariane 5 launchers have been used. Then from studies on existing launchers, simulations have been performed in order to find an optimum small launcher and later on to design more precisely the small launcher. As a requirement an upper stage has been newly designed for the purpose of the study. At the end, two small launchers have been considered: a three-stage launcher using the Zefiro 23 as a first stage, the Zefiro 9 as a second stage, and an upper stage using a 3kN thrust engine; a two-stage launcher using the Zenit booster engine in the first stage, and an upper stage using a 22kN thrust engine.

This paper presents the method used to develop the small launcher and the results.

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Nomenclature

a	Semi-major axis of the ellipse	$[m]$
a_{x_b}	Longitudinal acceleration	$[m/s^2]$
a_{y_b}	Lateral acceleeration	$[m/s^2]$
a_g	Gravitational acceleration	$[m/s^2]$
d_{sec}	Section diameter	$[m]$
e	Ellipse eccentricity	$[-]$
g	Gravitational acceleration	$[m/s^2]$
g_0	Gravitational acceleration at sea level	$[m/s^2]$
h	Specific angular momentum	$[m^2/s]$
inc	Inclination angle of the orbit	$[deg]$
k_{ax}	Max. axial acceleration	$[-]$
k_{lat}	Max. lateral acceleration	$[-]$
l_{sec}	Section length	$[m]$
m	Spacecraft mass	$[kg]$
m_{dry}	Spacecraft dry mass	$[kg]$
m_{engine}	Engine mass	$[kg]$
m_i	Initial spacecraft mass	$[kg]$
m_f	Final spacecraft mass	$[kg]$
m_{fuel}	Fuel mass	$[kg]$
m_{fair}	Fairing mass	$[kg]$
m_{ox}	Oxidizer mass	$[kg]$
m_p	Propellant mass	$[kg]$
$\dot{m}$	Mass flow rate	$[kg/s]$
$n_{F_{ax}}$	Axial load per perimeter	$[N/m]$
n_M	Bending moment per area	$[N/m]$
p	Pressure	$[Pa]$
r	Radius	$[m]$
r_a	Apogee radius	$[m]$
r_p	Perigee radius	$[m]$
t_1	Gravity turn beginning time	$[s]$
t_2	Gravity turn ending time	$[s]$
t_b	Burning time	$[s]$
v	Spacecraft velocity	$[m/s]$
v_{circ}	Circular orbit speed	$[m/s]$
v_e	Exhaust velocity	$[m/s]$
v_{ell}	Elliptic orbit speed	$[m/s]$
v_x	Longitudinal velocity	[m/s]
v_y	Lateral velocity	[m/s]
C_l	Lift coefficient	$[-]$

C_d	Drag coefficient	$[-]$
D	Drag	$[N]$
E	Young modulus	$[N/m^2]$
F	Thrust	$[N]$
G	Gravitational constant	$[m^3/(kg\,s^2)]$
H	Altitude	$[m]$
I_{sp}	Specific impulse	$[s]$
L	Lift	$[N]$
M	Earth mass	$[kg]$
MR	Mixture ratio	$[-]$
R_E	Earth Radius	$[m]$
R	Gas perfect constant	$[J/(K\,mol)]$
S	Reference area	$[m^2]$
SF	Safety factor	$[-]$
SI	Structural index	$[-]$
ΔV	Velocity change	$[m/s]$
V	Volume	$[m^3]$
V_{ull}	Ullage volume	$[m^3]$
T	Temperature	$[K]$
X	Ground range	$[m]$
α	Angle of attack	$[deg]$
ϵ	Specific orbital energy	$[-]$
μ	Gravitational parameter	$[m^3/s^2]$
γ	Flight path angle	$[deg]$
$\dot{\gamma}_0$	gravity turn rate	$[deg/s]$
ρ	Air density	$[kg/m^3]$
ρ_0	Air density at sea level	$[kg/m^3]$
σ	Yield stress	$[N/m^2]$
θ	Pitch angle	$[deg]$

Indices

1	Stage 1
2	Stage 2
3	Stage 3
cyl	Cylinder tank
$dome$	Tank dome
f	Fuel
$fair$	Fairing
He	Helium
ox	Oxidizer
$prop$	Propellant
sec	Section
sph	Spherical tank
ull	Ullage

Abbreviations

BDVM	burst disk passivation MMH
BDVN	burst disk passivation NTO
BDM	burst disk MMH
BDN	burst disk NTO
BM	branching manifold with filter
CRVH	fill and drain valve Helium
CRVM	fill and drain valve MMH
CRVN	fill and drain valve NTO
CVM	check valve MMH
CVN	check valve NTO
ETF	engine thrust frame
FCV	flow control valve
FDV1	pressurant fill and drain valve
FDV2	propellant fill and drain valve
FDV3	feedline fill and drain valve
FH	helium filter
HVO	Helium Venting Orifice
ISPR	I stage pressure regulator
IISPR	II stage pressure regulator
LVM	Latch valve MMH
LVN	Latch valve NTO
MEOP	maximum expected operating pressure
MMH	monomethylhydrazine
NTO	dinitrogen tetroxide
PO	priming orifice
PT	redundant pressure transducers
PV1	priming pyrovalve
PV2	mainflow pyrovalve
PV3	depletion pressure transducers
PVAM	pressure valve assembly (MMH)
PVAN	pressure valve assembly (NTO)
PVH	pyrovalve Helium
PVM	pyrovalve MMH
PVN	pyrovalve NTO
RACS	roll and attitude control system
RDCM	vent valve MMH
RDCN	vent valve NTO
TCA	thrust chamber assembly
TCM	thruster cluster module
TH	helium tank
TM	MMH tank

TN	NTO tank
TSPM	passivation exhaust device (MMH)
TSPN	passivation exhaust device (NTO)

Chapter 1

Introduction

Miniaturized satellites represent the current forefront of space technology. Indeed, the progress in the area of structure and electronics has led to smaller satellites and thereby reduced the size requirements of launch vehicles. Nowadays, the current launchers are able to launch several satellites at once. For instance, the Ariane 5 payload is usually made up of two or three satellites. When a company wishes to set a satellite in orbit, this company has to wait for another company which would like to set a satellite as well; if not, the launch would be too expensive. Thereby, the time schedule is a constraint and so the need to develop a small launcher stands to reason. The development of a small launcher will produce more flexibility, reactivity and will reduce costs as well. Furthermore, the trend seen in small launcher augmentation is to develop a small launcher based on existing stage in order to use existing technologies and thereby to reduce the cost of a new development.

In that frame, the future development of a small launcher will reduce the launch cost and allow time flexibility as well. Therefore, the objective of this master thesis is to do the conceptual lay-out of a small launcher. A conceptual lay-out of a new small launcher includes several aspects: a performance aspect which includes trajectory simulations, trajectory optimization and launcher staging optimization, and a design aspect which includes stage preliminary design. The performance part in this master thesis is not based on a previous work, and so a special tool to simulate and optimize the trajectory has been developed. From the launcher characteristics, namely the thrust, specific impulse, propellant and structural mass, and by resolving the equations of motions of a rocket in a later on defined frame of reference, the amount of payload this launcher can carry to a certain orbit can be found. That is why a special program has been developed to study the launcher performances. Later, the design of the possible stages is based on work done by a student during his thesis [3]. In order to design a new stage, propellant and pressure gas budgets have to be carried out as well as structural calculations. Those budgets and calculations are based on the previous thesis, however improvements and changes have been made for the purpose of this thesis.

This paper presents the conceptual lay-out of two small launchers: a three-stage launcher with two solid stages and one liquid upper-stage, and a two-stage launcher, both containing liquid fuel.

Chapter 2

Small Launcher Basic Conditions

This chapter summarizes the mission and the requirements of the small launcher. The method which has been set up in order to design the conceptual lay-out is also described.

2.1 Mission and Requirements

The objective of this study is to develop a conceptual layout of a small launcher. This small launcher is supposed to deliver a payload to a circular orbit around the Earth. The lift-off location of this small launcher is Kourou in French Guyana. In order to reach the targeted orbit, the small launcher will first reach a transfer elliptical orbit and then a circularization is achieved thanks to a re-ignition of the upper stage. The mission has to take into account the satellite de-orbiting at life end. That leads to another re-ignition for de-orbiting the satellite. The mission is schematized on the figure 2.1, the possible de-orbiting is not represented.

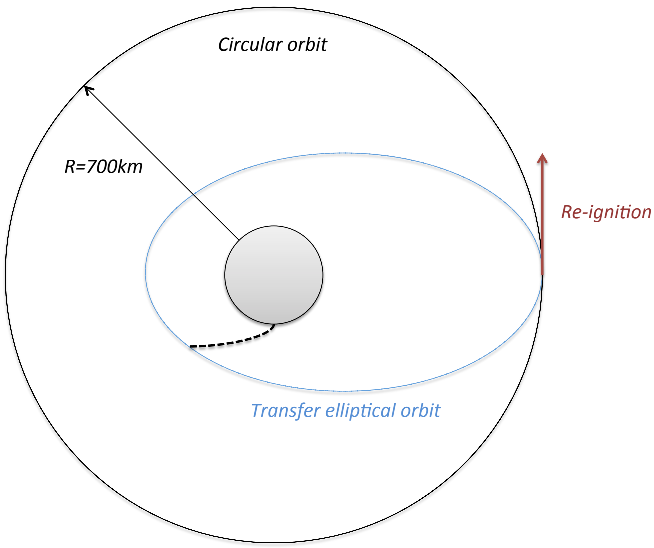


Figure 2.1: Small launcher mission

For this study, several requirements have been imposed:

- The payload mass is between 100 and 250kg.
- The orbit is a circular polar orbit with a radius of 700km and an inclination of 90 degrees.
- An elliptical transfer orbit is reached first, and then a circularization is carried out.
- The small launcher can have 2 or 3 stages.
- The upper stage has to be designed with re-ignitable engines using liquid propulsion developed by Astrium.
- The stages have to be existing ones or designed by using an engine developed by Astrium.

The launcher can be composed of solid and liquid stages, several configurations are therefore possible:

- **2 stages:** 1 liquid booster + 1 liquid upper stage
- **3 stages:** 1solid booster + 1 solid booster + 1 liquid upper stage
- **3 stages:** Several solid boosters + 1 solid booster + 1 liquid upper stage

These configurations have been represented in the figure 2.2.

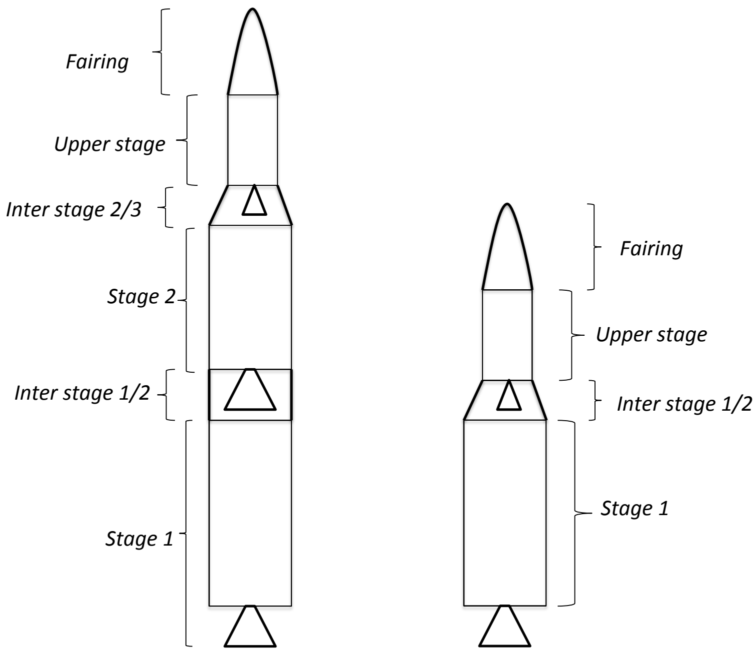


Figure 2.2: Small launcher configurations

In order to perform the conceptual lay-out of the small launcher, a performance study as well as a design study have to be carried out. Indeed, a performance study is necessary in order to determine, depending on the launcher characteristics, the amount of payload this launcher can carry to put in a certain orbit. So by setting the launcher characteristics, namely its thrust, specific impulse and propellant and structural mass, and resolving the equations of motions in a later on defined frame of reference, it is possible to get the speed, the altitude, and the flight path angle of the launcher at the complete burn-out. From these three values, the apogee and perigee

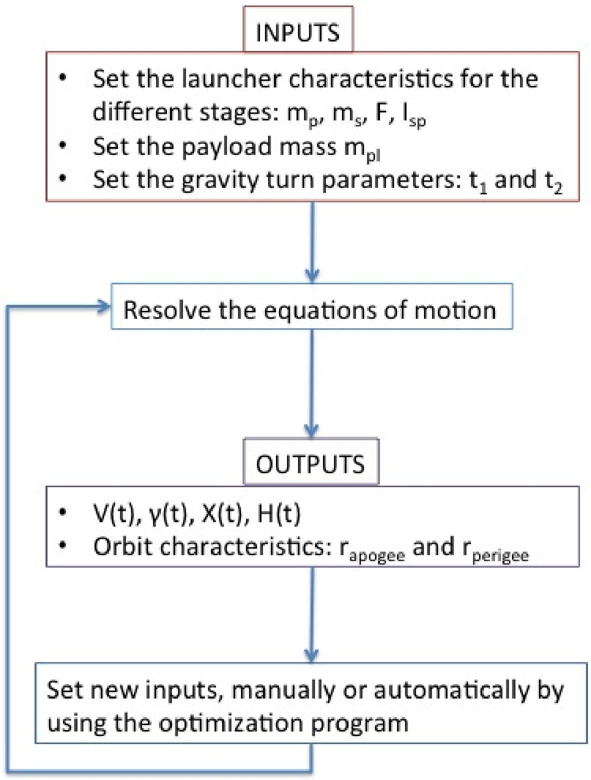


Figure 2.3: A basic performance calculation loop

of the orbit can be calculated. At complete burnout, the spacecraft has to have a flight path angle equal to zero and the right altitude and speed needed to stay in a certain orbit. The rocket is launch vertically, then at time t_1 the rocket is turned by a maneuver called the gravity turn trajectory which will be defined later on, until a time t_2, and then the rocket turns automatically by itself until burn-out. By changing these two times, the trajectory changes as well. So, if the launcher characteristics are fixed, the performances of the launcher can be optimized by changing these two times and the payload. The figure 2.3 illustrates this performance calculation.

However, the purpose of this study is to do the conceptual lay-out of a small launcher, and so to define all the characteristics of this launcher. Therefore, the characteristics are variables as well as t_1 and t_2. However, if this study had to consider all the variables (thrust, specific impulse, propellant mass, structural mass, t_1 and t_2) as independent variables, the problem would be very complicated and would take a lot of time. In this study, one requirement is to use existing stages. Therefore, the launcher characteristics can not take all the possible values, furthermore, as the characteristics of the existing stages are known it is possible to relate some variables to one variable. Indeed, by collecting existing launcher characteristics it is possible to obtain the variation of the structural index, the specific impulse and the thrust as a function of the propellant mass. By setting the propellant mass, the three others characteristics are automatically calculated. In that case, the launcher performances can be optimized by changing the amount of propellant and the times t_1 and t_2. The figure 2.4 illustrates this second performance calculations.

It is important to notice that only the upper stage has its variables completely altered. The upper stage characteristics have to be chosen in order to fulfill all the requirements. Therefore, design of this upper stage needs to be carefully considered. The amount of propellant, the thrust and the specific impulse will be chosen based on the performance study. In order to perform the design, a propellant and gas pressure budgets will be evaluated as well as structural calcula-

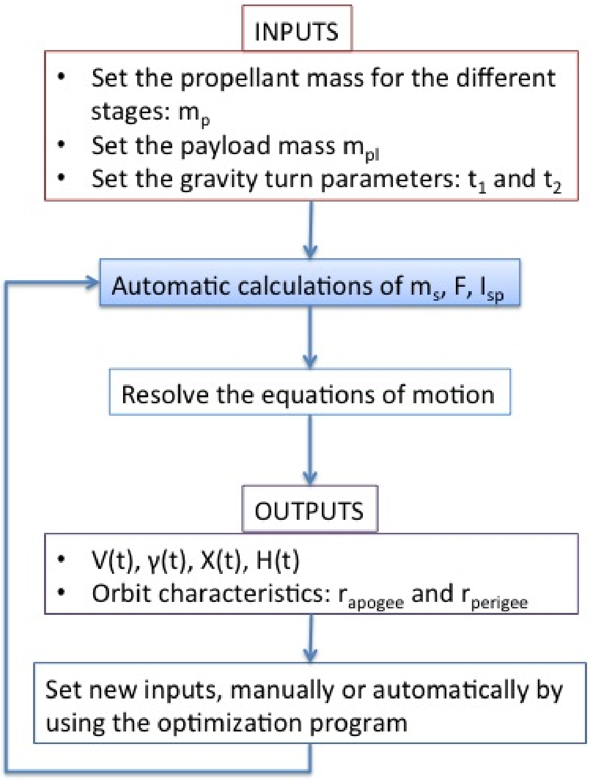


Figure 2.4: A second performance calculation loop

tions. To perform these budgets and calculate the structural mass, an existing tool will be used. Improvements will be made for the purpose of this study.

2.2 Method

This section explains the approach which has been decided in order to develop the small launcher.

2.2.1 Stage research

Research has been done on existing solid and liquid stages. A requirement in the development of the small launcher is that it must use existing liquid or solid stages. The research has been focused on stages with not too large a thrust value. From the research, the goal was to obtain the optimum curves, e.g. the thrust, the specific impulse and the structure index as a function of the propellant mass, as can be seen on the figure 2.5.
The structural index is defined as the ratio between the structural mass and the total mass.

$$SI = \frac{m_s}{m_p + m_s} \tag{2.1}$$

where:	m_p	propellant mass $[kg]$
	m_s	structural mass $[kg]$
	SI	structural index $[-]$

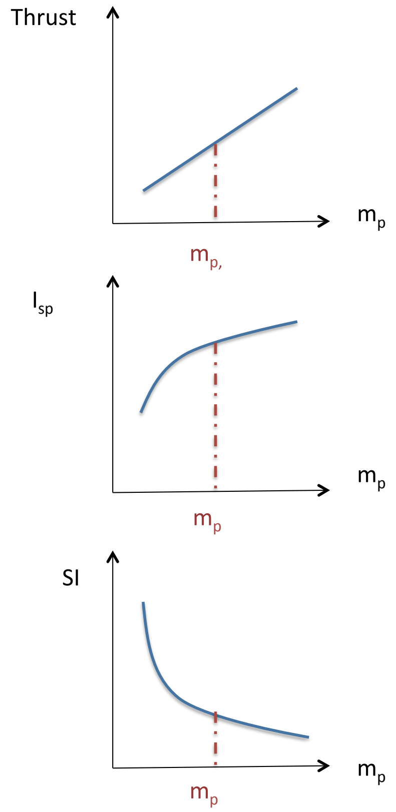


Figure 2.5: Optimum curves

2.2.2 Upper stage design

As can be seen from the requirements, the upper-stage has to be designed with a liquid propulsion engine developed by Astrium. So the second part of this study is to do a preliminary design on the upper stage in order to run simulations later on. The design of the upper stage includes a propellant budget, a helium budget and structural mass calculations. Choices concerning the tank configuration, the material for the stage structure and tank structure, and the systems have to be taken. This engine with liquid propulsion is still under investigation at Astrium and it is not yet commercially available, so some engine characteristics are still flexible. In that frame, the thrust is considered flexible and will be chosen based on the performance calculations.

The engine mass is directly linked to the thrust, whereas the stage structural mass is linked to the propellant mass of the stage. As the propellant mass and the thrust are not fixed and will be optimized, the structural mass is separated in two parts:

$$m_{dry,upperstage} = m_{dry,w/o\,engine}(m_{prop}) + m_{engine}(F) \tag{2.2}$$

- The structural mass, engine mass excluded, which depends on the propellant mass.
- The engine mass which depends on the the thrust engine.

In order to choose the characteristics of the upper stage in the small launcher, an optimization will be done on the amount of propellant and on the thrust.

Engine

The liquid propulsion engine in development at Astrium is a re-ignitable engine and uses the propellant combination of MMH (monomethylhydrazine) and NTO (nitrogen tetroxide). Several engines using these propellants are under investigation, and more precisely, studies with 3kN, 8kN, 16kN, 22kN, 30kN and 50kN thrust engines have been done. From these studies, it is possible to relate the engine mass to the thrust.

Structure

At first, two different upper stages have been designed in order to determine the dependency between the structural mass and the propellant mass, and so to be able to run some simulations. These designs are a first estimation and have been done with margins, but at the end, the final upper stage has been designed in more detail. To design the upper stage, an existing Excel tool was available, some modifications have been done in order to improve the tool and will be explained later on.

2.2.3 Matlab code

A MATLAB code has been developed in order to simulate rocket trajectories by resolving numerically the equations of motion. This program enables the performance calculations to be done. By changing the variables it is possible to change the performance of the launcher. Later on, an algorithm is used in order to optimize the variables.

2.2.4 Small launcher

Then, simulations will be run in order to find an optimum launcher which meets all the requirements. This launcher is called optimum launcher because the stage characteristics are directly calculated from the optimum curves as defined previously, i.e. by choosing the amount of propellant, the thrust, specific impulse and structural index are automatically defined. The upper stage characteristics depend on the amount of propellant and the engine thrust. Two different methods will be used depending on the characteristics of the launcher.

First configuration: two solid stages and a liquid upper stage

Simulations have been done for a three-stage launcher. The two first stages are solid boosters and the characteristics of these boosters are calculated thanks to the optimum curves represented in Figure 2.5. Thus, the thrust, the specific impulse and the structural index are functions of the propellant mass in the stage. By changing the propellant mass in one of the two stages the characteristics of this stage are automatically recalculated. The thrust and the propellant mass of the upper stage are inputs and need to be optimized.

The purpose is to find optimum launchers by using the optimum curves, in order to see if the optimum launchers converge in the same direction. So, different amount of propellant for each stage will be investigated. Only the amount of propellant is relevant, all the characteristics of the stages are calculated thanks to the propellant mass of the considered stage. The different amounts of propellant investigated for each stage are written in the table 2.1. In the first stage, the propellant mass will be considered between 22,000 and 26,000kg, in the second stage, the propellant mass will be considered between 10,000kg and 12,000kg, in the third stage the propellant mass will be considered between 500kg and 1500kg, and the thrust, between 5kN and 15kN. That leads to the study of 81 possible small launchers.

$m_{p,1}$ $[kg]$	22,000	24,000	26,000
$m_{p,2}$ $[kg]$	10,000	11,000	12,000
$m_{p,3}$ $[kg]$	500	1,000	1,500
T_3 $[N]$	5,000	10,000	15,000

Table 2.1: Possible configurations

In order to clarify, the method is schematized in Figure 2.6 as a tree. Each branch of the tree will be investigated and an optimum small launcher will be found from these initial characteristics by using the optimization program.

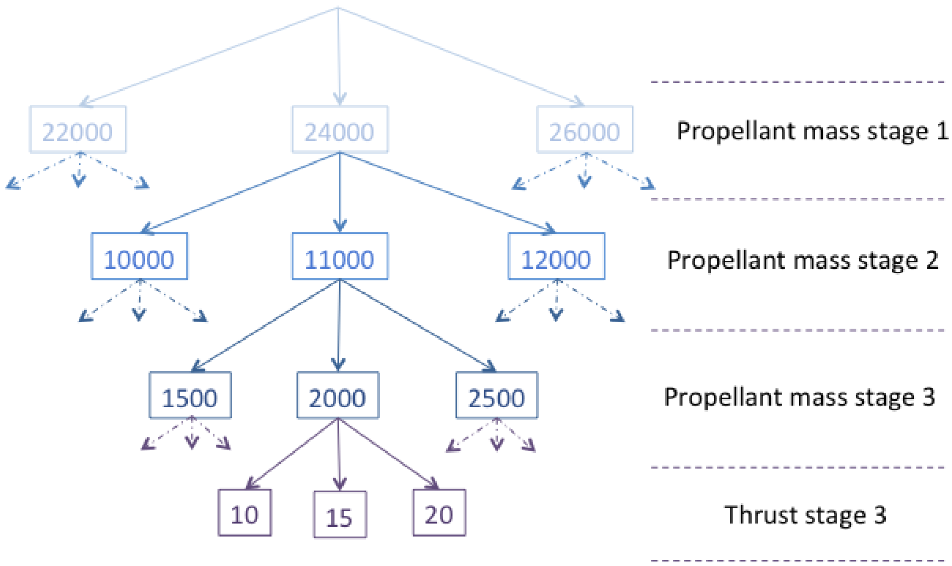


Figure 2.6: Method to find optimum small launcher

Finally, from the optimum launchers, the purpose is to find the existing boosters which are closest to the optimum results and so to focus only on a maximum of three boosters. Then, new simulations have to be run in order to choose the existing boosters and to design the upper stage of the launcher in more detail.

Second configuration: a liquid stage and a liquid upper stage

Simulations will also be done for a two-stage launcher with two liquid stages. The method used to design this launcher is different from the one used with the three-stage launcher. Indeed, the existing liquid stages are either too big or they do not have enough thrust to lift the launcher. So the liquid stages need to be modified. This modification will be made on the stage structure. So, the method used is to pick first a liquid stage, to keep the thrust and the specific impulse as constants and to calculate the structural mass as a function of the propellant mass. So, for the liquid stage, only the optimum curve concerning the structural index is relevant in this study.

By reducing the amount of propellant in the stage, and so by reducing the structural mass, the launcher is capable of lift off. The purpose is to optimize the amount of propellant in the liquid stage whilst also achieving optimization of the upper stage (propellant mass and thrust).

Chapter 3

Calculation Methods

The objective of this thesis is to perform a conceptual lay-out of a small launcher. The method used to perform the conceptual-lay-out has been explained in the previous chapter but in order to understand, some theoretical fundamentals need to be defined and explained.
The following chapter describes the theoretical fundamentals needed to perform a conceptual lay-out.

3.1 Flight Physics

As explained previously, the performance calculations are done by resolving the equations of motion, and so these equations of motion and the frame of reference need to be introduced. The following part explains the flight physics which is essential to understand how the equations of motion are solved.

The forces acting on a rocket are the thrust, the aerodynamic forces, the gravitational attractions, the wind and solar radiation pressure [1]. The two last forces are usually small and are therfefore neglected. In this study, the rocket trajectory is considered two dimensional. This assumption has been made because the tool developed to simulate the rocket trajectory has to be relatively simple and fast. Also this tool is used for making preliminary estimates and so a two dimensional problem is enough for this study. Thus, the trajectory is two dimensional and is contained in a fixed plane. The flight is assumed over a non-rotating spherical Earth [6].

The body-fixed coordinate system is the frame of reference of the rocket. The x-axis corresponds to the longitudinal axis of the rocket, the y-axis is perpendicular to the x-axis in the lateral plane. This frame of reference is represented on the Figure 3.1, the forces acting on the rocket are also represented, namely the drag D, the lift L, the weight $m\,g$ and the thrust T.In Figure 3.1, α is the angle of attack, it is the angle between the direction of flight and the thrust direction; θ is the pitch angle, it is the angle between the horizontal reference and the thrust direction. The sum of these two angles is called the flight path angle γ.

The linear velocity due to the Earth rotation (in the orbital plane) is calculated for the start location from its latitude and orbit inclination.

3.1.1 Newtons equations of motion

The Newtons equations of motion considered in the body-fixed axis system are written below.

$$m\,a_{x_b} = -D\,\cos\alpha - L\,\sin\alpha + F\,\cos\alpha - m\,g\,\sin\theta \tag{3.1}$$

$$m\,a_{y_b} = -D\,\sin\alpha + L\,\cos\alpha - F\,\sin\alpha - m\,g\,\cos\theta \tag{3.2}$$

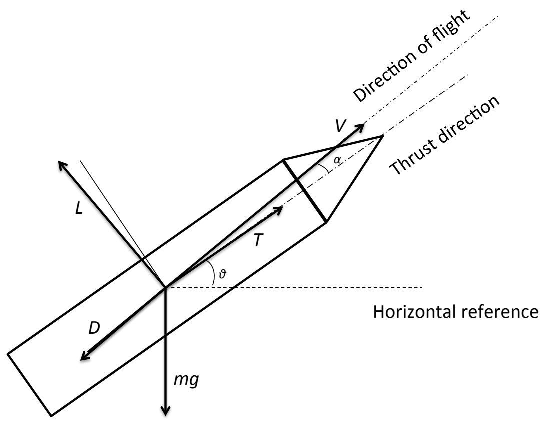


Figure 3.1: Frame of reference and forces acting on the rocket

where:	a_{x_b}	acceleration projected on the rocket's longitudinal axis $[m/s^2]$
	a_{y_b}	acceleration projected on the rocket's lateral axis $[m/s^2]$
	D	drag $[N]$
	F	thrust $[N]$
	g	gravitational acceleration $[m/s^2]$
	L	lift $[N]$
	m	mass $[kg]$
	α	angle of attack $[deg]$
	θ	pitch angle $[deg]$

The flight path angle is

$$\gamma = \theta + \alpha \tag{3.3}$$

In those equations, D represents the aerodynamic drag, L the lift, F the thrust and $m\,g$ the weight of the rocket. Some assumptions have been done in order to simplify the equations. As the angle of attack is usually small it has been considered equal to zero and so the lift has been neglected. Finally, the system of equations which describe the rocket motion are written below.

$$\dot{X} = v\,\cos\gamma \tag{3.4}$$

$$\dot{H} = v\,\sin\gamma \tag{3.5}$$

$$\dot{v} = \frac{F(t)}{m(t)} - \frac{D(H)}{m(t)} - g\sin\gamma \tag{3.6}$$

$$v\,\dot{\gamma} = -g\cos\gamma + \frac{v^2}{R_E + H}\cos\gamma \tag{3.7}$$

where: D drag $[N]$
F thrust $[N]$
g gravitational acceleration $[m/s^2]$
H altitude $[m]$
m spacecraft mass $[kg]$
v spacecraft velocity $[m/s]$
X ground range $[m]$
γ spacecraft flight path angle $[rad]$

The expressions of the drag is written below.

$$D(H) = 0.5\,\rho(H)\,S\,C_d\,v^2 \tag{3.8}$$

where: C_d drag coefficient $[-]$
D drag $[N]$
H altitude $[m]$
S reference area $[m^2]$
v spacecraft velocity $[m/s]$
ρ air density $[kg/m^3]$

The drag coefficient is a function of the local flow Mach number and the geometry of the rocket, but in this study it has been considered constant. This assumption has been done because the changes in drag coefficient have no significative impact on the results of the simulation. Indeed, the local flow Mach number depends strongly on the geometry shape of the vehicle. Concerning the rocket, the drag coefficient is typically constant and approximately equal to 0.3 in the subsonic and hypersonic flight phase. Around Mach 1, the drag coefficient is higher, about 0.5. Simulations have been done with a constant drag coefficient equal to 0.3 and a non constant drag coefficient. The drag coefficient was set to 0.3 for a Mach number smaller than 0.9 and larger than 1.1, and equal to 0.5 for a Mach number between 0.9 and 1.1. These changes in the drag coefficient are not real but the purpose was to see the impact on the simulation results. From two similar simulations, by changing only the drag coefficient as explained previously, it has been shown than there is no significant change on the simulation results. The velocity change is around 0.1% and the altitude change is around 6%. So finally, no function for the drag coefficient has been implemented in the code but it is a possibility to do so.

3.1.2 Environment model

The International Standard Atmosphere model is used. The equation 3.9 shows the dependency in altitude of the air density.

$$\rho(H) = \rho_0\,\exp(-\frac{H}{H_0}) \tag{3.9}$$

with

$$H_0 = \frac{R\,T(H)}{G\,M} \tag{3.10}$$

$$T(H) = T_0 + a\,H \tag{3.11}$$

where: a lapse rate $[K/m]$
H_0 scale height $[m]$
H altitude $[m]$
G gravitational constant $[m^3/kg\,s^2]$
M molecular mass of air $[kg/mol]$
R perfect gas constant $[J/molK]$
T_0 temperature at sea level $[K]$
T local temperature $[K]$
ρ_0 density at sea level $[kg/m^3]$
ρ local density $[kg/m^3]$

The local gravity is calculated with the following equation.

$$g = g_0\,(\frac{R_E}{R_E + H})^2 \tag{3.12}$$

where: H altitude $[m]$
g_0 sea level gravity $[m/s^2]$
g local gravity $[m/s^2]$
R_E Earth radius $[m]$

The following constants are implemented in the code:

Entity		Value
Lapse rate $[K/m]$	a	-0.0065
Gravity at sea level $[m/s^2]$	g_0	9.80665
Gravitational constant $[m^3/kg\,s^2]$	G	3.9893 10^{14}
Molecular mass of air $[kg/mol]$	M	28.97 10^{-3}
Temperature at sea level $[K]$	T_0	288
Density at sea level $[kg/m^3]$	ρ_0	1.29
Perfect gas constant $[J/molK]$	R	8.31432
Earth radius $[km]$	R_E	6371

Table 3.1: Model constants

3.2 Trajectory Basics

3.2.1 Gravity turn trajectory

A launch vehicle typically starts its ascent with a vertical rise and in order to reach a horizontal position at burnout, the vehicle has to turn and this is automatically done by dynamics, in what is termed a gravity turn trajectory, or the pitch over maneuver. The typical trajectory is represented in Figure 3.2. After the vertical rise, the vehicle is nudged during a certain amount of time in order to reach a horizontal position at burnout. The pitch over maneuver has to be executed in order to minimize the gravity losses, as during the vertical ascension the gravity directly acts against the thrust of the rocket, and also the pitch over should be applied when the vertical velocity is small in order to reduce the aerodynamic losses during the maneuver [1].

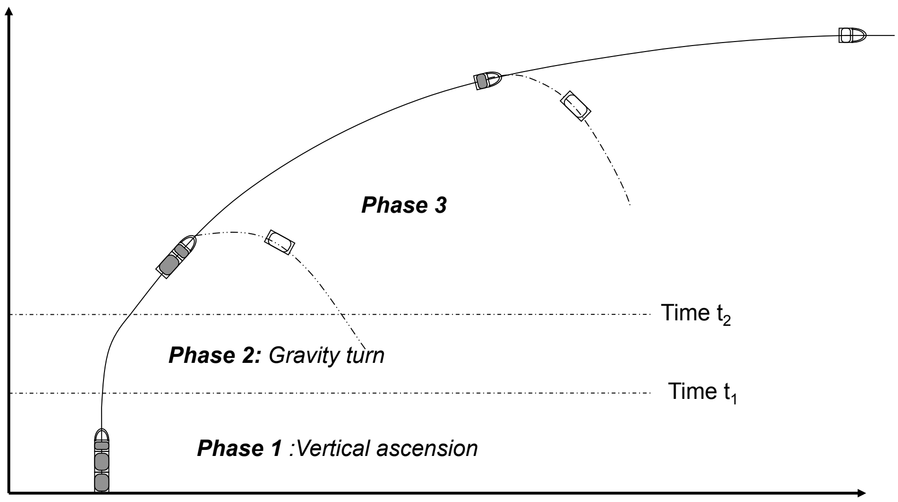


Figure 3.2: The gravity turn trajectory

The pitch over maneuver consists of changing the thrust direction. This force creates a torque, and so turns the vehicle so that it no longer points vertically. The pitch over angle is directly implemented in the rocket guidance system.

During phase 1 and 3, the rocket trajectory is calculated by the system of equations presented before. During phase 2, a rate γ_0 is added to equation 3.7, this equation becomes:

$$v\,\dot{\gamma} = -g\,\cos\gamma + \frac{v^2}{R_E + H}\,\cos\gamma - \gamma_0 \tag{3.13}$$

In the tool, this decreasing rate can be constant or dependent on other variables and a function of time.

3.2.2 Hohmann transfer

In order to reach the circular orbit, the spacecraft first reaches an elliptical transfer orbit and then a circularization is done. This particular transfer is called the Hohmann transfer.

The Hohmann transfer was theorized by a German engineer, Walter Hohmann in 1972. The Hohmann transfer is the most fuel-efficient method which uses an elliptical orbit to transfer between two circular orbits. By definition, the Hohmann transfers are limited to coplanar, circular orbits with impulsive velocity changes. The velocity changes are considered instantaneous because the duration of the engine ignition is very short compared to the Hohmann Transfer flight period. In reality an extra amount of fuel is required to compensate the fact that it takes time to burn the fuel. This transfer is used when there is a need to put a satellite in a circular orbit. It costs less energy to first reach a transfer orbit, i.e. an elliptical orbit, and therefore into its final circular orbit. To transition from the transfer orbit to the final orbit, the orbit's energy has to be changed by shifting the spacecraft's velocity by an amount of ΔV.

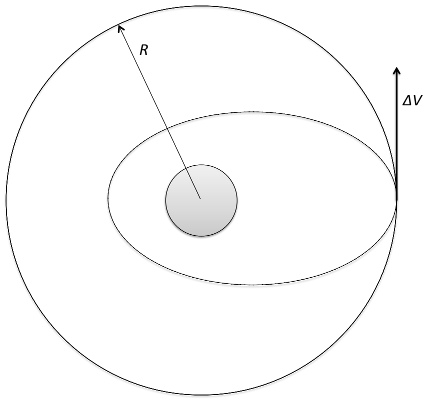


Figure 3.3: The Hohmann transfer

ΔV calculation

The ΔV is the difference between the circular orbit speed and the elliptical orbit speed at the apogee. The speed of any object on a circular orbit around the planet is defined by the radius of the orbit and the gravitational parameter of the planet.

$$v_{circ} = \sqrt{\frac{\mu}{R}} \tag{3.14}$$

where: R circular orbit radius $[m]$
v_{circ} circular orbit speed $[m/s]$
μ gravitational parameter $[m^3/s^2]$

From this equation, it can be noticed that the speed on a lower circular orbit is higher than on an orbit with an increased radius.

If the orbit is an ellipse, the speed of any object is defined by the ellipse's dimensions and the planet's gravitational parameter.

$$v_{ell} = \sqrt{\frac{2\,\mu}{r} - \frac{\mu}{a}} \tag{3.15}$$

where: a semi-major axis of the ellipse $[m]$
r ellipse radius at the analyzed point $[m]$
v_{ell} elliptic orbit speed $[m/s]$
μ gravitational parameter $[m^3/s^2]$

Finally the ΔV required for the orbit transfer is

$$\Delta V = \sqrt{\frac{\mu}{R}} - \sqrt{\frac{2\mu}{R} - \frac{\mu}{a}} \tag{3.16}$$

Specific impulse

The specific impulse I_{sp} of a spacecraft engine is a way to determine the engines's efficiency and can be described by the ratio of the engine's thrust F and mass flow $\dot{m}_{engine}$, implicating the gravitational acceleration constant of the closest planet. The formula can also be written as the ratio of the exhaust velocity of the engine nozzle with the gravitational acceleration constant.

$$I_{sp} = \frac{F}{\dot{m}_{engine}\,g_0} = \frac{v_e}{g_0} \tag{3.17}$$

where: F engine's thrust $[N]$
g_0 gravitational acceleration constant $[m/s^2]$
I_{sp} specific impulse $[s]$
$\dot{m}_{engine}$ engine's mass flow $[kg/s]$
v_e exhaust velocity $[m/s]$

The higher the specific impulse is, the higher the thrust is for the same amount of propellant. The specific impulse indicates which rocket is the most efficient when comparing rockets.

Tsiolkovsky equation

The Tsiolkovsky equation is a basic equation in astronautics which describes the velocity change of a spacecraft after a maneuver by relating it with the effective exhaust velocity and the initial and final total mass of the spacecraft before and after the maneuver.

$$\Delta V = v_e \ln(\frac{m_i}{m_f}) \tag{3.18}$$

where: m_i initial total mass $[kg]$
m_f final total mass $[kg]$
v_e exhaust velocity $[m/s]$
ΔV velocity change $[kg]$

From equation 3.18, it is possible to determinate the amount of propellant needed for the Hohmann transfer maneuver. The initial total mass consists of the propellant mass needed for the maneuver and the final total mass: $m_i = m_p + m_f$. The amount of propellant needed for the maneuver can easily be derived:

$$m_p = m_f\,(\exp(\frac{\Delta V}{I_{sp}\,g_0}) - 1) \tag{3.19}$$

where: g_0 gravitational acceleration constant $[m/s^2]$
I_{sp} specific impulse $[s]$
m_f final total mass $[kg]$
m_p propellant mass $[kg]$
ΔV velocity change $[kg]$

Orbital characteristics

This part aims to explain characteristics of an ellipse orbit, and more precisely how to determinate the apogee and perigee of an elliptical orbit. The elliptic orbit and its principal characteristics are represented in Figure 3.4. A point M on the orbit is represented by its radius from the focus and by its true anomaly θ.

The polar equation of the elliptic orbit is written in equation 3.20, it depends on the specific relative angular momentum h, the eccentricity e and the true anomaly θ.

$$r = \frac{h^2}{\mu}\,\frac{1}{1 + e\,\cos\theta} \tag{3.20}$$

$$h = v\,cos(\gamma)\,r \tag{3.21}$$

$$e = \sqrt{1 + \frac{2\,\epsilon\,h^2}{\mu^2}} \tag{3.22}$$

$$\epsilon = \frac{v^2}{2} - \frac{\mu}{r} \tag{3.23}$$

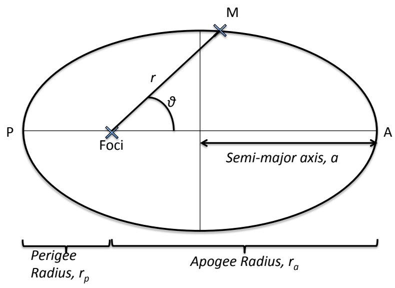


Figure 3.4: Elliptical orbit characteristics

where:
- e eccentricity
- h specific angular momentum $[m^2/s]$
- r ellipse radius $[m]$
- v velocity $[m/s]$
- ϵ specific orbital energy
- γ flight path angle $[rad]$
- μ gravitational parameter $[m^3/s^2]$
- θ true anomaly $[rad]$

From equation 3.20, the longest distance from the foci, named the apogee (Equation 3.25) and the shortest distance from the foci, named the perigee (Equation 3.24) can be easily calculated.

$$r_p = \frac{h^2}{\mu} \frac{1}{1+e} \tag{3.24}$$

$$r_a = \frac{h^2}{\mu} \frac{1}{1-e} \tag{3.25}$$

Finally, by knowing the velocity, the flight path angle, the altitude of a spacecraft, the apogee radius and perigee radius can be calculated.

3.3 Optimization: Nelder-Mead Method

In the performance calculations, in order to optimize the different variables, an algorithm has been used: it is the Nelder-Mead algorithm which is presented in the following part.

The Nelder-Mead method is a nonlinear optimization technique. This method is based on the concept of a simplex, which is a special polytope of $N+1$ vertices in N dimensions. This method approximates a local optimum of a problem with N variables when the objective function varies smoothly [4]. As it is a local optimum, it is important to start the optimization with points already close to a good solution. This method has been chosen regarding the simplicity of the algorithm and mostly because of the need for smoothness in the solution. The algorithm is represented in Figure 3.5. The understanding of the algorithm in n dimensions can be difficult so in order to make it clear, the explanation of the algorithm will be done in two-dimensional space.

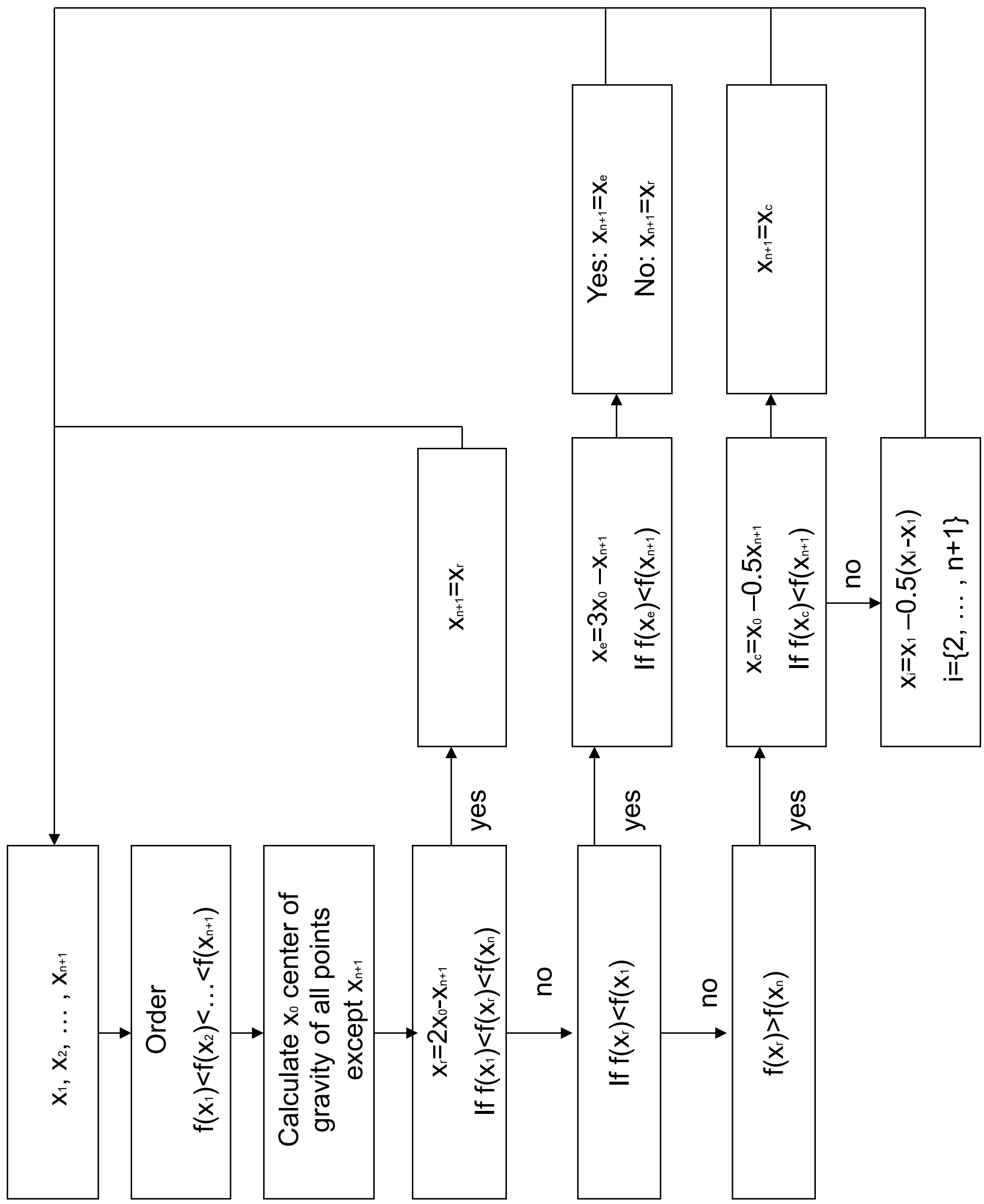


Figure 3.5: The Nelder-Mead algorithm

Example in 2D

In a two-dimensional space, three vertices are required to find the optimum points by using the simplex method. So, three points x_1, x_2 and x_3 are considered. In two dimensions, those three points form a triangle. The objective function f is calculated for the three points, and so the points can be derived from the one which has the smallest objective function to the worst one, for instance considering $f(x_1) < f(x_2) < f(x_3)$. The point which has the worst objective function is replaced by its reflected image in a mirror where the mirror is the segment formed with the two other points. A representation is done in Figure 3.6. In the first triangle the worst point is x_3, the point is replaced and a second triangle is formed, the blue one. In this new triangle, the worst point is x_1, the point is replaced and a new triangle is formed again, the green one. And so on, the points move in the space and converge to the optimum point. On the following figure, six triangles are represented.

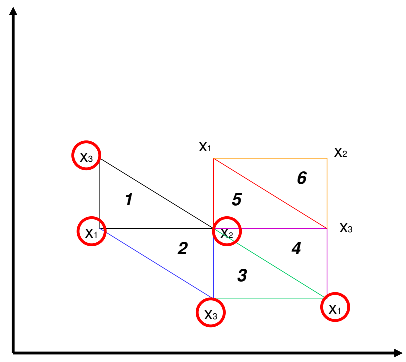


Figure 3.6: Example of the simplex method in a two-dimension space

The Nelder-Mead algorithm is based on the simplex method as explained previously. The differences are on the triangle change. In the simplex method, it is based on reflection, but in the Nelder-Mead algorithm it is based on the reflexion, the expansion, the contraction or the reduction depending on the objective function. These four triangle changes are represented in Figure 3.7.

The choice in the triangle is done as follows. The objective functions of the three points are calculated. The point with the worst objective function will be replaced. For instance, considering Figure 3.7, x_1 will be replaced. First the reflected point x_r is calculated. If the objective function of x_r is between the objective functions of the two other points x_2 and x_3, the worst point x_1 is so replaced by x_r. If it is not the case, there are two possibilities.

- If the objective function of x_r is even better than the objective functions of both points x_2 and x_3, the expansion point x_e is calculated. If the objective function of x_e is smaller than the objective function of x_1, the worst point x_1 is replaced by x_e; if not x_1 is replaced by x_r.

- If the objective function of x_r is larger than the objective functions of both points x_2 and x_3, the contraction point x_c is calculated. If the objective function of x_c is smaller than the objective function of x_1, the worst point x_1 is replaced by x_c, if not all the points are replaced except the one with the best objective function, that is called the reduction.

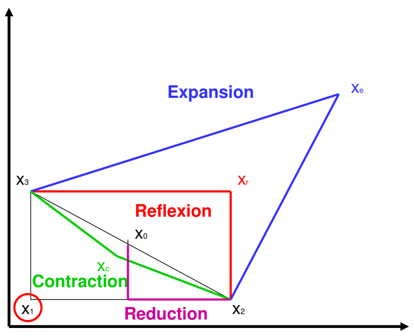


Figure 3.7: The different move possibilities in the Nelder-Mead method

In the developed code, there are several possibilities to choose the variables. But, the simplest program is the one which optimizes only the time when the gravity turn program is applied and all the launcher characteristics are fixed. The gravity turn program variables are the beginning and the end of the gravity turn program, so only two variables. For this program, there are three vertices of two coordinates: $x = \begin{pmatrix} t_1 & t_2 \end{pmatrix}$ where t_1 and t_2 define the interval during which the gravity turn program is running. The objective function is $(r_{calculated,perigee} - r_{desired,perigee})^2 + (r_{calculated,apogee} - r_{desired,perigee})^2$. $r_{calculated,apogee}$ and $r_{calculated,perigee}$ are the apogee and perigee radius calculated from a simulation. So, by changing the gravity turn program, the optimization program tries to find the best solution to reach the elliptical orbit if possible.

Other optimization programs have been developed, for instance in order to optimize the amount of propellant in the stages. In this optimization program the vertices are points of four coordinates $x = \begin{pmatrix} t_1 & t_2 & m_{prop,i} & m_{prop,j} \end{pmatrix}$ where $m_{prop,i}$ and $m_{prop,j}$ correspond to the propellant mass of the stage i and j respectively. If a change is done in the amount of propellant of one stage, this change has a direct impact on the trajectory and normally for such a change the gravity turn needs to be optimized as well. Actually, it would be more relevant to have two algorithms working at the same time. But, from experience, this optimization takes a long time to run, and so it has been chosen to change all the variables in the same algorithm when the purpose is to find the optimum launcher. Once the launcher characteristics are fixed, only the variables from the gravity turn are changed to do the performance calculations.

3.4 Functional Budget

The upper stage needs to be designed using a engine under investigation at Astrium. From the performance calculations, the upper stage characteristics are chosen. The main point is to evaluate all the different masses included in the upper stage. It is therefore required to calculate a propellant budget in order to evaluate the required propellant reserve, and a helium budget as well, in order to evaluate the amount of helium needed for pressurization [5].

In the following part, the Functional Budget Calculation Software Tool is presented. This tool has been designed by a student as part of a Master Thesis. As the tool has been used without making any modifications, the most important aspects of a functional budget are explained here; to get more information about the tool the report for this tool is given in the references [3].

The Functional Budget Calculation Software Tool is a Microsoft Excel based tool and contains four main sheets: propellant budget, propellant tank volume, pressure budget and pressure gas compression. This tool can be applied for the MMH/NTO propellant combination and for LOX/LH2 as well. In this study, the developed upper-stage uses a MMH/NTO engine, so only the aspects of the functional budget concerning this propellant combination are presented in this paper.

3.4.1 Propellant budget

The rocket mass is constituted mostly of propellant mass, so each kilogram of propellant has to be optimized because a decrease in propellant mass means a gain in payload. So the propellant budget has to be as accurate as possible.

In Figure 3.8, the propellant budget sheet is illustrated. It presents the propellant masses taken into account in the propellant budget: the nominal propellant, the performance reserve, the lines priming, the transient phase engine consumption, the geometrical residuals, the project design margin and the loading process inaccuracy of loaded propellant masses. The inputs or calculations of all these different masses are mostly separated for oxidizer and fuel.

Number of boosts
Propellant mixture ratio (MR)
Module propulsive mixture ratio (MRu)
Nominally used propulsive propellant mass (main. prop. + att. control) [kg]
Performance reserve (input either as % or comp. with MRu)
Transient phase engine consumption [kg] (conservative)
Lines priming: m_lines [kg]
m_1: Enhanced nom. propellant mass [kg]
Geometrical residuals [kg]
Project design margin (of nominally used prop. mass) [%]
m_2: Loaded propellant mass [kg]
Loading process inaccuracy: I (of loaded mass) [%]
Total loaded propellant mass [kg]

Figure 3.8: Propellant budget

Propellant mixture ratio MR

The mixture ratio of the propellant combination describes the ratio between the used oxidizer and the fuel. It is normally specific to an engine.

The transient phase engine consumption

The number of boosts of an engine has an important role in the propellant budget because the engine ignition and shut-down require propellant. The transient phase engine consumption is equal to the propellant mass needed for the engine ignition and shut down time for the number of boosts. The required propellant for the ignition and shut down of the engine are inputs in the excel sheet.

The nominal propellant

The nominal propellant mass is directly linked to the performance calculation and it is a manual input. It includes the needed propellant for the velocity change and for the attitude control maneuvers.

The propellant reserve

The propellant reserve is a percentage of the nominal propellant which allows a margin on the nominal propellant mass in order to avoid premature run-out.

The residual propellant

The residual propellant is a propellant which can not be used to produce thrust as it includes the propellant which clings to tank walls, which is trapped in systems by valves or pipes and which is used to feed the lines. This residual propellant includes so the geometrical residual and the lines priming. They are both inputs in the excel files and the amount of propellant needed in both cases have been validated by experts.

The project design margin

The project design margin is a percentage of the nominal propellant which covers the possible change in the nominal propellant.

The loading process inaccuracy

While loading the propellant, some inaccuracies can appear and so a margin is taken on the loaded propellant to compensate those inaccuracies.

Total loaded propellant

The total loaded propellant is the sum of all the different propellant masses presented previously.

3.4.2 Propellant tank volume

The propellant tank volume can be determined once the propellant budget has been done and once the total needed propellant mass is known. Figure 3.9 illustrates the propellant tank volume sheet. The volume calculations are done separately between the fuel and the oxidizer.

The manual inputs are the total loaded propellant mass, the density of the propellants, the ullage volume and the project design margin. The propellant volume is equal to the propellant mass divided by the propellant density.

$$V_{ox,load} = \frac{m_{ox,load}}{\rho_{ox}} \tag{3.26}$$

$$V_{f,load} = \frac{m_{f,load}}{\rho_f} \tag{3.27}$$

Ullage volume

The ullage volume corresponds to the volume in the tank which is not filled with propellant.

Total loaded propellant mass [kg]
Temperature propellant at lift-off [K] (for information only)
Density propellants [kg/m³]
Total loaded propellant volume: V_load [m³]
Ullage volume needed: V_ull [input: % or m³]
Internal equipments
Project margin M_proj [%]
Tank Volume [m³]

Figure 3.9: Propellant tank volume

Project design margin

The project design margin is a volume which shall cover any possible change in the nominal value.

Total tank volume

The total tank volume is equal to the loaded propellant volume, the ullage volume and the margin volume for both oxidizer and fuel.

3.4.3 Helium budget

The engine type requires an inert gas in order to expel and pressurize the liquid propellants. The considered inert gas is Helium and it is stored under pressure in a separate vessel. The total needed pressure gas mass has to be calculated. Figure 3.10 illustrates the pressure gas budget sheet and the different items are described below. Only the items concerning the MMH/NTO configuration will be described. The calculations are done separately between the oxidizer and the fuel.

Flow rates

The engine mass flow rate is a manual input and it is a characteristic of the engine. The propellant flow rates are calculated thanks to the propellant mixture ratio defined in the propellant budget calculation and the engine mass flow.

$$\dot{m}_{ox} = \frac{MR\,\dot{m}_{engine}}{MR+1} \tag{3.28}$$

$$\dot{m}_{f} = \frac{\dot{m}_{engine}}{MR+1} \tag{3.29}$$

The propellant volume flow rate is the ratio between the propellant flow rate and the propellant density. The volume flow rate pressure gas is equal to the propellant volume flow rate.

Pressure, temperature, density

The tank pressure, the ullage pressure gas temperature and the specific gas constant pressure gas are implemented as manual inputs. The density of the pressure gas is calculated with the formula

Engine mass flow: m-dot_engine [kg/s]
Propellant flowrate: m-dot [kg/s]
Density propellant: ρ [kg/m³]
Prop. volume flowrate: v-dot [m³/s]
Vol. flowrate pressure gas: v-dot_He [m³/s]
Tank pressure: p [bar]
Ullage temperature press. gas: T_ull [K]
Specific gas const. pressure gas: R_He [J/(mol*K)]
Density pressure gas: ρ_He [kg/m³]
Mass-flowrate pressure gas: m-dot_He [kg/s]
Propellant to be fed out [kg]
Pressurization duration [s]
Pressure Gas need [kg]
Tank pressurization: 5bars - 20 bars [kg]
Command system consumption [kg]
Project margin: M_proj [kg]
Total need pressure gas: m_He,tot [kg]

Figure 3.10: Pressure budget

for the ideal gas equation. The calculations is done for both tank, fuel and oxidizer.

$$\rho_{He} = \frac{p}{T_{He} R} \tag{3.30}$$

where
- p propellant pressure $[Pa]$
- R specific gas constant $[J/molK]$
- T_{He} Helium temperature $[K]$
- ρ_{He} Helium density $[kg/m^3]$

Pressure gas need

The pressure gas need is equal to the product between the mass flow rate of the pressure gas and the pressurization duration which is equal to the ratio between the propellant mass for the operational consumption and the propellant flow rate.

Margins

The project margin is inserted as a manual input.

Total need pressure gas

The total need pressure gas is the sum of the total gas needed and the margins.

Once the helium mass is known, the maximal expected operating pressure (MEOP) has to be defined. This MEOP is important in the dimensioning of the helium vessels. Figure 3.11 illustrates the pressure gas compression sheet.

Maximum He regulated pressure [bar]
Needed Helium mass m_He, tot [bar]
He consumption variations (5% incl. leakage) [kg]
Helium need for propellant expulsion only [bar]
Helium budget non-consumable [bar]
He consumption deviations [bar]
He budget uncertainty & margin [bar]
Total press. need for He storage. P_He,stor [bar]
He pressure decrease due to temp. compensation [bar]
He pressure rise due to temp. compensation [bar]
MEOP He vessel [bar]

Figure 3.11: Maximum helium pressure

Maximum He regulated pressure

The value for the maximum helium regulated pressure is dependant on the propellant system and it is implemented as a manual input.

He needed for propellant expulsion only

The helium mass needed is directly calculated from the previous calculations. The helium consumption variation is set at 5% in this study. The formula which enables the calculation of the helium needed for propellant expulsion is written below.

$$p_{He,prop} = \frac{m_{He,tot} + m_{He,var}}{M_{He}\,10^{-3}}\, z_{He}\, R_{He}\, \frac{T_{He}}{V_{He,vessel}\,10^{-3}}\, 10^{-5} \tag{3.31}$$

where		
	M_{He}	helium molar mass $[g/mol]$
	z_{He}	helium compressibility factor $[-]$
	R_{He}	general gas constant $[J.mol]$
	T_{He}	helium temperature $[K]$
	$V_{He,vessel}$	helium vessel (manual input) $[L]$

Margins

Due to the design of the vessel, a pressure residual of helium can not be expelled. From experience the non consumable helium pressure is fixed to 30bars. 10bars are considered for the helium consumption deviations, for uncertainty and margin, for the pressure decrease due to temperature compensation and for the pressure rise due to temperature compensation. The maximum operating pressure is the sum of the different pressure defined before.

By changing the volume of the helium vessel, the maximum operating pressure can be varied.

3.5 Structural Mass

In order to design the upper-stage, the different parts of the structure have to be defined. The following part explains the calculations needed to define the upper-stage structure.

Structures of spacecraft have to be resistant enough to support the load and accelerations, and they have to be as light as possible. Nowadays, Finite-Element Methods enable analysis of nearly all the possible structures but the calculations usually takes a long time. For the purpose of this study, the need to get a rough and fast estimation is essential, therefore simplified calculation methods are used to estimate loads and structural masses.
In this section, the Structural Mass Calculation Software Tool is explained. It has been designed by the same student and it is an Microsoft Excel based tool [3]. Unlike the previous tool, some improvements have been made in the Structural Mass Calculation Software Tool, and so the calculations will be explained in the following parts. The changes have been made in order to develop in more detail the upper-stage structure.

3.5.1 Upper stage components

Figure 3.12 describes the main components of an upper stage [5]:

- Upper cone
- Avionics platform
- Helium vessel
- Intermediate structure
- Tanks
- Engine thrust frame
- Interstage skirt

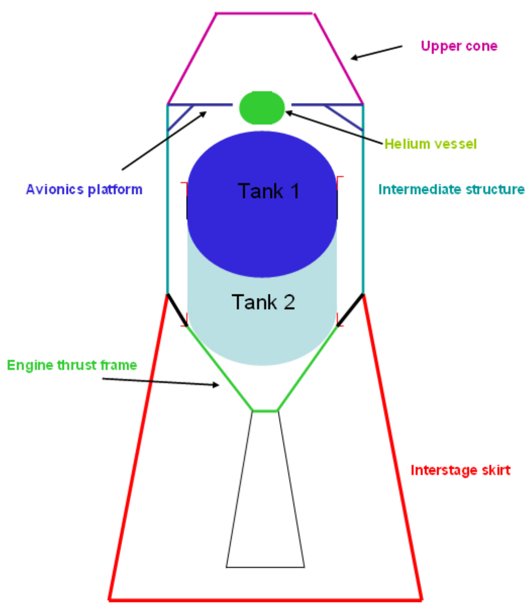


Figure 3.12: Main components of an upper stage

All these components have been considered in the Structural Mass Calculation Tool. The upper cone is the intermediate surface between the stage and the payload. The avionics platform allows a fixture for all the instruments. The helium vessel is fixed to the avionics platform or the engine thrust frame. The intermediate structure has a shape of a cylinder and carries the loads from the structure above and builds the junction to the interstage structure below. The interstage structure can have the shape of a cylinder or can be conical depending on the diameter of the stage below. The interstage skirt is the intermediate structure between the upper stage and the stage below. This structure is dropped when the upper stage separates itself from the stage below.

3.5.2 Structural Mass Calculation Software Tool

Illustrations of the complete tool can found in the Appendix D.

The Structural Mass Calculation Tool has been modified in the purpose of this study. This tool can support three different stage architectures: the common bulkhead tank as load-bearing (on the left of Figure 3.13) or not load-bearing version(in the middle of Figure 3.13) and the spherical-cylindrical tank as not load-bearing version (on the right of Figure 3.13).

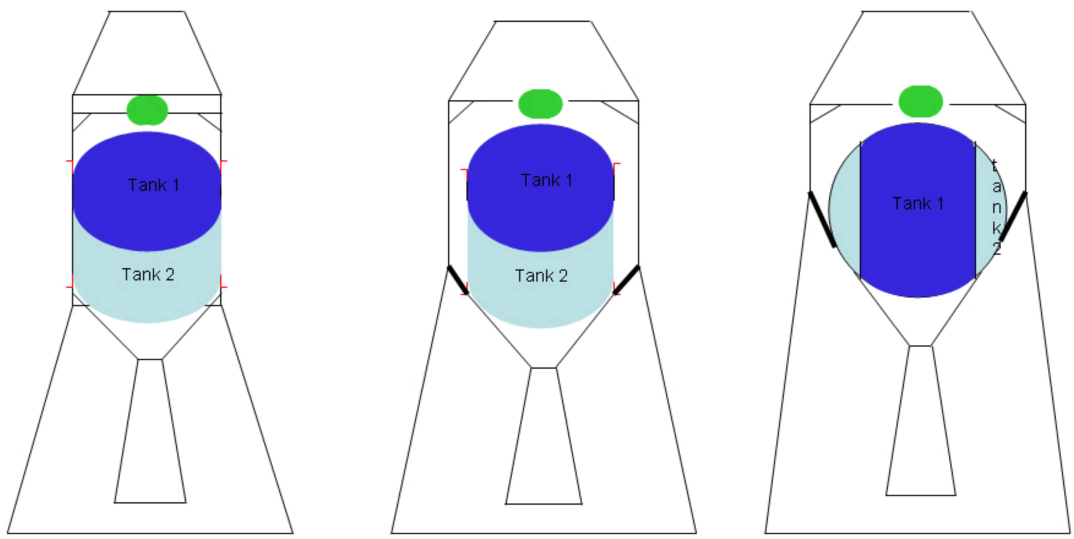


Figure 3.13: Tank configurations

The structural masses are evaluated regarding the loads and resulting fluxes and moments during the whole mission. As the need is to get a fast estimation of the structure masses, some assumptions have been made and they are listed below:

- all dome tanks are treated as ellipsoidal and the length of the dome is fixed.
- relatively small parts such as skirts, clampband and jettisons for separation systems are not considered.

Input Data

This section summarizes the manual inputs for the Structural Mass Calculation Tool.

First of all, the tank configuration has to be chosen by activating check-boxes, the calculations are done depending on the tank configuration.

Accelerations

The acceleration factors are important in the dimensioning of a stage and describe the gravitational force which affects the stage during its ascent.

Acceleration	
Max Axial Acceleration: k_ax [ ]	5,0
Max Lateral Acceleration: k_lat [ ]	1,0

Propellant Tank 1:	
Propellant Density: ρ_1 [kg/m³]	1433,00
Tank Volume: V_tank1 [L]	1659
ullage pressure: p_ull,1 [bars]	3,0
Propellant mass: m_prop,1 [kg]	2168,00
Cylinder Radius: r_cyl,1 [mm]	800,0

Propellant Tank 2:	
Density Propellant: ρ_2 [kg/m³]	875,00
Tank Volume: V_tank2 [L]	1432
ullage pressure: p_ull,2 [bars]	3,0
Propellant mass: m_prop,2 [kg]	1139,00

Helium:	
Helium mass [kg]	17,5
Tank Volume [L]	250,0
Tank Mass [kg]	63,0

Other Fluid:	
Nitrogen mass [kg]	
Hydrazine mass [kg]	

Propellant:	
Useful propellant	3000,0
performance reserve	75,0
propellant reserve	292,0

Structural Materials	
Alu 2219	
σ_Alu [N/mm2]	358,0
E_Alu [N/mm²]	72000,0
ρ_Alu [kg/m3]	2851,0
CFRP M40/6376 Sandwich	
E_CFRP [N/mm²]	96386,0
ρ_skins [kg/m³]	2296,0
ρ_core [kg/m³]	50,0

P/L, Fairing (Helios), Engine	
payload mass: m_P/L [kg]	250,0
fairing mass: m_frng [kg]	180,0
fairing length: l_frng [mm]	2000,0
fairing diam: d_frng [mm]	1200,0
engine mass: m_eng [kg]	90,0

ETF Geometry	
big radius: R_ETF [mm]	1000,0
small radius: r_ETF [mm]	100,0
height: h_ETF [mm]	400,0
surface height cone: l_ETF [mm]	984,9

Wind Loads	
F_wind [N]	7889,7

Figure 3.14: Structural mass inputs

Manual inputs: Max. axial acceleration: k_{ax} $[-]$
Max. lateral acceleration: k_{lat} $[-]$

Propellant tank 1

For the propellant tank 1, the volume and mass of propellant have to be implemented as well as the ullage pressure in the tank.

Manual inputs: Tank volume: V_{tank1} $[L]$
Ullage pressure: $p_{ull,1}$ $[bar]$
Propellant mass: $m_{prop,1}$ $[kg]$

If the common bulkhead configuration is chosen, another manual input is required; it is the radius of the tank cylinder ($r_{cyl,1}$ $[mm]$).

Propellant tank 2

For the propellant tank 2, the volume and mass of propellant have to be implemented as well as the ullage pressure in the tank.

Manual inputs: Tank volume: V_{tank2} $[L]$
Ullage pressure: $p_{ull,2}$ $[bar]$
Propellant mass: $m_{prop,2}$ $[kg]$

Structural Materials
Two different structural materials are used in this tool: an aluminum alloy and a composite material. The aluminum alloy is Aluminum 2219 which is a common material in aerospace design. The composite material is a sandwich structure which consists of carbon fiber skins and a hexagonal aluminum honeycomb as core material.
For aluminum 2219, the yield stress, the modulus of elasticity and the material's density have to be entered.

Manual inputs: Yield stress: σ_{Alu} $[N/mm^2]$
Modulus of elasticity: E_{Alu} $[N/mm^2]$
Density: ρ_{Alu} $[kg/m^3]$

For the composite material, the modulus of elasticity and the density of the CFRP as well as the density of the honeycomb core material have to be implemented.

Manual inputs: CFRP modulus of elasticity: E_{CFRP} $[N/mm^2]$
CFRP density: ρ_{CFRP} $[kg/m^3]$
Core density: ρ_{core} $[kg/m^3]$

Payload fairing and engine
The fairing is the structure placed on the top of the upper stage which covers the payload during the ascent. The payload mass and the fairing mass, length and diameter have to be defined. The engine mass is required as well.

Manual inputs: Payload mass: m_{pay} $[kg]$
Fairing mass: m_{fair} $[kg]$
Fairing length: l_{fair} $[mm]$
Fairing diameter: d_{fair} $[mm]$
Engine mass: m_{eng} $[kg]$

Engine thrust frame
The engine thrust frame has a conic geometry. The cone characteristics are also required inputs.

Manual inputs: Big radius: R_{ETF} $[mm]$
Small radius: r_{ETF} $[mm]$
Height: h_{ETF} $[mm]$

Axial fluxes
The stage has to be designed in order to withstand the axial fluxes via the structure. For the calculation of theses axial fluxes, the stage has been separated into sections. Each section has a constant load. The sections are represented in Figure 3.15 for each tank configuration. The loads for each section are summarized below.

- Section 1: Payload + Fairing + Wind loads
- Section 2: section 1 mass + Propellant mass + Helium storage
- Section 3: section 2 mass + Engine mass

For the calculations concerning the tank structure, in the load-bearing version, the load considered is the load of section 2, in the non load-bearing version, the only load considered is the mass of propellant.

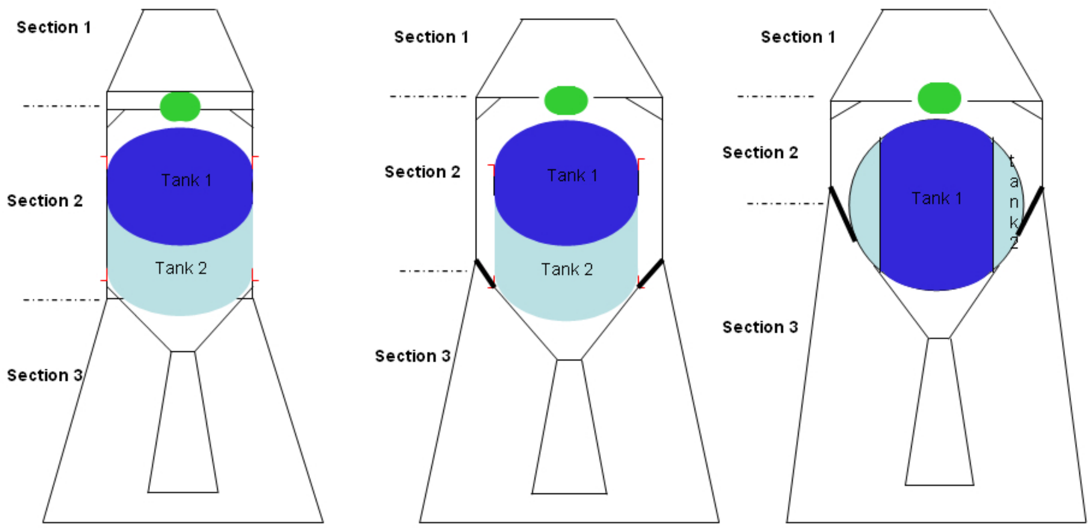


Figure 3.15: Section level of the evaluated tank configuration

The length and the diameter of each section has to be manually implemented.

Sections	Length [mm]	Diam. [mm]	Axial Fluxes: Fax [N]	M [Nmm]	n_Fax	n_M	n_max	considered masses
length - fairing top to sec 1	2000	1200	21092	12107970	6	11	16	P/L +fairing +wind
length - sec 1 to sec 2	600	1350	54396	21371049	13	15	28	+ prop+helium storage
length - sec 2 to sec3	2200	1350	55377	62878581	13	44	57	+engine
load for not loadb. tanks:								
length - sec 1 to sec 2	1144	1144	660	3703583	0	4	4	prop

Figure 3.16: Structural load

For each section, the axial fluxes are calculated separately as explained below.
The axial load is calculated thanks to the section mass and the axial acceleration factor.

$$F_{ax} = m_{sec}\, k_{ax}\, g \tag{3.32}$$

The bending torque for each section is calculated by considering the point of action to be situated in the middle of the section.

$$M_{sec,i} = (m_{sec,i})\, \frac{l_{sec,i}}{2}\, g\, k_{lat} + F_{wind}\, \frac{l_{sec,1}}{2} \tag{3.33}$$

where
- F_{wind} Wind loads $[N]$
- g gravitational acceleration $[m/s^2]$
- k_{lat} max. lateral acceleration factor $[-]$
- $m_{sec,i}$ mass of the section i $[kg]$
- $l_{sec,i}$ length of the section i $[m]$

The axial load per perimeter and the bending moment per area are calculated for each section.

$$n_{F_{ax}} = \frac{F_{ax}}{\pi\, d_{sec}} \tag{3.34}$$

$$n_M = \frac{M}{\pi \frac{d_{sec}^2}{4}} \tag{3.35}$$

Finally, the total axial fluxes are the sum of the axial load per perimeter and the bending moment per area.

Structural mass calculations

The total structural mass of the stage is calculated by the sum of all the components of the stage. For some parts, the calculations have been done by scaling from a known design. This is done in order to quickly approximate the different parts.

The upper cone structure
The upper cone structure consists of the payload adaptor.The payload adapter is on what the payload is attached to during the flight. Its mass is scaled from the AVUM payload adapter [25].

The avionics platform
The avionic platform is fixed to the intermediate structure and secures all the avionics equipment. It also supports the Helium and Hydrazine tanks if needed. Its mass is calculated by scaling from a known avionics platform mass.

Tank geometry
The geometry and surfaces of the tank are calculated automatically thanks to the tank volume and with the input of the cylinder radius for the common bulkhead configuration. No input is required for the spherical-cylindrical configuration. The calculations are explained below for the two configurations. The tank geometry calculations are done in a separate sheet in the Excel tool. Figure 3.17 illustrates this sheet.

The common bulkhead configuration
The tanks have a cylinder shape and the dome is an ellipsoidal shape. The length of the dome is equal to the radius of the cylinder divided by a fixed coefficient [19].

$$r_{dome} = \frac{r_{cyl}}{\sqrt{2}} \tag{3.36}$$

The dome surface is calculated as the half of an ellipsoid surface:

$$S_{dome} = \pi[r_{cyl}^2 + \frac{r_{cyl}^2}{2\sin\alpha}\ln(1 + \frac{\sin\alpha}{\cos\alpha})] \tag{3.37}$$

$$\alpha = \arccos\frac{1}{\sqrt{2}} \tag{3.38}$$

Both cylinder tanks have the radius r_{cyl} defined as a manual input. The cylinder length of tank 1 is calculated with the tank volume, the cylinder radius and the dome volume.

$$h_{cyl,1} = \frac{1}{2\pi r_{cyl}}[V_{tank,1} - \frac{4}{3\sqrt{2}}\pi r_{cyl}^3] \tag{3.39}$$

The cylinder length of tank 2 is calculated with the volume of the tank 2.

$$h_{cyl,2} = \frac{V_{tank,2}}{\pi r_{cyl}^2} \tag{3.40}$$

The cylinder surface areas are also calculated thanks to the formulae:

$$S_{cyl,1} = 2\pi r_{cyl} h_{cyl} \tag{3.41}$$

Common Bulkhead Configuration

Tank 1:	
Tank Volume: V_tank1 [m^3]	1,659
Cylinder Radius: r_cyl,1 [mm]	800
Cylinder Length: h_cyl,1 [mm]	70,87
Cylinder Surface: S_cyl,1 [m^2]	0,36
Dome Radius: r_d [mm]	565,69
Dome Surface: S1_d [m²]	2,44
Tank 1 Surface: S1_tank [m²]	5,23

Tank 2:	
Tank Volume: V_tank2 [L]	1,432
Cylinder Radius: r_cyl,2 [mm]	800
Max. Length Cyl.: l_cyl,2,CB [mm]	712,22
Cylinder Surface: S_cyl,2 [m^2]	3,58
Dome Radius: r_d [mm]	565,69
Dome Surface: S2_d [m²]	2,44
Tank 2 Surface: S2_tank [m^2]	6,02

Spherical Configuration

Tanks geometry	
Volumes	
Total Tank Volume Need [m3]	3,1
Basic dimensions	
Sphere Radius [mm]	904
Cylinder Radius [mm]	572
Cylinder Area [m2]	1,0
Heights	
Tank 1 Dome [mm]	204
Tank 1 Cylinder [mm]	1398
Tank 2 [mm]	331
Tank Total [mm]	1807
Surfaces	
Tank 1 Dome [m2]	1,16
Tank 1 Cylinder [m2]	5,03
Tank 2 [m2]	7,94

Figure 3.17: Tank geometry calculations

Spherical-cylindrical configuration
The sphere radius is calculated from the total tank volume.

$$r_{sph} = (\frac{3}{4\pi}(V_{tank,1} + V_{tank,2}))^{1/3} \tag{3.42}$$

The cylinder length is calculated thanks to the sphere radius and the tank volume.

$$h_{cyl} = 2 * [r_{sph}^3 - \frac{3}{4\pi} * V_{tank,1}]^{1/3} \tag{3.43}$$

From the cylinder length, the dome length is calculated.

$$r_{dome} = r_{sph} - \frac{1}{2} h_{cyl} \tag{3.44}$$

The surface areas of the cylinder, the dome and the spherical tank are calculated as shown below.

$$S_{dome} = 2\pi \, r_{cyl} \, h_{dome} \tag{3.45}$$

$$S_{cyl} = 2\pi \, r_{cyl} \, h_{cyl} \tag{3.46}$$

$$S_{tank,2} = 4\pi \, r_{sph}^2 - 2 * S_{dome} \tag{3.47}$$

Tank 1 (The tank parts are all in Aluminum)
Common bulkhead configuration
The tank 1 has a cylinder shape with two domes. The total tank mass includes:

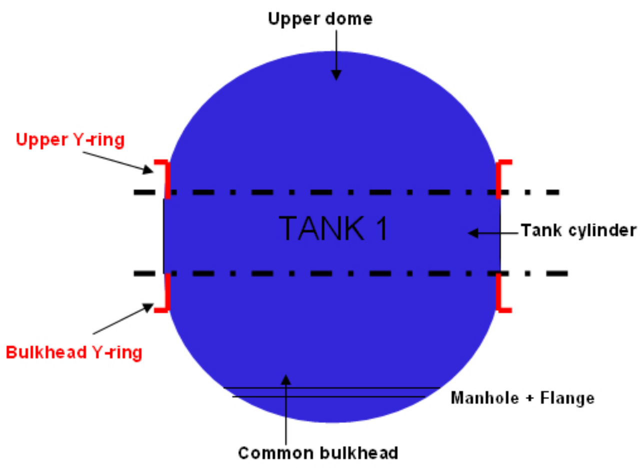


Figure 3.18: Common bulkhead tank 1 components

- Upper dome

  The upper dome mass is calculated from the dome surface, the aluminum density and the evaluated thickness t_{dome} including a contingency factor to cover possible variations of the component's mass during its design process.

$$m_{dome} = \rho_{Alu}\, S_{dome}\, t_{dome}\, C \tag{3.48}$$

  The thickness of the upper dome is evaluated to withstand the stresses and buckling due to internal pressure. The thickness required for strength is calculated using the Von Mises strength formula for an elliptical tank. The internal pressure P includes the ullage pressure (input), the dynamic pressure ($p_{dyn} = 0.5 N/mm^2$) and the hydrostatic pressure ($p_{hydro} = \rho\, n\, g\, h_{dome}$). For an elliptical tank the stresses have the following characteristics:

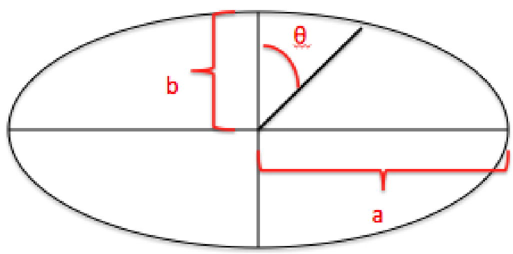


Figure 3.19: Ellipse characteristics

$$n_{1\theta} = \frac{-P\, a^2}{2} \frac{1}{(a^2 \sin^2\theta + b^2\, \cos^2\theta)^{0.5}} \tag{3.49}$$

$$n_{2\theta} = \frac{-P\, a^2}{2\, b^2} \frac{b^2 - (a^2 - b^2) \sin^2\theta}{(a^2 \sin^2\theta + b^2\, \cos^2\theta)^{0.5}} \tag{3.50}$$

$$\sigma_{Von-MISES} = \frac{\sqrt{n_{1\theta}^2 + n_{2\theta}^2 - n_{1\theta}\, n_{2\theta}}}{t_\theta} \tag{3.51}$$

As can be seen from these formulas, the thickness depends on the angle theta where the stress is observed. So for the mass calculations, the average thickness will be considered and a safety factor will be applied too for production tolerance.

$$t_{dome} = t_{average}\, SF \tag{3.52}$$

Independently of the calculation result, the minimum required thickness for the dome due to inspection criteria is $1.3mm$. If there is buckling, the dome thickness concerning buckling is evaluated with the NASA SP-8007 method [18]. The method is fully adapted and therefore not explained in this report.

- Cylinder structure mass

  The cylinder mass is calculated like the dome surface.

$$m_{cyl,1} = \rho_{Alu}\, S_{cyl,1}\, t_{cyl,1}\, C \tag{3.53}$$

  The tank shell thickness has to withstand the tank's internal pressure and the load in case of buckling. The tank thickness for the construction concerning stress due to internal pressure is calculated using the Boiler formula as for the dome thickness.

$$t_{cyl,1} = \frac{SF\, p_{max}\, r_{cyl,1}}{\sigma_{Alu}} \tag{3.54}$$

  Where SF is a safety factor for production tolerance ($j = 1.1$) and p_{max} includes the ullage pressure (input), the dynamic pressure ($p_{dyn} = 0.5N/mm^2$) and the hydrostatic pressure ($p_{hydro} = \rho\, n\, g\, h_{cyl,1}$). The tank thickness for the construction concerning stress due to buckling is evaluated with the NASA SP-8007 method.
  Independently of the calculation result, the minimum required thickness for the dome due to inspection criteria is $1.3mm$.

- Common bulkhead mass

  The calculations are similar to the ones for the upper dome. The only difference is in the hydrostatic pressure: $p_{hydro} = \rho\, n\, g\, (2h_{dome} + h_{cyl,1})$.

- Upper Y-ring mass

  The upper Y-ring mass is scaled from an existing upper Y-ring by considering the diameter and the tank's internal pressure. The upper Y-ring is the connection between the tank structure and the skirt.

- Bulkhead Y-ring mass

  The bulkhead Y-ring mass is scaled from an existing bulkhead Y-ring by considering the diameter and the tank's internal pressure. The bulkhead ring is the lower interface of the tank 1.

The spherical-cylindrical configuration
Tank 1 has a cylinder shape with two caps. The total tank mass includes:

$$m_{tank1} = S_{cyl,1}\, t_{cyl,1} \rho_{Alu} + 2\, S_{cap}\, t_{cap}\, \rho_{Alu} \tag{3.55}$$

The cap thickness is calculated using the Boiler formula in order to withstand the tank's internal pressure.

$$t_{cap} = \frac{SF\, p_{max}\, r_{cap}}{\sigma_{Alu}} \tag{3.56}$$

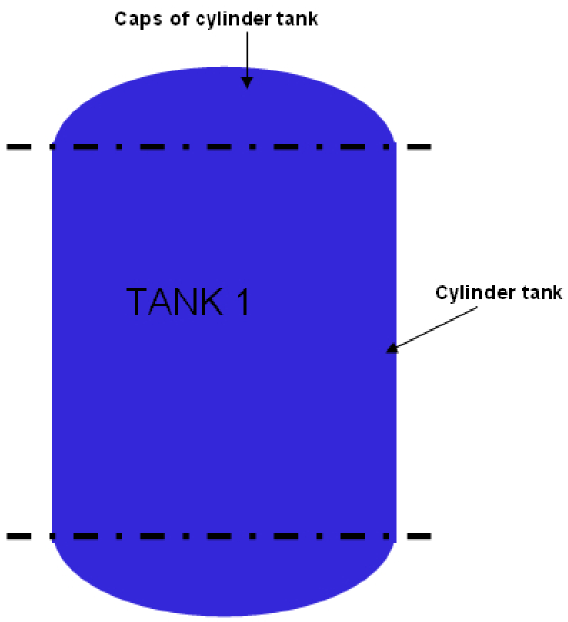


Figure 3.20: Spherical-Cylindrical tank 1 components

Where SF is a safety factor for production tolerance ($j = 1.1$) and p_{max} includes the ullage pressure (input), the dynamic pressure ($p_{dyn} = 0.5N/mm^2$) and the hydrostatic pressure ($p_{hydro} = \rho\, n\, g\, h_{cap}$).
Independently of the calculation result, the minimum required thickness for the dome due to inspection criteria is $1.3mm$. The cylinder tank thickness has to be dimensioned for bearing the tank's internal pressure as well as buckling, as the cylinder is placed inside the sphere with a pressure difference. The formula used to calculate the thickness is:

$$t_{cyl,1} = \left(\frac{p_{ull}}{1.5\,0.6\,0.91\,\frac{r_{cyl,1}}{h_{cyl,1}}\,E_{Alu}}\right)^{0.8} r_{cyl,1} \tag{3.57}$$

Tank 2 (The tank parts are all in Aluminum)
Common bulkhead configuration
Tank 2 has a cylinder shape with two domes. The total tank mass includes:

- Tank cylinder

  The cylinder mass is calculated from the cylinder surface area and the cylinder thickness.

  $$m_{cyl,2} = \rho_{Alu}\, S_{cyl,2}\, t_{cyl,2}\, C \tag{3.58}$$

  The tank shell thickness has to withstand the tank's internal pressure as well as buckling loads. The tank thickness for the construction concerning stress due to internal pressure is calculated using the Boiler formula.

  $$t_{cyl,2} = \frac{SF\, p_{max}\, r_{cyl,2}}{\sigma_{Alu}} \tag{3.59}$$

  Where SF is a safety factor for production tolerance ($j = 1.1$) and p_{max} includes the ullage pressure (input), the dynamic pressure ($p_{dyn} = 0.5N/mm^2$) and the hydrostatic pressure

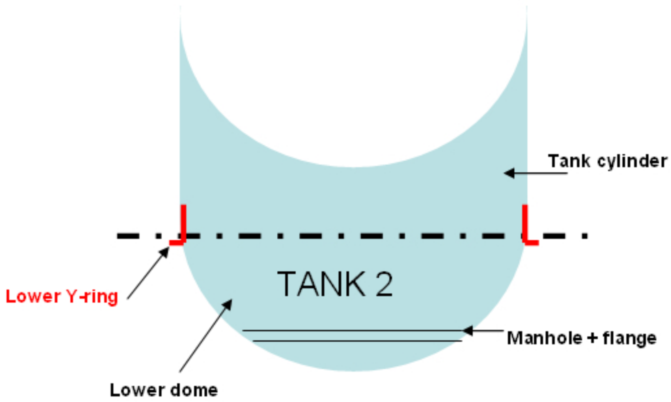


Figure 3.21: Common bulkhead tank 2 components

($p_{hydro} = \rho \, n \, g \, h_{cyl,2}$). The tank thickness for the construction concerning stress due to buckling is evaluated with the NASA SP-8007 method.
Again, the minimum required dome thickness is 1.3mm.

- Lower dome

  The lower dome mass is calculated from the dome surface area, the aluminum density and the evaluated thickness t_{dome}.

$$m_{dome} = \rho_{Alu} \, S_{dome} \, t_{dome} \, C \tag{3.60}$$

  The thickness of the lower dome is calculated as the two previous domes. The only difference is in the hydrostatic pressure: $p_{hydro} = \rho \, n \, g \, h_{cyl,2}$.

- Lower ring

  The lower Y-ring mass is scaled from an existing lower Y-ring by considering the diameter and the tank's internal pressure. The lower Y-ring is the connection between the tank structure and the skirt.

Spherical-cylindrical configuration
The mass of the spherical tank is calculated using the Aluminum's density, the spherical surface area and the spherical tank thickness.

$$m_{sph} = \rho_{Alu} \, S_{sph} \, t_{sph} \, C \tag{3.61}$$

The shell thickness of the the spherical tank for the construction concerning stress due to internal pressure is calculated using the Boiler formula.

$$t_{sph} = \frac{SF \, p_{max} \, r_{sph}}{2\sigma_{Alu}} \tag{3.62}$$

Where SF is a safety factor for production tolerance ($j = 1.1$) and p_{max} includes the ullage pressure (input), the dynamic pressure ($p_{dyn} = 0.5N/mm^2$) and the hydrostatic pressure ($p_{hydro} = \rho \, n \, g \, 2r_{sph}$).
Again, the minimum required dome thickness is 1.3mm.

The intermediate structure (using composite material)
The intermediate structure includes:

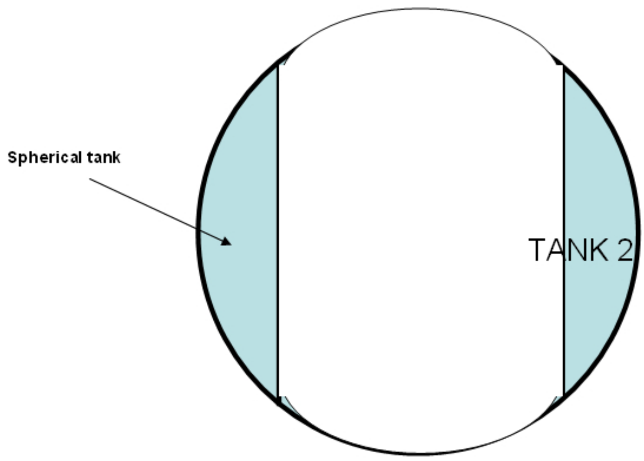


Figure 3.22: Spherical-Cylindrical tank 2 components

- The intermediate structure is made of a sandwich structured composite. So the structural mass of the intermediate stage is the sum of the face sheet mass and the core material mass.

$$m_{IS} = 2\pi(t_{IS,fs}\, d_{sec,2}\, l_{sec,2}\, \rho_{skin} + t_{IS,core}\, d_{sec,2}\, l_{sec,2}\, \rho_{core})C \tag{3.63}$$

  with
  $d_{sec,2}$ section 2 diameter $[m]$
  $l_{sec,2}$ section 2 length $[m]$
  $t_{IS,core}$ core thickness $[m]$
  $t_{IS,fs}$ face sheet thickness $[m]$
  ρ_{core} core density $[kg/m^3]$
  ρ_{fs} face sheet density $[kg/m^3]$
  C contingency factor

  The face sheet thickness is calculated using the NASA SP-8007 method. A minimum thickness is required and computed to 1.3mm. Concerning the core thickness, a reasonable approximation is done.

- Upper and lower rings

  The upper and lower rings connect the the intermediate structure to the other structural components above and below. These components are scaled from existing rings by using the diameter [11].

- Bolts and screw

  The mass of these components are estimated.

The engine thrust frame
The engine thrust frame is scaled from an existing ETF by using the surface area.

The interstage skirt
The interstage skirt includes:

- Interstage structure

  The interstage structure is made of a sandwich structured composite. The calculations are the same as the ones for the intermediate structure. Differences are the diameter and the length of the considered section.

- Upper and lower rings

  The upper and lower rings connect the intermediate structure to the other structural components above and below. These components are scaled from existing rings by using the diameter.

- Bolts and screw

  The mass of these components are estimated.

- Rivets

  The rivets are scaled from existing rings by using the diameter.

Chapter 4

Code Explanation

As explained previously, the launcher performances need to be evaluated, therefore a code has been written using the MATLAB software. By setting the launcher characteristics and the time when the gravity turn is applied, the MATLAB code enables calculations and representation of the rocket trajectory as well as the elliptical orbit characteristics that the launcher reaches. This chapter explains the written code.

The code has been changed and improved during the 6 months of this master thesis. The main changes have been done on the gravity turn program. There are actually two code versions depending on the gravity turn program: one where the rate of the gravity turn program is a constant and one where this rate depends on the other variables and so is a function of time. This chapter explains the code version with a non constant gravity turn which is the latest version of the code. A user's manual has been written and can be found in references [26].

The MATLAB code consists of three different folders: the VEGA folder, the ARIANE 5 folder and the SmallLauncher folder. In each of those folders, there are again two folders: one for a basic simulation and one for an optimization simulation. There are three major folders because the VEGA launcher and the ARIANE 5 launcher have specific properties that a small launcher does not have. For instance, the VEGA launcher has got four stages compared to the small launcher which shall be designed with two or three stages. Furthermore, the ARIANE 5 launcher has got two boosters which burn simultaneously then the second stage. In contrast, the small launcher burns its stages one after one. But it is important to notice that these folders are based on the same code, there are just minor modifications in the VEGA and ARIANE 5 folders in order to take into account the specific properties.
The following code explanations will focus on the SmallLauncher folder.

4.1 The Simulation Folder

The two important files in the simulation folder are the Main file and the RocketData file. The other files are functions. The RocketData file describes the rocket data and important constants, so it contains all the inputs except the inputs concerning the gravity turn program:

- Number of boosters N
- Specific impulsion of each stage $\begin{pmatrix} Isp_1 & Isp_2 & Isp_3 \end{pmatrix}$ $[s]$
- Propellant mass of each stage $\begin{pmatrix} mp_1 & mp_2 & mp_3 \end{pmatrix}$ $[kg]$
- Structural mass of each stage $\begin{pmatrix} ms_1 & ms_2 & ms_3 \end{pmatrix}$ $[kg]$

```
%%% Launcher DATA
clear all
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%

%Constants
g0 = gravity(0);%gravity field at sea level [m.s-2]
RE = 6371e3;    %Earth radius [m]

%initial altitude [m]
h_0=200;
%Desired inclination orbit
inc=0;
%initial velocity [m/s]
V_0=2*pi*RE/(24*3600)*sin(inc*pi/180);

%Stages Datas
N=3;%number of stages
Isp = [280.6 345 320.3 320.]; %Specific impulsion of the different stages [s]
ms  = [1493 2227.8 586.0662]; %Structural mass of the different stages [kg]
mp  = [11000 11187 1810]; %Propellant mass of the different stages [kg]
Thrust = [517.3 420 16]*1000; %Thrust [N] possibility of using fitted curves
mdot=[];
tb=[];
for i=1:N
     mdot(i)=Thrust(i)/(Isp(i)*g0);
     tb(i)=mp(i)/mdot(i);
end

%Payload mass
mpl=250;

%Geometric data
D = [2 1.5 0];
A = pi*D.^2/4;

%Aerodynamic coefficient
Cp=0.3; % Drag coeficient

%fairing characteristics
mfair=180;
tfair=110;
```

Figure 4.1: Rocketdata file

- Thrust of each stage $\begin{pmatrix} T_1 & T_2 & T_3 \end{pmatrix}$ $[N]$
- Mass flow rate of each stage $\dot{m} = \frac{T}{Isp\, g_0}$ $[kg/s]$
- Burn time of each stage $t_b = \frac{mp}{\dot{m}}$ $[s]$
- Required propellant to perform the circularization $m_{p,circ}$
- Fairing mass $[kg]$
- Fairing dropped time $[s]$
- Drag coefficient $[-]$
- Lateral surface of each stages $[m^2]$
- Earth radius R_e $[m]$
- Desired orbit inclination $[rad]$
- Initial velocity $V_0 = \frac{2\,\pi\, R_e}{24*3600}\, sin(inc)$ $[m/s]$

In Figure 4.1, the RocketData file is represented.

The functions files are:

area The area function returns the lateral area of the rocket depending on the time.

dens The dens function returns the local density depending on the altitude of the rocket.

gravity The gravity function returns the local gravity depending on the altitude of the rocket.

zval The zval function forces the zero value.

Thr The Thr function returns the thrust of the rocket depending on the time.

mass The mass function returns the mass of the rocket depending on the time.

rock1 The rock1 function consists of the system of equations of motion before and after the pitch program.

rock2 The rock2 function consists of the system of equations of motion during the pitch program.

stop The stop function enables to get the time when the circularization needs to be done, i.e. when the satellite reaches the apogee of the elliptical orbit.

All the function files screen shots are in Annex C.

The simulation steps are summarized in Figure 4.2. In the Main file is the code to run a simulation and to plot different variables as a function of time. There are two parameters which can be changed in the Main file and those parameters correspond to the pitch program. It is possible to change the time when the pitch program is applied.

First the Main file runs. It saves and loads all the variables from the RocketData file, then a code has been written in order to solve numerically the system of differential equations, the numerical solver used is the MATLAB solver ode45. This solver is a numerical solver based on an explicit Runge-Kutta (4,5) formula. It is a one step solver, that means it only needs the solution at the immediately preceding time point $y(t_{n-1})$ in order to compute $y(t_n)$. It solves a systems of non linear differential equations $\dot{y} = f(t, y)$. In this simulation the state vector is $y = \begin{pmatrix} V & \gamma & X & H \end{pmatrix}$.

The basic simulation is divided in three parts. The first part of the simulation is from 0 seconds to the beginning of the pitch program, the solver resolves the equations written in the **rock1** function. The equations taken into account are the equations 3.4, 3.5, 3.6 and 3.7. The rocket trajectory is a vertical ascension. The flight path angle stays constant during this interval.

The second part is from the beginning of the pitch program to the end of the pitch program, the solver resolves the equations written in the **rock2** function. The equations taken into account are the equations 3.4, 3.5 and 3.6. In order to apply the pitch program a term is added in the equation 3.7.

$$V\,\dot{\gamma} = -g\cos(\gamma) + \frac{V^2}{R_E + H} cos(\gamma) - \dot{\gamma_0} \tag{4.1}$$

The additional term is $\dot{\gamma_0}$ and it represents the rate of decrease of the flight path angle. This rate is a function of the other variables and so it is a function of time as well.

The last part is from the end of the pitch program to the end of the added time burning of each stage, the solver resolves the equations written in the **rock1** function. At the end of the simulation, a code is used in order to calculate which orbit the rocket has reached, so the perigee and the apogee are calculated as explained previously. At the end of the simulation, the rocket has reached the transfer elliptical orbit (point A on Figure 4.3).

Then there is also a possibility to perform the circularization of the orbit. The purpose of the small launcher is to put the satellite in a circular orbit with a radius of 700 km. In order to do that, the launcher first reaches an elliptical orbit and then there is a re-ignition of the last stage to circularize the orbit. But first, it is primordial to know at what moment the circularization

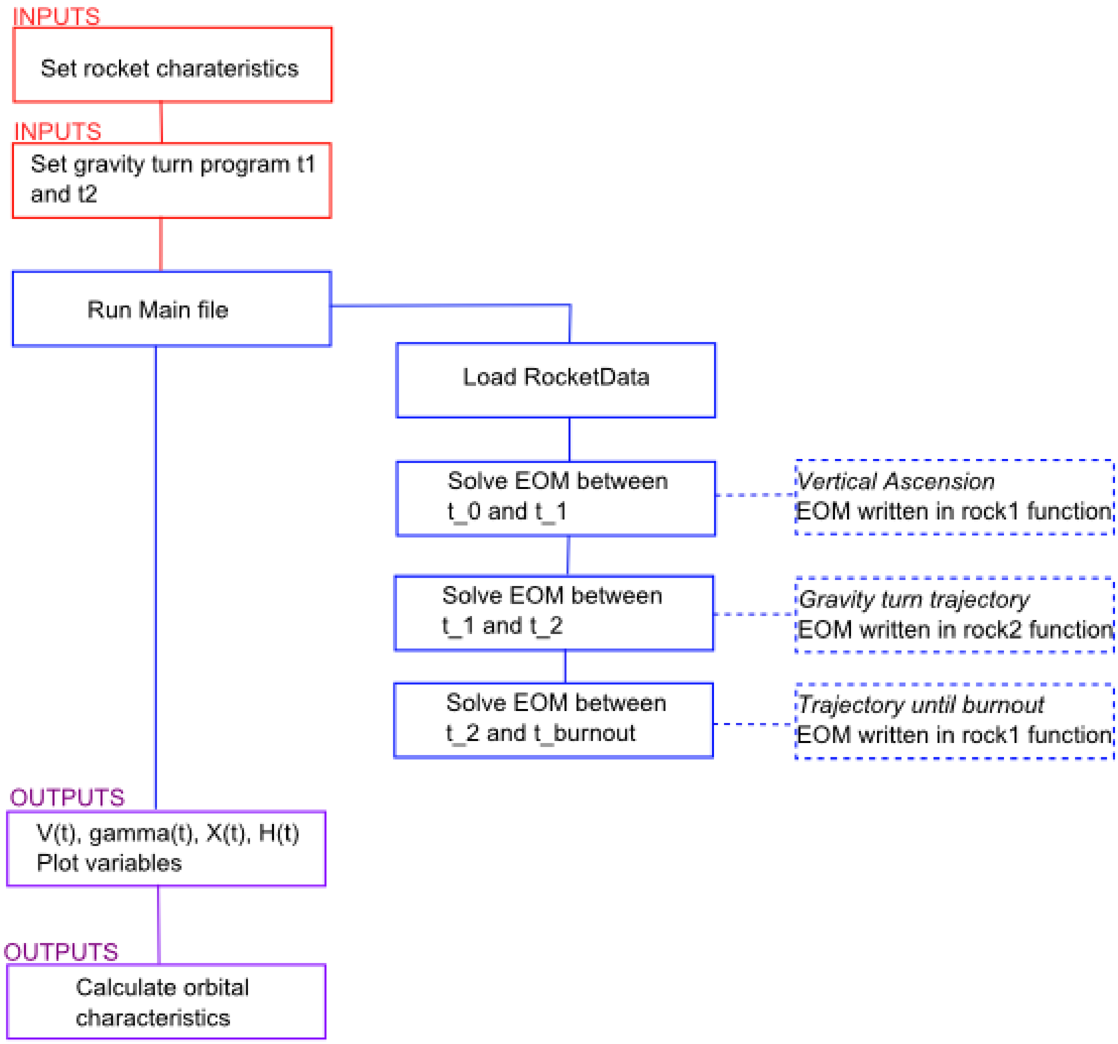


Figure 4.2: Matlab code explanation

has to be done. In order to find the time when the spacecraft reaches the apogee of the orbit a simulation is run from the end of the previous simulation to a certain time. This time has to be big enough in order to be sure that during the considered interval the spacecraft passes the apogee of the orbit. The solver is used with an option in order to know at what time the launcher reaches the apogee. The option uses the **stop** function. It is important to notice that during this phase, the launcher is in orbit so the thrust is equal to zero and the mass does not change. There is no need to change anything, the mass and thrust function are defined to handle the situation when the launcher is in orbit and when it has to be re-ignited as well. Once the time when the upper stage reaches the apogee is known, t_{circ}, the previous simulation is run again but this time until t_{circ}. Then another simulation is run from t_{circ} to $t_{circ} + 500$ in order to see the circularization. It is important to notice that the propellant needed for the re-ignition is considered in the code as part of the payload until it is used to perform the circularization.

In the Matlab workspace, all the variables can be seen. Some plots are already implemented to be shown at the end of the simulation.

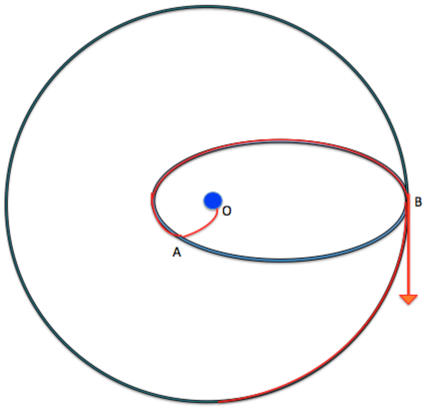


Figure 4.3: Rocket trajectory

4.2 The Optimization Folder

The rocket trajectory simulation in the optimization folder is done as explained previously. The only changes are most of the variables are defined as global variables in the main file in order to be changed whilst the optimization program is running. The Optimization folder contains the same functions as defined in the simulation folder. There is a main file in order to run a normal simulation and some optimization files depending on what variables can be changed.

The variables defined as global variables are:

- Specific impulsion of each stage $\begin{pmatrix} Isp_1 & Isp_2 & Isp_3 \end{pmatrix}$ $[s]$
- Propellant mass of each stage $\begin{pmatrix} mp_1 & mp_2 & mp_3 \end{pmatrix}$ $[kg]$
- Structural mass of each stage $\begin{pmatrix} ms_1 & ms_2 & ms_3 \end{pmatrix}$ $[kg]$
- Thrust of each stage $\begin{pmatrix} T_1 & T_2 & T_3 \end{pmatrix}$ $[N]$
- Mass flow rate of each stage $\dot{m} = \frac{T}{Isp\, g_0}$ $[kg/s]$
- Burn time of each stage $t_b = \frac{mp}{\dot{m}}$ $[s]$
- Fairing dropped time $[s]$
- Payload mass $[kg]$

There are several optimization files depending on what variables can be changed. The purpose of the optimization is always to reach a desired orbit. The optimization program is always at least done on the pitch program, but it is also possible to add others parameters, namely the payload mass and the propellant mass in the stages. The following description will describe the optimization file which optimizes the trajectory by changing gravity turn parameters. The other optimization files can be easily understood from this file.

Gravity turn optimization

First, the rocket data has to be defined at the beginning of the file. Then the vertices need to be defined. The vertices coordinates are the variables which will be changed in order to minimize the objective function. The vertices coordinates are:

- $tg(1)$: beginning of the pitch program

- $tg(2)$: end of the pitch program

The objective function is

$$F = (r_{a,calculated} - r_{a,desired})^2 + (r_{p,calculated} - r_{p,desired})^2 \tag{4.2}$$

where:	$r_{a,calculated}$	calculated apogee radius $[m]$
	$r_{a,desired}$	desired apogee radius $[m]$
	$r_{p,calculated}$	calculated perigee radius $[m]$
	$r_{p,desired}$	desired perigee radius $[m]$

As a vertex has two coordinates, three vertices are needed to do the optimization. They are defined as follows:

- $v_1 = \begin{pmatrix} tg_1 & tg_2 \end{pmatrix}$
- $v_2 = \begin{pmatrix} tg_1 + \epsilon_1 & tg_2 \end{pmatrix}$
- $v_3 = \begin{pmatrix} tg_1 & tg_2 + \epsilon_2 \end{pmatrix}$

So three simulations are run and respectively the three vertices and objective functions are calculated in the three cases. Then the coordinates of each vertices are changed in regards to the algorithm explained in the third chapter (Nelder-Mead algorithm). This is done a maximum of 200 times, and every 50 loops the optimization is stopped in order to control the convergence. If it seems there is no convergence, it is so possible to stop the optimization. This control is essentially done in order not to waste time as 200 loops take a substantial amount of time, around 10 minutes. The optimization stops by itself when the objective function is smaller than 0.01. The optimization loop is schematized in the figure 4.4.

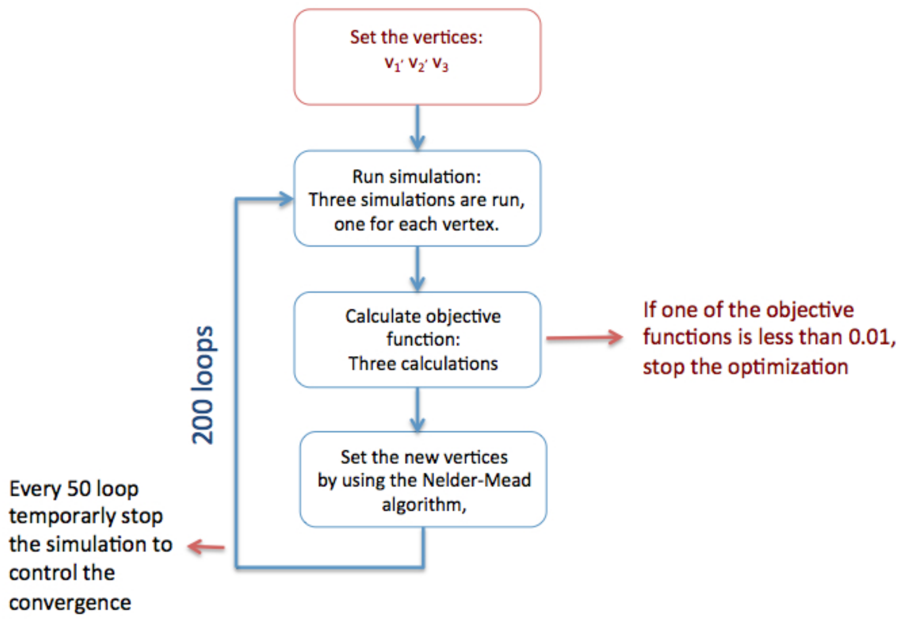


Figure 4.4: Optimization loop description

Chapter 5

Code Validation

The written code in Matlab has been validated by using the VEGA launcher and the Ariane 5 launcher. As said before, the code has been improved during the 6 months of this master thesis, and the main changes are on the rate of the gravity turn program. At first, the program used a constant gravity turn and it has been validated with the VEGA and Ariane 5 launchers. The results are presented in this chapter. Then, with the non constant gravity turn rate, the validation has been done once again using the VEGA launcher. These results are discussed with the constant gravity turn rate results in this chapter as well.

5.1 VEGA launcher simulation

A simulation with the VEGA launcher has been done in order to validate the code written in MATLAB. The goal of this simulation was to put 1500kg of payload in a circular orbit with a radius of 700km, by first reaching the transfer elliptical orbit with a perigee of 200 km and an apogee of 700 km. The characteristics of the VEGA launcher are shown in Table 5.1, namely the propellant mass, the dry mass, the specific impulse, the thrust, the burning time and the reference area of the different stages [9].

The thrust profile of the three first stages have been implemented in the code, they are represented in the Appendix A [14]. In order to reach a circular orbit of 700 km, the VEGA launcher first reaches the elliptical orbit with a perigee of 200 km and an apogee of 700 km by burning the first three stages and a part of the fourth stage, then a circularization is done by burning the propellant left in the fourth stage. In the written code, the payload takes into account the real payload plus the needed propellant to circularize the orbit. Table 5.1 shows the characteristics of the fourth stage, namely the propellant mass and the burning time needed to reach the elliptical orbit. In order to circularize the orbit, 100 kg of propellant is needed. The results have been

Entity	Stage 1	Stage 2	Stage 3	Stage 4
Engine	P80	Z23	Z9	AVUM
Propellant Mass [kg]	87,732	23,823	9,935	450
Dry Mass [kg]	8,624	2,666	1,397	661
Isp [s]	280	289	295	315.5
Thrust [kN]	2261	1196	225	2.45
t_b [s]	106.4	79.9	101.9	568.3
Reference Area [m^2]	7.07	7.069		
C_p [-]	0.310404	0.300009		

Table 5.1: Vega launcher characteristics

compared to the results from a DLR study [10].

5.2 VEGA launcher simulation results

A simulation has been done with the VEGA launcher in order to validate the MATLAB code. By using the optimization program, the best compromise between the payload, the trajectory and the launcher characteristics have been found. These results have been compared to the ones from a DLR study.

First, the optimization has been done on the gravity turn program, so by changing t_1, t_2 and the rate; and the results have been compared with the ones from the DLR study. From this optimization the VEGA launcher could reach an elliptical orbit with an apogee of 700km and a perigee of 118km with a payload of 1375kg. The stage characteristics used for this optimization are the ones used in the DLR study in order to be comparable. From the DLR study, the VEGA launcher could reach an elliptical orbit with an apogee of 700km and a perigee of 200 km with a payload of 1518kg.

The different variables, namely the speed, the flight path angle and the altitude, have been compared with the results form the DLR study, and it has been noticed that the results were significantly different from the second stage. For instance, as can be seen in Table 5.2, from the second stage there was a difference on the altitude of the launcher. In Table 5.2, the altitude, the speed and the flight path angle are written for the end of the three first stages. The results from this first optimization are listed in Table 5.2 at the line *Optimization*1. After this first optimization the VEGA reaches an elliptical orbit with a perigee of 135km and an apogee of 700km, and has a payload of 1371kg. The DLR study shows the VEGA reaches the elliptical orbit with a perigee of 200 km and an apogee of 700km with a payload of 1518kg.

	Stage 1			Stage 2			Stage 3		
	H [km]	V [km/s]	γ [deg]	H [km]	V [km/s]	γ [deg]	H [km]	V [km/s]	γ [deg]
DLR study	44	1.9	25	**110**	4.1	10	**162**	7.5	3
Optimization 1	44	1.89	20	**90**	4.1	6.3	**120**	7.4	1.2
Optimization 2	46	1.9	20	**97**	4.1	7.5	**134**	7.4	2.3

Table 5.2: Results comparison with the DLR study

Another optimization has been carried out by changing the variables of the gravity turn and the amount of propellant in the second and third stage as well. The results from this optimization are listed in Table 5.2 at the line *Optimization*2. After this second optimization the VEGA launcher reaches the elliptical orbit with a perigee of 200 km and an apogee of 700 km, and has a payload of 1371kg.

	Stage 2			Stage 3		
	m_p [kg]	m_s [kg]	m_{tot} [kg]	m_p [kg]	m_s [kg]	m_{tot} [kg]
DLR study	23,823.4	2,666.2	26,489.6	9,935.3	1,397.4	11,332.7
Simulation	23,291	2,666.2	25,957.2	9,335	1,397.4	10,732.4

Table 5.3: Stage 2 and 3 in comparison with the DLR study

The characteristics of the second and third stage have been compared with those from the DLR in Table 5.3. The propellant difference in the second stage is around 2% and in the third stage

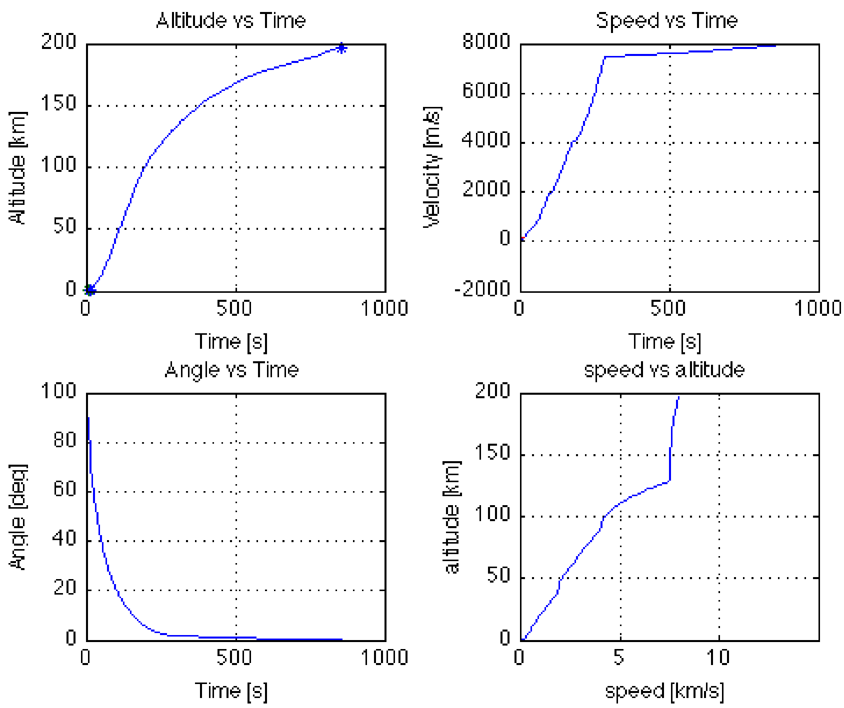


Figure 5.1: Vega launcher simulation from 0km to the elliptical orbit

it is around 6%. For the total amount of propellant, the difference is less than 1%. Compared to the real performance of the VEGA launcher, there is less than 10% of difference with the results from the simulation, so the mission is considered as validated.

The following figures show different variables depending on the time during the simulation. In Figure 5.1, the simulation results are shown from the launch pad to the elliptical orbit.

On Figure 5.2, the curves in red represent the trajectory of the fourth stage with the payload in orbit and thrust equal to zero. The curves in green represent the circularization.

On Figure 5.3, the spacecraft mass, the thrust, the dynamic pressure and the lateral and longitudinal velocity are represented as a function of time. Concerning the velocity curves, the blue curve is the longitudinal velocity and the red curve is the lateral velocity.

5.3 Ariane 5 launcher simulation

Another simulation has been done with the Ariane 5 launcher in order to validate the code. The goal of this simulation was to do a real mission executed by Ariane 5. The chosen mission is the mission realized by Ariane 5 on the 13th of October 2006. The specificity of Ariane 5 is that 2 solid boosters burn meanwhile the cryogenic stage burns. The drop of these two solid boosters is achieved around 130 seconds after the lift-off, and these two boosters fall down in the ocean. The real trajectory of the Ariane 5 launcher is not actually optimized in order to drop those two boosters in the ocean and not on a living area. In this simulation, the trajectory optimization has been done and so the results may be different from the reality.

The goal is to put 9031 kg of payload in an elliptical orbit with a perigee of 250 km, an apogee of 35942 km and an inclination of 7 degrees. Considered in the payload is the mass of the satellite and some equipment, namely the cone and the Syldas. The characteristics of the Ariane 5 launchers are summarized in Table 5.4 [8]. In this table are the propellant mass, the dry mass, the

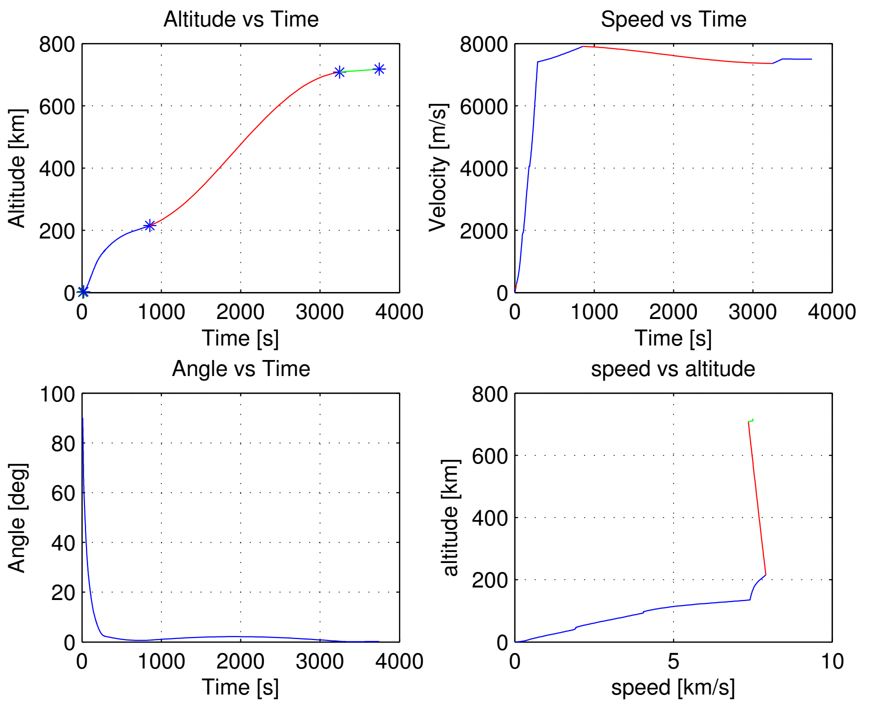


Figure 5.2: Vega launcher simulation from 0km to the circular orbit

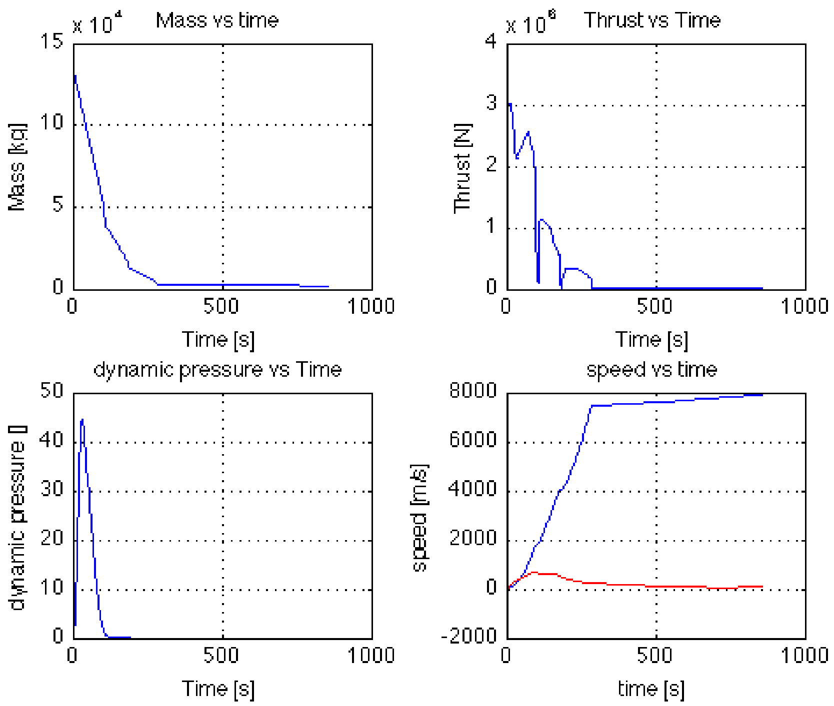


Figure 5.3: Vega launcher simulation, other characteristics

thrust, the specific impulse, the burning time and the reference area of the boosters and stages. In the Matlab code, the characteristics of the second stage are not the same in the function of the boosters are burning or have been dropped. While the two solid boosters are burning, the cryogenic stage characteristics are the one at sea level, after the boosters have been dropped, the cryogenic stage characteristics are the ones in the vacuum. This choice has been done in order to have a more realistic simulation.

Entity	Stage 1	Stage 2	Stage 3
Engine	EAP	EPC	ESC-A
Propellant mass $[kg]$	480,000	170,000	14,900
Dry mass $[kg]$	80,000	14,700	4,540
Thrust $[kN]$	14,000	960 (SL) 1390 (vac)	67
Isp $[s]$	274.5	310 (SL) 432 (vac)	446
t_b $[s]$	129	537	972
Reference Area $[m^2]$	14.61	22.90	
C_p [-]	0.3	0.3	

Table 5.4: Ariane 5 launcher characteristics

The thrust profile of the two solid boosters have been implemented in the code (cf. Annex B). The optimization has only been done on the gravity turn program (t_1, t_2, rate) as there was no special difficulties to reach the desired orbit with the desired payload.

5.4 Ariane 5 launcher simulation results

The optimization has only been done on the gravity turn program. At the end of the optimization, the Ariane 5 launcher could reach the elliptical orbit with a perigee of 250km and an apogee of 35942km with a payload of 9670kg. There is less than 10% of difference between the results from the simulation and the results from the real mission, so the mission is considered as validated.

Represented in figure 5.4 is the spacecraft velocity, the flight path angle and the altitude as a function of time. The fourth curve corresponds to the altitude as a function of velocity. Represented in Figure 5.5 is the mass, the thrust, the dynamic pressure and the velocities as functions of time. On the last curve, the blue curve corresponds to the longitudinal velocity whereas the red curve correspond to the lateral velocity.

In the following table, are the spacecraft speed, flight path angle and altitude at the burn out of the different stages.

Entity	Boosters	Stage 1	Upper stage
Time $[s]$	129	522	1495
Velocity $[km/s]$	2.17	7.02	9.97
γ $[deg]$	42.39	-1.14	9.74
Altitude $[km]$	125	389	509

Table 5.5: Ariane 5 trajectory characteristics at stage burn-out

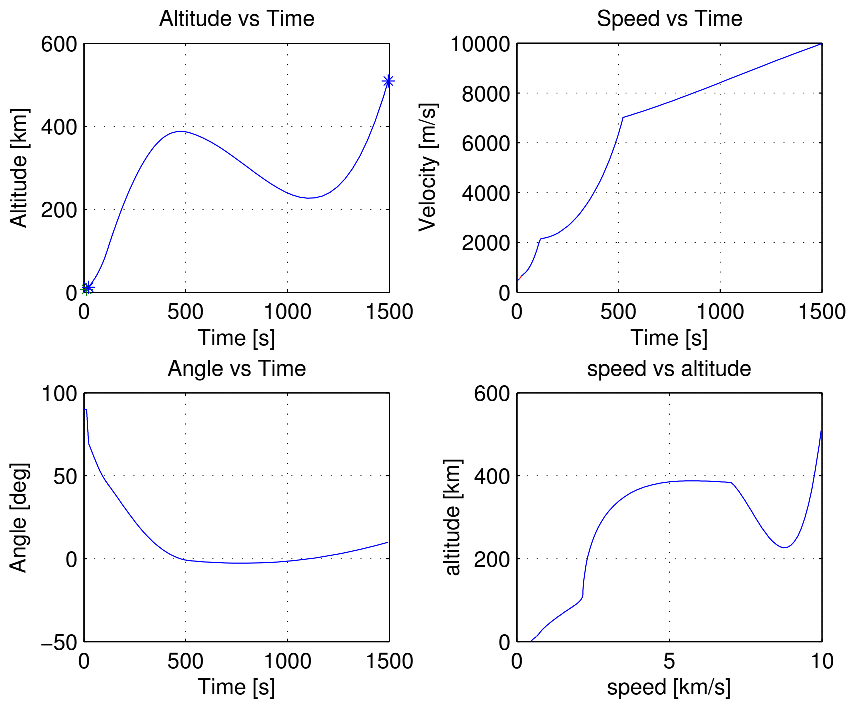


Figure 5.4: Ariane 5 launcher simulation from 0km to the elliptical orbit

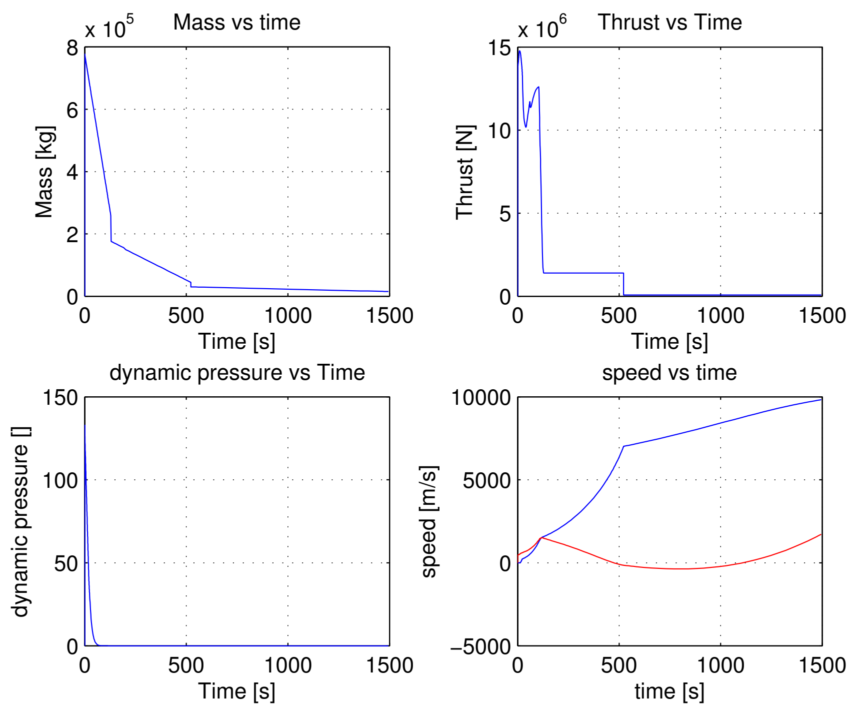


Figure 5.5: Ariane 5 launcher simulation, other characteristics

5.5 Discussion

From the simulation done with the VEGA and Ariane 5 launchers, the code has been validated as the results were less than 10% difference with the real missions. However, some comments can be made on these simulations. First, the simulation concerning the VEGA launcher, the mission was more difficult to realize than the mission concerning the Ariane 5 launcher. That is why a second optimization needed to be done with the VEGA launcher. Moreover, in reality, the Ariane 5 trajectory is not optimized. Indeed, if it was optimized, the stage will be dropped somewhere in Africa, but it is necessary to drop the stage somewhere in the ocean. From the simulation which has been done, the trajectory is so optimized and it has been checked that the stage is dropped in this simulation in Africa. But no simulation has been done in order to simulate a trajectory where the drop will be in the ocean. This is because the payload mass is greater in the simulation than in the real mission.

This code has been validated and used at first for the simulations on the small launcher. However, during this master thesis, from discussions with engineers with a better understanding and knowledge on launcher and trajectories, the gravity turn program has been changed and improved. Indeed, instead of considering the rate of flight path angle decreasing as a constant, this rate has been considered as a function of other variables, namely speed, flight path angle and altitude; so the rate is a function of time as well. Simulations with the VEGA launcher have been done once again in order to see the differences with the previous code. The code is still validated and gives better results. With a non constant rate in the gravity turn program, the VEGA launcher can place in orbit 1440 kg of payload in the elliptical orbit with a perigee of 200km and an apogee of 700km. It has to be noticed that the launcher characteristics do not need to be changed and so there are exactly the same than the ones used in the DLR study.

Chapter 6

Results

The objective of this diploma thesis is to develop a conceptual lay-out of a small launcher. The method, described in the previous chapters, is used to analyze the small launcher characteristics. The MATLAB tool is used to simulate the trajectory and to optimize the launcher staging. The upper stage is designed thanks to an existing Excel tool as explained previously. Several optimum small launchers have been designed, the results are summarized and discussed in the following chapters.
Thereafter the most appropriated small launcher is selected, the upper stage is described in more detail.

6.1 Existing Stages

A requirement for the mission is to use existing stages, research has been done to investigate the actual stages. Solid and liquid stages have been taken into account.

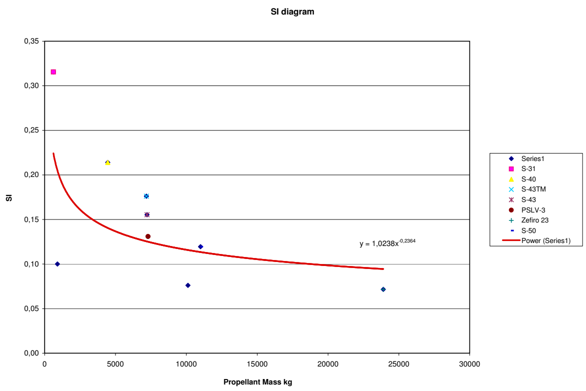
6.1.1 Solid stages

Existing solid boosters have been investigated and more precisely the study has been focused on small boosters. As the purpose of the mission is to developed a small launcher, there was no need to focus on high thrust boosters. The following table summarizes the characteristics of the investigated solid boosters, namely the specific impulse, the thrust, the propellant mass, the dry mass, and the structural index. From this table, it can be seen than mostly the Brazilian and Italian stages are investigated [2].

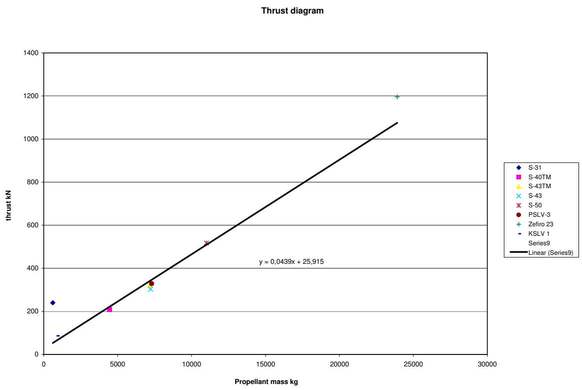
	S-31 Brazil	S-40 Brazil	S-43TM Brazil	S-40 Brazil	S-50 Brazil	PSLV-3 India	Zefiro 23 Italy	Zefiro 9 Italy
Isp $[s]$	260	275	277	260	280	294	289	295
Thrust $[kN]$	240	208	320	303	517	329	1196	225
m_p $[kg]$	616	4452	7184	7222	11000	7300	23906	10115
m_{dry} $[kg]$	284	1212	1536	1328	1493	1100	1845	833
SI $[-]$	0.32	0.21	0.18	0.16	0.12	0.13	0.07	0.08

Table 6.1: Solid boosters characteristics

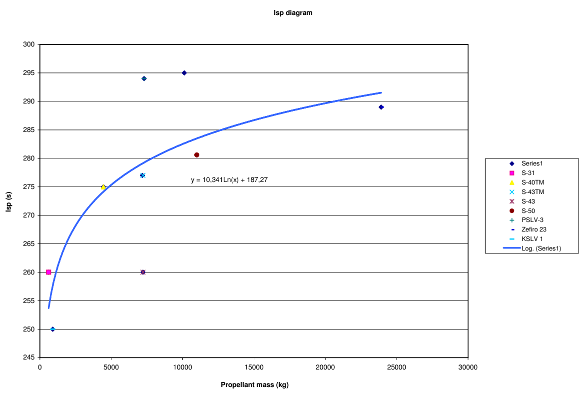
From the solid stage data, the optimum curves have been defined as explained previously. In Figure 6.1, theses optimum curves are represented, so the thrust, the specific impulse and the structural index as a function of the propellant mass. The equations which represent the optimum curves are written as well. Those equations have been implemented in the MATLAB code in order to find the optimum launchers.



$$SI = 5.3366\, m_p^{-0.4146}$$

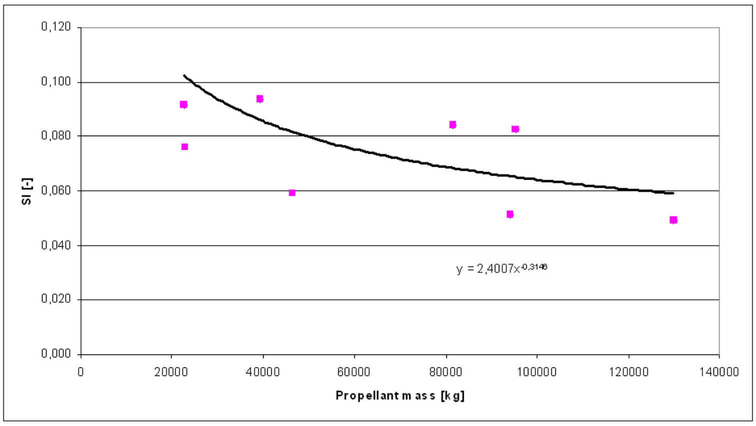


$$T = 0.0444\, m_p + 18.727$$

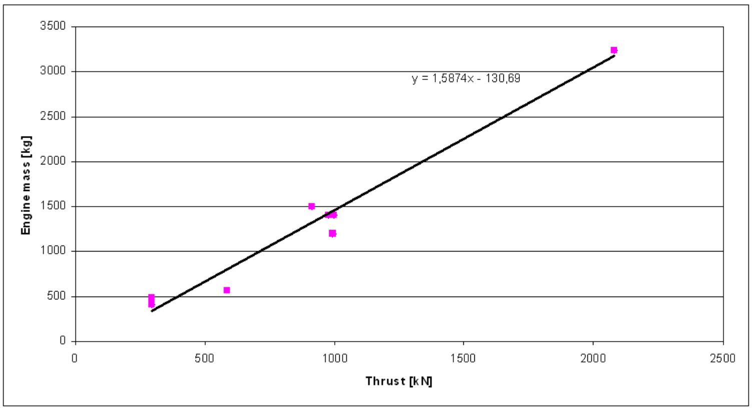


$$Isp = 8.4524\, ln(m_p) + 204.77$$

Figure 6.1: Optimum curves for solid stages



$2.4007 m_p^{-0.3146}$



$1.5873T - 130.69$

Figure 6.2: Optimum structural index curve and engine mass for liquid stage

6.1.2 Liquid stage

The same investigation, as previous, has been done for the liquid stages, also focusing on small thrust engines. The results of this study are shown in Table 6.2 [2].

	Angara stage 2	Proton stage2	Zenit stage 2	Zenit booster	Soyuz booster	Soyuz stage 1	Soyuz stage 1 U	KSLV I stage 1
Isp $[s]$	359	326	349	330	310	315	311	338
Thrust $[kN]$	294	583	912	849	994	977.7	997	2079
m_p $[kg]$	22845	46532	81600		39246	94000	95405	130000
m_{dry} $[kg]$	1875	2934	7495		4054	5100	8595	10000
m_{engine} $[kg]$	480	566	1505	1080	1200	1400	1400	3230
SI $[-]$	0.076	0.059	0.084		0.094	0.051	0.083	0.049

Table 6.2: Liquid boosters characteristics

In this table the Soyuz stage 1 U is a derivative stage from the Soyuz stage 1 with some modifications. The Zenit booster is derived from the Zenit second stage in order to have a first stage with similar technology. Actually the Zenit second stage presented in this table has one main engine and four steering engines. The total thrust of the Zenit stage 2 is the total thrust of these 5 engines. The Zenit booster will only have one main engine which is derived from the main engine of the Zenit stage 2.

From the liquid stage data, the curves concerning the structural index and the engine mass have been evaluated. Indeed, as explained previously, in the simulation, the thrust and the specific impulse are considered constant, and only the structural mass is a variable. So, the structural mass depends on the propellant mass and the engine mass. In Figure 6.2, the optimum structural index curve as a function of the propellant mass and the engine mass as a function of the thrust are represented.

6.2 Upper Stage Preliminary Design

As explained previously in the method, two upper-stages have been designed as a preliminary design in order to evaluate the relations between the structural mass and the propellant mass and thereby to be able to run simulations. The upper-stage design will be explained later on.

Two engines have been investigated in the preliminary design of the upper stage, one with a 8kN engine and another with a 16kN engine. Three possible tank configurations have been considered but at the end one configuration has been chosen for the design of the upper stage. The tank configuration in question is the common bulkhead configuration with a load bearing propellant tank. Other assumptions have been made as well regarding the fairing or the geometry for instance.

Diameter

The diameter of the upper stage has been set as a function of the diameter of the existing stages, the engine geometry, the tank volume and a design requirement angle. The angle between the stage structure and from the exit nozzle area to the stage / interstage structure must be greater than 10 degrees in order to guarantee no impact between the interstage structure and the nozzle during the stage separation. The angle is usually around 15 degrees on the existing upper-stages. The exit nozzle diameter is 638mm for the 8kN thrust engine and 990mm for the 16kN thrust engine. A diameter of 1.2 meters has been chosen for the first design.

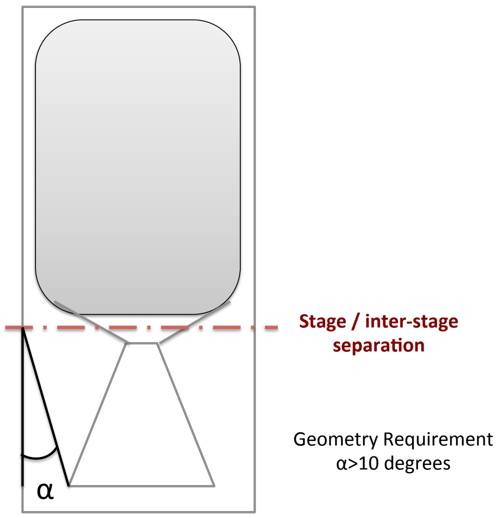


Figure 6.3: Angle requirement geometry

As the diameter is fixed, the separation system is placed in order to respect the design requirement angle presented previously.

Fairing

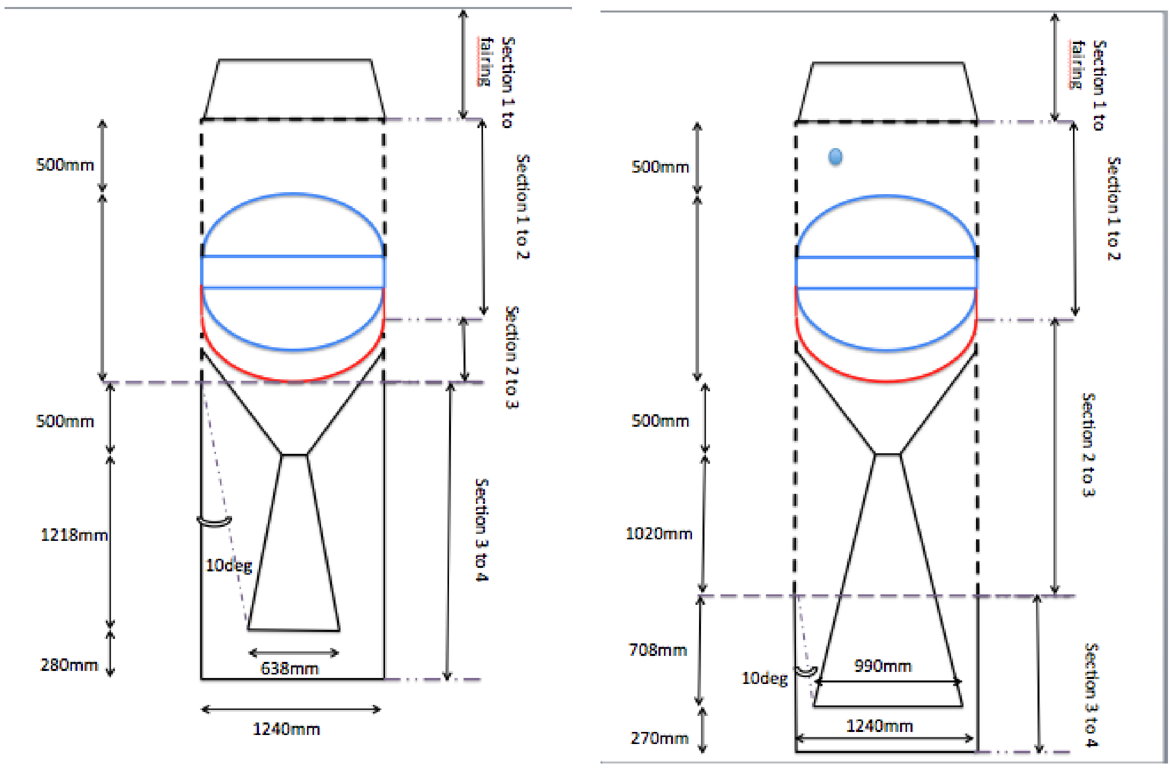
The determination of the fairing characteristics have been done by scaling the fairing of AVUM, the VEGA launcher upper stage. Table 6.3 presents the fairing characteristics of the AVUM upper stage and the ones chosen for the new small launcher upper stage.

Entity	Designed upper stage	AVUM
Stage diameter $[m]$	1.2	1.9
Fairing mass $[kg]$	180	490
Fairing length $[m]$	3.2	7.88
Fairing diameter $[m]$	1.2	2.6
Payload $[kg]$	250	1500

Table 6.3: Fairing Characteristics

Upper-stage design

The two designed upper stages are represented in Figure 6.4. It has been decided to leave a space of 500mm between the top of the tank and the upper cone in order to place the helium vessels and others systems. The thrust frame length has been chosen equal to 500mm.



(a) 8 kN thrust engine (b) 16 kN thrust engine

Figure 6.4: The upper stage design

From these two upper stages, the total structural mass has been implemented in the code as a function of the propellant mass loaded and the engine thrust.

$$m_{s,upperstage} = m_s(m_p) + m_{engine}(F) \tag{6.1}$$

where		
	F	Engine thrust $[N]$
	m_s	Structural mass w/o engine $[kg]$
	$m_{s,upperstage}$	Structural mass (engine mass included) $[kg]$
	m_p	Propellant mass $[kg]$

Engine mass $[kg]$	Engine thrust $[kN]$
20	3
34.3	8
70	16
140	55

Table 6.4: Engine mass and thrust datas

Table 6.4 shows the engine mass depending on the engine thrust. These values are from engines which are under investigation in the company. From these values, a function representing the engine mass depending on the thrust has been calculated. It is represented in Figure 6.5 and the equation is

$$m_{engine} = 2.2333\,F + 20.293 \tag{6.2}$$

where		
	m_{engine}	Engine mass $[kg]$
	F	Engine thrust $[kN]$

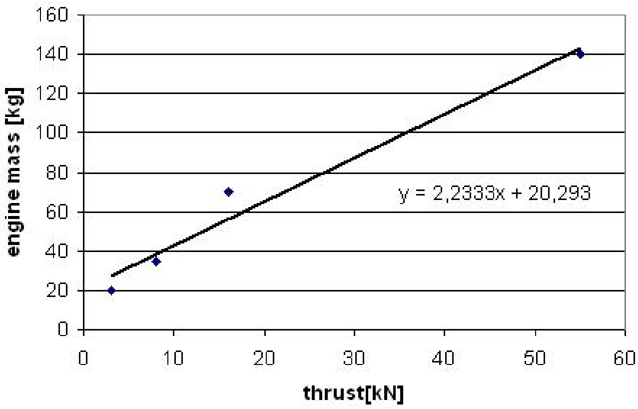


Figure 6.5: Engine mass as a function of the engine thrust

Table 6.5 shows the structural mass of the designed upper stage depending on the propellant mass. For the two different upper stages, the structural mass has been designed for several values of propellant mass, and the structural index has been calculated. In Figure 6.6, the structural index values as a function of propellant mass are represented. A power trend-line has been defined, the equation of this trend line is:

$$m_{dry} = 7.6139\,m_p^{-0.4266} \tag{6.3}$$

where		
	m_{dry}	Structural mass w/o engine w/o interstage structure $[kg]$
	m_p	Propellant mass $[kg]$

In the MATLAB code, the structural mass of the upper stage has been implemented as a function of the propellant mass and the engine thrust. The interstage structural mass has been considered constant and equal to 60kg. Depending on the engine used, the interstage structural mass is not the same, but those changes are small and so for the first estimation, this interstage mass is constant but it will be more defined with the final upper stage.

8kN			16kN		
m_p	m_{dry}	SI	m_p	m_{dry}	SI
500	572	0.5336	1200	749	0.3843
1100	683	0.3830	1500	782	0.3427
1300	705	0.3516	1800	816	0.3119
1600	738	0.3157			

Table 6.5: Structural mass calculation with the 8kN and 16kN thrust engine

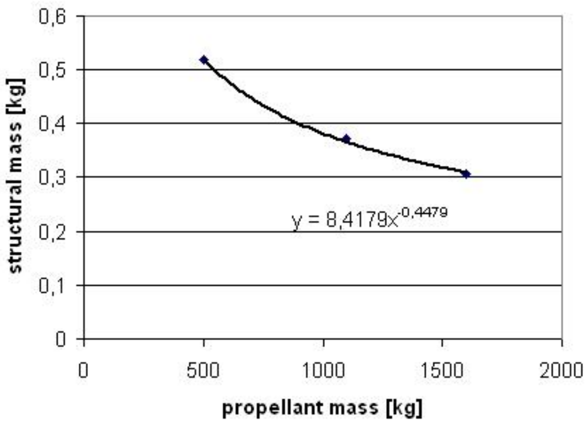


Figure 6.6: The structural mass as a function of propellant mass

6.3 3-stage launcher solution

The purpose is to propose a solution with two solid stages and a liquid upper stage.

6.3.1 Initial values, constants and inputs

The thrust, the specific impulse and the structural mass are calculated from the optimum curves for the first and second stage. The structural mass of the upper stage is calculated from the propellant mass and the thrust of the upper stage. The specific impulse is constant and equal to 320.9. The specific impulse is considered constant because the engines under investigation at the company have almost the same specific impulse of around 321, thereby for the optimum launcher calculations the specific impulse of the upper stage is considered as a constant.

The initial velocity is settled to zero as the required orbit has an inclination of 90 degrees. The drag coefficient is constant and equal to 0.3. The reference area is equal to $7.06858m^2$ for the first stage and $1.76715m^2$ for the second stage, which correspond to a diameter of 3 meters for the first stage and 1.5 meters for the second stage. The payload is constant and equal to 250 kg.

The inputs are

- the beginning and the end of the gravity turn program: t_1 and t_2
- the propellant mass in the first stage: $m_{p,1}$

- the propellant mass in the second stage: $m_{p,2}$
- the propellant mass in the third stage: $m_{p,3}$
- the thrust in the third stage: F_3

The optimization program used for the optimum launcher simulation has 6 parameters: two parameters from the gravity turn program, three parameters from the propellant mass in the three stages and one parameter for the thrust in the upper stage.

$$v_1 = \begin{pmatrix} t_1 & t_2 & m_{p,1} & m_{p,2} & m_{p,3} & F_3 \end{pmatrix} \tag{6.4}$$

6.3.2 Optimal launchers

The purpose is to find an optimum launcher with 2 solid stages and a liquid upper stage. This launcher is called optimum because its characteristics are calculated from optimum curves which are defined due to research. Simulations have been run by following the tree method explained previously, thereby optimum configurations have been found. The optimization program based on the simplex method only finds a local optimum, so depending on the first vertex, the optimum launcher is calculated. That is why the tree method has been defined in order to be sure to sweep a range of values without forgetting any possible optimum launcher. From this method the goal was to see if the convergence is done most of the time in the same direction. 50 simulations have been run based on the tree method. So not all the possible branches of the tree have been tried, but the results from these 50 simulations enabled to display the characteristics of an optimum small launcher.

The most important results are summarized in Table 6.6. In this table are the initial coordinates of the vertex and the optimum values found thanks to the optimization program. The payload ratio and the losses have been calculated and are written in this table. The losses include the change in velocity required to reach the orbit, the gravity losses and the drag losses.

Stage 1		Stage 2		Stage 3		Stage 3		Payload ratio	Losses
$m_{p,1,i}$ $[kg]$	$m_{p,1,opt}$ $[kg]$	$m_{p,2,i}$ $[kg]$	$m_{p,2,opt}$ $[kg]$	$m_{p,3,i}$ $[kg]$	$m_{p,3,opt}$ $[kg]$	$F_{3,i}$ $[kN]$	$F_{3,opt}$ $[kN]$	π $[\%]$	ΔV_{prop} $[m/s]$
22,000	24354	10,000	11,919	1,000	1,385	10	12.4	0.589	9154
22,000	24,258	10,000	11,880	1,500	1,514	15	13.7	0.589	9142
22,000	25,400	11,000	12,582	1,000	1,009	10	7.3	0.572	9380
22,000	23,772	11,000	12,780	1,500	1,288	10	10.8	0.587	9253
22,000	24,985	12,000	12,538	1,000	1,027	5	7.5	0.578	9116
22,000	23,952	12,000	12,670	1,500	1,206	10	7.7	0.588	9195
24,000	24,259	10,000	12,027	1,000	1,388	10	10.7	0.589	9152
24,000	24,460	10,000	11,222	1,500	1,768	15	14.5	0.592	9157
24,000	25,373	11,000	12,707	500	940	5	5.7	0.572	9167
24,000	24,498	11,000	11,642	1,500	1,473	10	11.3	0.590	9151
24,000	25,594	12,000	12,115	1,000	1,051	5	5.9	0.575	9239
24,000	23,848	12,000	12,102	1,500	1,468	15	13	0.593	9119
26,000	25,324	10,000	10,975	1,500	1,613	15	14.8	0.585	9216
26,000	25,987	12,000	12,905	500	825	5	5.3	0.562	9122
26,000	24713	12,000	11,426	1,500	1,575	15	14.4	0.588	9141

Table 6.6: Optimum launchers

From these results, it can be seen that the convergence is done in the same direction

- The propellant mass is the first stage is between 23,800kg and 25, 900kg.
- The propellant mass in the second stage is between 10,900kg and 12,900kg.
- The propellant in the upper stage is between 800kg and 1700kg.
- the thrust in the the upper stage is between 5.3kN and 14.8kN.

The existing solid stages which are the closest one from the optimum solution must be chosen. By comparing these results with the data from the existing solid stage, 3 boosters are selected: the italian boosters Zefiro 23 and Zefiro 9, and the brazilian boosters S-50 and S-43.
The characteristics of these boosters are summarized in the following table.

Entity	Z 23	Z 9	S-50	S-43
Isp $[s]$	289	295	280.6	260
Thrust $[kN]$	1,196	225	517	303
m_p $[kg]$	23,906	10,115	11,000	7,222
m_{dry} $[kg]$	1,845	833	1,493	1328

Table 6.7: Characteristics of the chosen solid stages

So, the possible launcher configurations are:

- Zefiro 23 + Zefiro 9 + Upper stage
- 2*S-50 + S-50 + Upper stage
- (S-50 & 2*S-43) + S-50 + Upper stage

Other configurations are possible by mixing the Zefiro 23 and the S-50, but as these two stages are from two different countries, it seems reasonable to think they could be conflicts on the realization of such a launcher so these configurations will not be taken into account in the following study.

The purpose now is to focus on the upper stage in order to design it in the best interest of the mission.

Solution 1: Zefiro 23 + Zefiro 9 + Upper stage

In this section the results concerning the small launcher with the Zefiro 23 and 9 boosters are presented. The purpose is to design more precisely the upper stage which could be used for the purpose of the study. That means the thrust and the propellant mass of the upper stage have to be defined and so the structural mass has to be designed in more detail. For this purpose, new simulations have been run. Now, the characteristics of the first and second stage are fixed (thrust, specific impulse, structural mass), the amount of propellant can be modified in these two stages if needed. The characteristics of the upper stage are calculated at first as previously in order to optimize them. From the optimization the upper stage characteristics will be chosen.

At first, simulations have be run with a payload of 250kg, the characteristics of the first and second stages are fixed, and the optimization was on the gravity turn program and the thrust and propellant mass of the third stage. From these simulations, it has been noticed that the small launcher has too strong a performance in regards to the requirements, actually the launcher velocity at burn out was higher than the expected one and so the launcher could reach a higher orbit. So, two modifications could be done:

- Increase the payload in order to see what could be the maximum payload with the launcher
- Decrease the amount of propellant in the second and third stage.

Simulations and optimizations have been done for both these changes. An optimization has been done on the gravity turn program, the upper stage (thrust and propellant mass) and the payload mass in order to see what could be the maximum payload put into the desired orbit with this small launcher. The results are shown in Table 6.8. This configuration enables to put in the required orbit 290 kg of payload, which is 40 kg more than the desired payload. In this configuration the payload ratio is 0.761%.

Propellant mass [kg]			Dry mass [kg]			Thrust [kN]			Payload [kg]	Payload ratio [%]
1	2	3	1	2	3	1	2	3		
23,906	10,115	438	1,845	833	557	1,196	225	2.9	290	0.761

Table 6.8: Solution 1, Optimization on the gravity turn, upper stage and payload

So in order to keep the required payload, another optimization can be done by decreasing the amount of propellant in stage 1 and 2. Another optimization has been done on the gravity turn the propellant mass in stage 1, 2, and the upper stage; the payload is fixed and equal to 250kg. The results from this optimization are shown in Table 6.9.

Propellant mass [kg]			Dry mass [kg]			Thrust [kN]			Payload [kg]	Payload ratio [%]
1	2	3	1	2	3	1	2	3		
23,855	9,932	465	1,845	833	557	1,196	225	2.7	250	0.662

Table 6.9: Solution 1, Optimization on gravity turn, stage 1, stage 2, upper stage

The propellant mass needs to be decreased by 0.2% in the first stage and of 1.8% in the second stage to be able to put maximum 250kg of payload on the required orbit. In this configuration the payload ratio is 0.662%. It has to be noticed that the structural mass for the first and second stages have not been changed in this optimization.

Solution 2: 2*S-50 + S-50 + Upper stage

In this section the results with the Brazilian boosters are presented. The configuration of this small launcher is to have 2 S-50 boosters at stage 1, one S-50 booster at stage 2 and an upper stage which has been defined in the purpose of the study. It has to be noticed that the first stage of this configuration is not in the range of the optimum launcher. The amount of propellant is smaller than what is needed with the optimum launcher.

At first, simulations have been run with a payload of 250kg, the characteristics of the first and second stage are fixed, and the optimization was on the gravity turn and the upper stage (thrust and propellant mass). From these simulations it appears that the launcher is not able to put 250 kg of payload in the required orbit. The payload has to be decreased. An optimization has been done to find the maximum payload which can be put on orbit. The results are shown in the following table.

Propellant mass [kg]			Dry mass [kg]			Thrust [kN]			Payload [kg]	Payload ratio [%]
1	2	3	1	2	3	1	2	3		
22,000	11,000	2,605	2,986	1,493	954	1,551	517	22.9	90	0.219

Table 6.10: Solution 2, Optimization on the gravity turn, upper stage and payload

As the propellant needed for the orbit circularization is around 35kg, that means this configuration can only put 55kg of payload on the desired orbit.
Another optimization has been done by altering the amount of propellant in the two first stages in order to see if it was possible to get an higher payload mass. The results are shown in Table 6.11. The payload is chosen and fixed in this optimization.

Propellant mass $[kg]$			Dry mass $[kg]$			Thrust $[kN]$			Payload $[kg]$	Payload ratio $[\%]$
1	2	3	1	2	3	1	2	3		
21,880	11,238	2,406	2,986	1,493	929	1,551	517	23.6	100	0.244

Table 6.11: Solution 2, Optimization on the gravity turn, stage 1, stage 2, upper stage

From this optimization, the propellant mass in the first stage has been decreased by 0.5%, and by 2.1% in the second stage, and the payload has increased by 11%.

Solution 3: (S-50 & 2*S-43) + S-50 + Upper stage

Another possible configuration is to have three boosters in the first stage, one S-50 and two S-43, one S-50 for the second stage and the liquid upper stage. The diameter of the first stage with this configuration is larger than the one considered when the optimum launcher was studied.

At first, simulations have been run with a payload of 250kg, the characteristics of the first and second stage are fixed, and the optimization was on the gravity turn and the upper stage (thrust and propellant mass). From these simulations it appears that the launcher is not able to put 250 kg of payload in the required orbit. The payload has to be decreased. An optimization has been done to find the maximum payload which can be put on orbit. The results are shown in the following table.

Propellant mass $[kg]$			Dry mass $[kg]$			Thrust $[kN]$			Payload $[kg]$	Payload ratio $[\%]$
1	2	3	1	2	3	1	2	3		
25,444	11,000	2,127	4,149	1,493	873	1,123	517	18.9	147	0.323

Table 6.12: Solution 3, Optimization on the gravity turn, upper stage and payload

The payload which can be put on the desired circular orbit would be around 110kg with this launcher configuration.

Another optimization as been done on the amount of propellant of the two first stage as well. In this optimization the parameters were the gravity turn program, the propellant mass of the three stages and the thrust of the upper stage. The payload was constant and equal to 160kg. The results are shown in the table below.

Propellant mass $[kg]$			Dry mass $[kg]$			Thrust $[kN]$			Payload $[kg]$	Payload ratio $[\%]$
1	2	3	1	2	3	1	2	3		
25,763	11,337	2,186	4,149	1,493	885	1,123	517	20.1	160	0.348

Table 6.13: Solution 3, Optimization on the gravity turn, stage 1, stage 2, upper stage

From this optimization, the propellant mass in the first stage has been increased by 1.2%, and by 2.9% in the second stage, and the payload has increased by 8.1%.

6.3.3 The retained solution

From the three possible configurations exposed previously, the small launcher with the best performances is the one with the Zefiro boosters. The characteristics, namely specific impulse, thrust and propellant mass, of the retained solution are written in the table below.

	Z-23	Z-9	Upper stage
I_{sp} $[s]$	289	295	320.9
Thrust $[kN]$	1,196	225	3
m_p $[kg]$	23,906	10,115	500

Table 6.14: Three-stage small launcher solution

6.3.4 The upper stage design

The upper stage now needs to be designed in detail. This section presents the functional budget results and the structural mass calculation results.

The engine

From the simulations, it has been shown that the upper stage needs 500 kg of propellant mass and a thrust of 3kN. The 3kN thrust engine which is under study in the company is the chosen one. The characteristics of this engine are summarized in the table below.

Entity	Value
Oxidizer	NTO
Fuel	MMH
Mixture Ratio $[-]$	2.09
Mass flow $[kg/s]$	1
Nozzle diameter $[mm]$	460
Engine length $[mm]$	960
Nozzle length $[mm]$	544

Table 6.15: The 3kN thrust engine characteristics

Functional budget results

The functional budget has been done using the Excel tool presented in chapter 3.5. This section summarizes the results.

Propellant budget
The manual inputs for the propellant budget are summarized below. The Excel sheet illustration is shown in Figure 6.7.

- The number of boosts is equal to 2, as a re-ignition is done to circularize the orbit.
- The propellant mixture ratio is preset by the engine characteristics and so MR=2.09.

		NTO	MMH
Number of boosts	2		
Propellant mixture ratio (MR)	2,09		
Module propulsive mixture ratio (MRu)	2,09		
Nominally used propulsive propellant mass (main. prop. + att. control) [kg]	500	338	162
Performance reserve (input either as % or comp. with MRu)	2,5%	8	4
Transient phase engine consumption [kg] (conservative)		20	8
Lines priming: m_lines [kg]		5,0	4,0
m_1: Enhanced nom. propellant mass [kg]	**550**	372	178
Geometrical residuals [kg]		15	9
Project design margin (of nominally used prop. mass) [%]	5%	17	8
m_3: Loaded propellant mass [kg]	**599**	404	195
Loading process inaccuracy: I (of loaded mass) [%]	0,8%	3	2
		406,78	196,51
Total loaded propellant mass [kg]		**603**	

Figure 6.7: Propellant budget results for the upper stage of the 3-stage launcher

- The nominally used propellant is adopted from the simulations and so $m_{prop} = 500kg$.
- The value of a performance reserve is evaluated by an expert and it has been considered 2.5% of the nominally used propellant.
- The propellant mass for the geometrical residuals is calculated considering that 10 liters of each propellant is lost in geometrical residuals.
- The project design margin is set up at 5%.
- The value of 0.8% is used for the loading process inaccuracy and this value is estimated by experts.

With those inputs and the associated calculations, the total loaded propellant for NTO is ca. **407kg**, and for MMH of ca. **197kg**. So the total loaded propellant mass is **603kg**.

Tank volume

With the results from the propellant budget, the tank volume can be calculated as explained in the method in chapter 3.5. The manual inputs needed to do the calculations are summarized below. The Excel sheet illustration is shown in Figure 6.8.

	[%]		NTO	MMH
Total loaded propellant mass [kg]			407	197
Temperature propellant at lift-off [K] (for information only)			*300*	*300*
Density propellants [kg/m³]			1428	875
Total loaded propellant volume: V_load [m³]			**0,285**	**0,225**
	NTO	*MMH*		
Ullage volume needed: V_ull [input: % or m³]			0,050	0,050
			0,050	0,050
Internal equipments			0,015	0,015
Project margin M_proj [%]	5%		0,014	0,011
			0,364	**0,301**
Tank Volume [m³]			**0,7**	

Figure 6.8: Tank volume results for the upper stage of the 3-stage launcher

- Density for both propellants are required: $\rho_{NTO} = 1428 kg/m^3$ and $\rho_{MMH} = 875 kg/m^3$.
- A volume of 15L for both propellants has been considered for the internal equipments. This value is validated by experts.
- The ullage volume has been evaluated at 50L for both propellants.
- The project design margin is fixed to 5%.

With those inputs and the associated calculations, the total tank volume for NTO is ca. $\mathbf{0.4} m^2$, and for MMH ca. $\mathbf{0.3} m^2$. This results in a total tank of $\mathbf{0.7} m^2$.

Helium budget
With the results from the tank volume calculations, the required helium mass can be computed. The manual inputs needed to do the calculations are summarized below. The Excel sheet illustration is shown in Figure 6.9.

		NTO	MMH
Engine mass flow: m-dot_engine [kg/s]	1		
Propellant flowrate: m-dot [kg/s]		0,7	0,3
Density propellant: ρ [kg/m³]		1428	875
Prop. volume flowrate: v-dot [m³/s]		0,000	0,000
Vol. flowrate pressure gas: v-dot_He [m³/s]		0,000	0,000
Tank pressure: p [bar]		20	20,0
Ullage temperature press. gas: T_ull [K]		250	250
Specific gas const. pressure gas: R_He [J/(mol*K)]		2075,7	2075,8
Density pressure gas: ρ_He [kg/m³]		3,854	3,854
Mass-flowrate pressure gas: m-dot_He [kg/s]		0,002	0,001
Propellant to be fed out [kg]	550	372	178
Pressurization duration [s]		549,5	549,6
Pressure Gas need [kg]	2	1,0	0,8
Tank pressurization: 5bars - 20 bars [kg]		0,3	
Command system consumption [kg]		0,3	
Project margin: M_proj [kg]	10%	0,2	
Total need pressure gas: m_He,tot [kg]		**2,6**	

Figure 6.9: Pressure gas budget results for the upper stage of the 3-stage launcher

- The engine mass flow is an engine characteristic: $\dot{m}_{engine} = 1 kg/m^3$.
- The tank pressure in both tanks is equal to 20bars. This value is evaluated from todays upper stage.
- The ullage temperature pressure in both tanks is assumed for 250K.
- The mass needed to pressurize the tank from 5bars to 20 bars has been evaluated by experts and is equal to 0.3kg.
- The Helium consumption of the command system is set at 0.3kg, and this has been chosen according to an expert's statement.

With those inputs and the associated calculations, the Helium mass needed for pressurization is of **2.6kg**.

With the previous results, the maximal expected operating pressure is calculated. The Excel sheet illustration is shown in Figure 6.10

	NTO	MMH
Maximum He regulated pressure [bar]	25,00	25,00
Needed Helium mass m_He, tot [bar]	3	
He consumption variations (5% incl. leakage) [kg]	0,13	
Helium need for propellant expulsion only [bar]	245,36	
Helium budget non-consumable [bar]	55	
He consumption deviations [bar]	10,00	
He budget uncertainty & margin [bar]	10,00	
Total press. need for He storage. P_He,stor [bar]	320,36	
He pressure decrease due to temp. compensation [bar]	10,0	
He pressure rise due to temp. compensation [bar]	10,0	
MEOP He vessel [bar]	**340,4**	

Figure 6.10: MEOP Helium results for the upper stage of the 3-stage launcher

- The maximum regulated pressure in both tanks is 25bar. This pressure represents the maximum propellant tank pressure during the mission plus margins.
- From today's upper stage, a value of 10 bar is considered for the consumption deviations.
- 10 bar are considered for the uncertainty and margins.
- The Helium pressure decreases or rises after the last pressurization on ground due to temperature compensation requires a pressure amount of 10 bar.

The helium vessel volume has to be adjusted in order to get a MEOP below 410 bar. In this study, a volume of **70L** has been considered, that means that the MEOP is equal to ca. **340 bars**. From existing Helium vessels, the vessel has been scaled and the resulting mass is **15kg**.

Structural mass calculations results

The structural mass calculations tool has been modified for the purpose of this study. This chapter summarizes the results of the upper stage design. Three designs have been done corresponding to the three possible tank configurations, but at the end one design has been chosen. The reason of this choice will be explained later on.
The functional budget analysis has determined the required propellant masses and tank volumes for the 3-stage small launcher.

Common inputs for the three possible configurations
Indenpendantly of the tank architecture, inputs are required for the structural mass calculations. These inputs are summarized below.

- The maximal axial and lateral acceleration are assumed by an expert: $k_{ax} = 5g$ and $k_{lat} = 1g$.

- The NTO propellant is placed on the top in the common bulkhead configuration and in the sphere in the spherical configuration. The propellant mass, tank volume and ullage pressure are the ones used in the functional budget: $m_{NTO} = 448kg$, $V_{NTO} = 430L$ and $p_{ull,NTO,CB} = 20bar$ in the common bulkhead configuration and $p_{ull,NTO,Sph} = 20.5bar$ in the spherical configuration.
- The MMH propellant is placed on the bottom or in the cylinder. The propellant mass, tank volume and ullage pressure are the ones used in the functional budget: $m_{MMH} = 212kg$, $V_{MMH} = 354L$ and $p_{ull,MMH,CB} = 20,5bar$ in the common bulkhead configuration and $p_{ull,MMH,Sph} = 20bar$ in the spherical-cylinder configuration.
- The characteristics of the structural material have to be implemented. The Aluminum 2219 and the CFRP M40/6376 sandwich are used. For the Aluminum, the required inputs are: yield stress $\sigma_{Alu} = 358N/mm^2$, modulus of elasticity $E = 72000N/mm^2$ and density $\rho_{Alu} = 2851kg/m^3$. Concerning the composite materials, the following inputs are defined: modulus of elasticity $E_{CFRP} = 96386N/mm^2$, core density $\rho_{core} = 50kg/m^3$ and skin density $\rho_{skin} = 2296kg/m^3$.
- The payload mass is fixed to $250kg$.
- The fairing mass is calculated from the AVUM fairing. As the considered payload is six times smaller than the AVUM payload, the payload mass is considered 3 times smaller. As the AVUM fairing mass is equal to $560kg$, in this study it has been considered a fairing of $180kg$.

The load-bearing common bulkhead tank architecture
As the tanks are parts of the structure, the radius of the stage is the radius of the tank. A radius of 500mm has been considered for this configuration. With the tank volume and the tank radius it is possible to automatically calculate the geometry of the tank. The length of the cylinder and domes as well as the surfaces of the different parts of the tanks are summarized in Table 6.16

	Tank 1	Tank 2
Cylinder radius $[mm]$	500	500
Cylinder length $[mm]$	77	451
Dome length $[mm]$	354	354
Cylinder surface $[m^2]$	0.24	1.42
Dome surface $[m^2]$	0.95	0.95

Table 6.16: Tank geometrical characteristics, Common Bulkhead load bearing configuration

For the calculations of the axial fluxes, the diameter and length of the different sections are required. The diameter of all the section is the diameter of the tank (1000mm). The length of the section, which is the fairing length, is 3200mm. The section 2, which is the length of the intermediate structure, is 750mm long and the section 3, which the length of the interstage structure, is 1460mm long. These dimensions have been chosen regarding the place needed for all the components and tank length, furthermore the interstage length (section 3) has been chosen in order to respect an angle of minimum 10 degrees between the separation system and the exit nozzle part. The axial fluxes are shown in Figures 6.11.

The structural mass calculations results are shown in Figures 6.12 and 6.13. The calculation method is the one presented in chapter 3, section 5.

The total structural mass of the upper stage is **191.7kg**. This mass does not include the fairing, payload, avionics and subsystems masses.

load for loadb. tanks:

Sections	Length [mm]	Diam. [mm]	Axial Fluxes: Fax [N]	M [Nmm]	n_Fax	n_M	n_max	considered masses
length - fairing top to sec	3200	1700	21092	32083887	4	14	**18**	***P/L +fairing +wind***
length - sec 1 to sec 2	750	1000	54396	49621081	17	63	**80**	***+ prop+helium storage***
length - sec 3 to sec 4	1460	1000	55377	88765900	18	113	**131**	***+ engine***

load for not loadb. tanks:

length - sec 1 to sec 2	1587	1000	660	5139134	0	7	7	***prop***

Figure 6.11: Axial fluxes, common bulkhead load bearing tank, 3-stage launcher

Used Material	**Upper Cone structure**	thickness t [mm]	m [kg]	with Contingency
Cone 3936 scaled from AVUM	Upper Cone / Payload adapter		28	
	Avionic Platform		12	
	Total Mass		**40,0**	
	Tank 1	thickness t [mm]	m [kg]	
ALU 2219	Upper Dome	1,6	4,3	*1*
	due to strength (yield)	1,6		
ALU (scaled from ESC-A)	Upper Y-Ring		11,4	
	shell thickness: pressure	*3,2*		
	shell thickness: buckling	*0,6*		
	due to Buckling	*0,0*		
ALU 2219	Cylinder Structure with t_max	3,2	2,2	*1*
ALU 2219	Common Bulkhead	1,6	4,4	*1*
	due to strength (yield)	1,6		
ALU (scaled from ESC-A)	Bulkhead Y-Ring		11,4	
	Total Mass		**33,6**	
	Tank 2	thickness t [mm]	m [kg]	
	Tank 2 Cylinder			
	shell thickness: pressure	*3,3*		
	shell thickness: buckling	*0,6*		
ALU 2219	Cylinder Structure with t_max	3,26	13,1	*1*
ALU 2219	Lower Tank Dome	1,6	4,4	*1*
	due to strength (yield)	1,6		
ALU (scaled from ESC-A)	Lower Y-Ring		11,4	
	Tank 2 Sphere:			
	shell thickness: pressure	*0,00*		
	due to strength (yield)	*0,00*		
ALU 2219	Sphere Structure	0,00	0,0	*1,0*
	Total Mass		**29,0**	
	Helium Vessel	thickness t [mm]	m [kg]	
	Total Mass		**15,0**	

Figure 6.12: Structural mass part 1, common bulkhead load bearing tank, 3-stage launcher

The non load-bearing common bulkhead tank architecture

The space between the tank and the structure has been considered equal to 150mm. This assumption has been made regarding an existing upper stage. A diameter of 1200mm has been chosen, and so the tank radius was deducted to be equal to 450mm. The length of the cylinder and domes as well as the surfaces of the different parts of the tanks are summarized in Table 6.17

	Tank 1	Tank 2
Cylinder radius $[mm]$	450	450
Cylinder length $[mm]$	252	557
Dome length $[mm]$	318	318
Cylinder surface $[m^2]$	0.71	1.57
Dome surface $[m^2]$	0.77	0.77

Table 6.17: Tank geometrical characteristics, common Bulkhead non-load bearing configuration

For the calculations of the axial fluxes, the diameter and length of the different section are required. The diameter of the sections is 1200mm. The length of the section 1, which is the fairing

	Intermediate Structure (1 to 2)	thickness t [mm]	m [kg]	
Sandwich	*2 x t_face sheet: strength (scaled)*	0,2		
	2 x t_face sheet: buckling (approximation)	0,2		
	thickness core (estimate)	10		
	Structure sec 1 to 2	1,3	9,9	*1,2*
	Upper ring + Lower ring		2,9	
	Bolts + screws		4,0	
	Total Mass		**16,8**	
	Engine Thrust Frame	**thickness t [mm]**	**m [kg]**	
ALU (scaled from ESC-B)	ETF Structure		22,14	*1,5*
	Total Mass		**22,1**	
	CTAR	**thickness t [mm]**	**m [kg]**	
scaled	CTAR		0,00	
	Total Mass		**0,0**	
	Interstage Skirt (2 to 3)	**thickness t [mm]**	**m [kg]**	
Sandwich	*2 x t_face sheet: strength (scaled)*	0,3		
	2 x t_face sheet: buckling (approximation)	0,2		
	thickness core (estimate)	10,0		
	ISS Structure	1,3	27,7	*1,2*
ALU (scaled from AVUM)	ISS upper ring+lower ring		2,9	
ALU (scaled from AVUM)	Rivet		3,1	
	Bolts, Screws		4,0	
	Total Mass		**37,7**	
	Total Structural Mass [kg]		**194,3**	
	Total Tank Mass [kg]		**62,6**	
	Total Structural Mass without tanks [kg]		**131,7**	

Figure 6.13: Structural mass part 2, common bulkhead load bearing tank, 3-stage launcher

length, is 3200mm. Section 2, which is the length of the intermediate structure, is 1400mm long and section 3, which the length of the interstage structure, is 1760mm long. These dimensions have been chosen regarding the space needed for all the components and tank length, furthermore the interstage length (section 3) has been chosen in order to respect an angle of minimum 10 degrees between the separation system and the exit nozzle part. The axial fluxes are shown in Figure 6.14 The structural mass calculation results are shown in Figures 6.15 and 6.16. The

load for loadb. tanks:

Sections	Length [mm]	Diam. [mm]	Axial Fluxes: Fax [N]	M [Nmm]	n_Fax	n_M	n_max	*considered masses*
length - fairing top to sec	3200	1700	21092	32083887	4	14	18	***P/L +fairing +wind***
length - sec 1 to sec 2	1400	1200	54396	64819982	14	57	72	***+ prop+helium storage***
length - sec 3 to sec 4	2200	1200	55377	123805325	15	109	124	***+ engine***
load for not loadb. tanks:								
length - sec 1 to sec 2	1763	900	660	5706389	0	9	9	***prop***

Figure 6.14: Axial fluxes, common bulkhead non load bearing tank, 3-stage launcher

calculation method is the one presented in chapter 3, section 5. The total structural mass of the upper stage is **224kg**. This mass does not include the fairing, payload, avionics and subsystems masses.

The spherical-cylindrical tank architecture

From the propellant volumes, the dimensions of the tanks are calculated. The geometries of the cylinder and sphere as well as the surfaces of the different parts of the tanks are summarized in Table 6.18

For the calculation of the axial fluxes, the diameter and length of the different sections are required. The diameter of the sections is 1350mm.This diameter has been chosen in order to have a space of 100mm between the tanks and the structure. The fairing length is 3200mm. Section 2, which is the length of the intermediate structure, is 600mm long and section 3, which the length of the interstage structure, is 2200mm long. These dimensions have been chosen regarding the

Used Material	Upper Cone structure	thickness t [mm]	m [kg]	with Contingency
Cone 3936 scaled from AVUM	Upper Cone / Payload adapter		33	
	Avionic Platform		14	
	Total Mass		**47,0**	
	Tank 1	thickness t [mm]	m [kg]	
ALU 2219	Upper Dome	1,4	3,1	*1*
	due to strength (yield)	1,4		
ALU (scaled from ESC-A)	Upper Y-Ring		8,3	
	shell thickness: pressure	2,9		
	shell thickness: buckling	*0,6*		
	due to Buckling	0,0		
ALU 2219	Cylinder Structure with t_max	2,9	5,9	*1*
ALU 2219	Common Bulkhead	1,5	3,2	*1*
	due to strength (yield)	1,5		
ALU (scaled from ESC-A)	Bulkhead Y-Ring		8,3	
	Total Mass		**28,8**	
	Tank 2	thickness t [mm]	m [kg]	
	Tank 2 Cylinder			
	shell thickness: pressure	2,9		
	shell thickness: buckling	*0,6*		
ALU 2219	Cylinder Structure with t_max	2,94	13,2	*1*
ALU 2219	Lower Tank Dome	1,5	3,2	*1*
	due to strength (yield)	1,5		
ALU (scaled from ESC-A)	Lower Y-Ring		8,3	
	Tank 2 Sphere:			
	shell thickness: pressure	*0,00*		
	due to strength (yield)	*0,00*		
ALU 2219	Sphere Structure	0,00	0,0	*1,0*
	Total Mass		**24,7**	
	Helium Vessel	thickness t [mm]	m [kg]	
	Total Mass		**15,0**	

Figure 6.15: Structural mass part 1, common bulkhead non load bearing tank, 3-stage launcher

	Intermediate Structure (1 to 2)	thickness t [mm]	m [kg]	
	2 x t_face sheet: strength (scaled)	0,2		
	2 x t_face sheet: buckling (approximation)	0,2		
Sandwich	thickness core (estimate)	10		
	Structure sec 1 to 2	1,3	22,1	*1,2*
	Upper ring + Lower ring		3,5	
	Bolts + screws		4,0	
	Total Mass		**29,6**	
	Engine Thrust Frame	thickness t [mm]	m [kg]	
ALU (scaled from ESC-B)	ETF Structure		19,16	*1,5*
	Total Mass		**19,2**	
	CTAR	thickness t [mm]	m [kg]	
scaled	CTAR		11,32	
	Total Mass		**11,3**	
	Interstage Skirt (2 to 3)	thickness t [mm]	m [kg]	
Sandwich	*2 x t_face sheet: strength (scaled)*	*0,3*		
	2 x t_face sheet: buckling (approximation)	*0,2*		
	thickness core (estimate)	10,0		
	ISS Structure	1,3	44,3	*1,2*
ALU (scaled from AVUM)	ISS upper ring+lower ring		3,5	
ALU (scaled from AVUM)	Rivet		3,7	
	Bolts, Screws		4,0	
	Total Mass		**55,5**	
	Total Structural Mass [kg]		**231,0**	
	Total Tank Mass [kg]		**53,5**	
	Total Structural Mass without tanks [kg]		**177,5**	

Figure 6.16: Structural mass part 2, common bulkhead non load bearing tank, 3-stage launcher

	Tank 1(cylinder)	Tank 2 (sphere)
Cylinder radius $[mm]$	367	
Cylinder length $[mm]$	878	
Cap length $[mm]$	133	
Cylinder surface $[m^2]$	2.02	
Cap surface $[m^2]$	0.48	
Sphere radius $[mm]$		572
Sphere surface $[m^2]$		3.15

Table 6.18: Tank geometrical characteristics, spherical-cylindrical configuration

space needed for all the components and tank length, furthermore the interstage length (section 3) has been chosen in order to respect an angle of minimum 10 degrees between the separation system and the exit nozzle part. The axial fluxes are shown in Figure 6.17.

load for loadb. tanks:

Sections	Length [mm]	Diam. [mm]	Axial Fluxes: Fax [N]	M [Nmm]	n_Fax	n_M	n_max	considered masses
length - fairing top to sec	3200	1700	21092	32083887	4	14	**18**	*P/L +fairing +wind*
length - sec 1 to sec 2	600	1350	54396	46113642	13	32	**45**	*+ prop+helium storage*
length - sec 3 to sec 4	2500	1350	55377	113142441	13	79	**92**	*+ engine*

load for not loadb. tanks:

Sections	Length [mm]	Diam. [mm]	Fax [N]	M [Nmm]	n_Fax	n_M	n_max	considered masses
length - sec 1 to sec 2	1144	1144	660	3703583	0	4	**4**	*prop*

Figure 6.17: Axial fluxes, spherical-cylindrical, 3-stage launcher

The structural mass calculations results are shown in Figures 6.18 and 6.19. The calculation method is the one presented in chapter 3, section 5.
The total structural mass of the upper stage is **232.3 kg**. This mass does not include the fairing, payload, avionics and subsystems masses.

The solution

The tank configuration, which gives the lightest structure, is the common bulkhead load-bearing tank configuration. However, considering advice from mechanical engineers, this configuration does not allow direct access between the upper part and the lower part of the stage. This might be an issue if there is any problem during the production process. Therefore there is a need to have direct access. Furthermore, all the lines between the upper and lower part would be outside the structure. Therefore, it has been decided to consider the second lightest version which is the common bulkhead non load-bearing version. This upper stage has been studied in more detail. A mass break-down has been done and simulations with the complete launcher have been done again in order to see the performances.

6.3.5 3-stage launcher configuration

With the results of the simulations and the systems engineering software tools, the configuration of the three-stage small launcher is defined. In this part, the architecture of the launcher and more precisely the architecture of the upper stage thanks to CAD drawings, is described. The total dry mass of the upper stage has been calculated as well. The upper-stage CAD drawings are specifically generated for this study by the design department of the Astrium GmbH in Bremen.

Used Material	Upper Cone structure	thickness t [mm]	m [kg]	with Contingency
Cone 3936	Upper Cone / Payload adapter		36	
scaled from AVUM	Avionic Platform		16	
	Total Mass		**52,0**	
	Tank 1	thickness t [mm]	m [kg]	
ALU 2219	Upper Dome	1,3	0,8	*1*
	due to strength (yield)	1,0		
ALU (scaled from ESC-A)	Upper Y-Ring		0,0	
	shell thickness: pressure	*2,1*		
	shell thickness: buckling	*0,6*		
	due to Buckling	*8,5*		
ALU 2219	Cylinder Structure with t_max	8,5	46,9	*1*
ALU 2219	Common Bulkhead	1,3	0,8	*1*
	due to strength (yield)	1,1		
ALU (scaled from ESC-A)	Bulkhead Y-Ring		0,0	
	Total Mass		**48,4**	
	Tank 2	thickness t [mm]	m [kg]	
	Tank 2 Cylinder			
	shell thickness: pressure	*0,0*		
	shell thickness: buckling	*0,6*		
ALU 2219	Cylinder Structure with t_max	0,00	0,0	*1*
ALU 2219	Lower Tank Dome	1,3	0,0	*1*
	due to strength (yield)	0,0		
ALU (scaled from ESC-A)	Lower Y-Ring		0,0	
	Tank 2 Sphere:			
	shell thickness: pressure	*1,89*		
	due to strength (yield)	*1,89*		
ALU 2219	Sphere Structure	1,89	19,8	*1,0*
	Total Mass		**19,8**	
	Helium Vessel	thickness t [mm]	m [kg]	
	Total Mass		**15,0**	

Figure 6.18: Structural mass part 1, spherical-cylindrical, 3-stage launcher

	Intermediate Structure (1 to 2)	thickness t [mm]	m [kg]	
	2 x t_face sheet: strength (scaled)	0,1		
	2 x t_face sheet: buckling (approximation)	0,1		
Sandwich	thickness core (estimate)	10		
	Structure sec 1 to 2	1,3	10,6	*1,2*
	Upper ring + Lower ring		4,0	
	Bolts + screws		4,0	
	Total Mass		**18,6**	
	Engine Thrust Frame	thickness t [mm]	m [kg]	
ALU (scaled from ESC-B)	ETF Structure		13,03	*1,5*
	Total Mass		**13,0**	
	CTAR	thickness t [mm]	m [kg]	
scaled	CTAR		19,08	
	Total Mass		**19,1**	
	Interstage Skirt (2 to 3)	thickness t [mm]	m [kg]	
Sandwich	*2 x t_face sheet: strength (scaled)*	*0,2*		
	2 x t_face sheet: buckling (approximation)	*0,2*		
	thickness core (estimate)	10,0		
	ISS Structure	1,3	52,7	*1,2*
ALU (scaled from AVUM)	ISS upper ring+lower ring		4,0	
ALU (scaled from AVUM)	Rivet		4,1	
	Bolts, Screws		4,0	
	Total Mass		**64,8**	
	Total Structural Mass [kg]		**250,8**	
	Total Tank Mass [kg]		**68,3**	
	Total Structural Mass without tanks [kg]		**182,5**	

Figure 6.19: Structural mass part 2, spherical-cylindrical, 3-stage launcher

The architecture

The three-stage small launcher is composed of the Zefiro 23 stage, the Zefiro 9 stage and an upper stage specially designed for this study. A complete drawing of the launcher is represented in Figure 6.20. No changes are done on the Zefiro stages, the complete definition of the upper stage is done in the following part.

For the upper stage with the AESTUS 3kN engine, the Common Bulkhead configuration as a non-load bearing propellant tank is chosen regarding the structural mass and mostly, regarding the fact this configuration allows an access between the lower and upper parts of the launcher. The stage diameter is 1200mm and the cylindrical tank diameter is equal to 900mm. This facilitates a space of 150mm between the tank shells and the stage structure. The interstage between the second stage and the upper stage has a conical shape; the big diameter is equal to 1900mm, which is the Zefiro stage diameter and the small diameter, is equal to 1200mm. The fairing is placed on the top of the upper-stage. The fairing length is equal to 3200mm and the maximal diameter of the fairing is equal to 1700mm. The fairing geometry is chosen regarding the existing fairing of the VEGA launcher.

Figure 6.21and 6.22 illustrates the upper stage concepts with all components, explained below. The technical drawing of the upper stage can be found in Annex E. The upper cone is designed to be the upper interface to the payload and its design is based on an existing payload adapter. The big diameter of the cone is equal to the stage diameter and the small diameter is equal to 800mm. Additionally, an avionic platform is implemented on the top of the stage. This platform enables the attachments of all the avionics equipments and a vessel is placed in the middle of the platform. The avionics platform is a circular band of the size 200mm. The attitude control system is the one used on the VEGA launcher: RACS (Roll and Attitude Control System). This system requires a hydrazine vessel with 38 kg of Hydrazine. This vessel has a diameter of 482mm and is placed in the middle of the avionics platform. The two thruster cluster modules of the RACS system are dimetrically opposite placed at the top of the stage.
At the stage's interstage structure, the common bulkhead NTO/MMH tank is attached thanks to the CTAR. The engine thrust frame is directly attached to the lower cylinder and the AESTUS 3kN engine is attached to the engine thrust frame. The pressure control system (PCA) is attached to the engine thrust frame as well as the helium vessel. 70 liters of helium is required as explained in the helium budget calculations. The helium vessel configuration is cylindrical with two spherical domes. One helium vessel with a diameter of 430mm is needed. The interstage structure represents the connection with the previous stage Zefiro 9. The total length of the upper stage including the cone, is 3410mm and the total length of the launcher including the fairing is 17,901mm.

In Appendix E, the fluid schematics of the upper stage is illustrated. The pressurization system, the RACS and all the different feed lines are represented as well as the propellant fill and drain valves required for the fueling on ground.

The mass breakdown

After the definition of the upper stage architecture, the total mass has to be evaluated. The Structural Calculations Software Tool has enabled the evaluation of the tank mass and the main structural component masses, but there is still the need to evaluate the mass of the remaining subsystems. That is why a mass breakdown has been established for the upper stage. The different subsystem masses have been chosen regarding existing studies and validated by an expert. The following items have been considered in the mass breakdown [12]:

- propulsion system

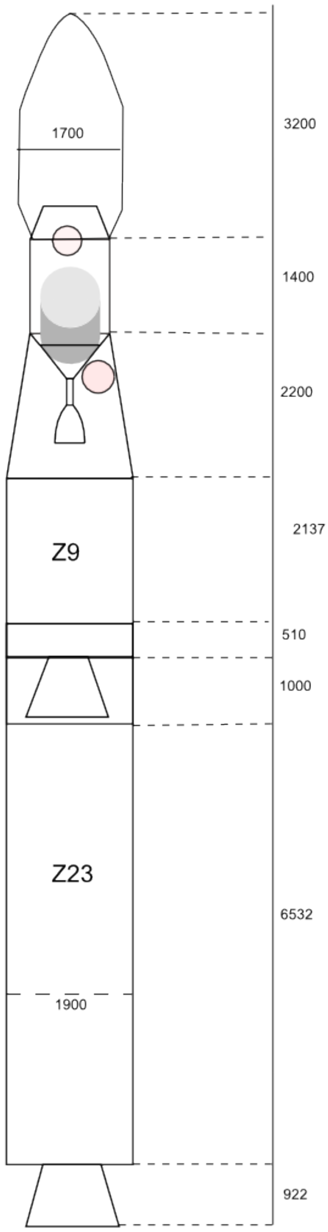


Figure 6.20: The three-stage launcher architecture

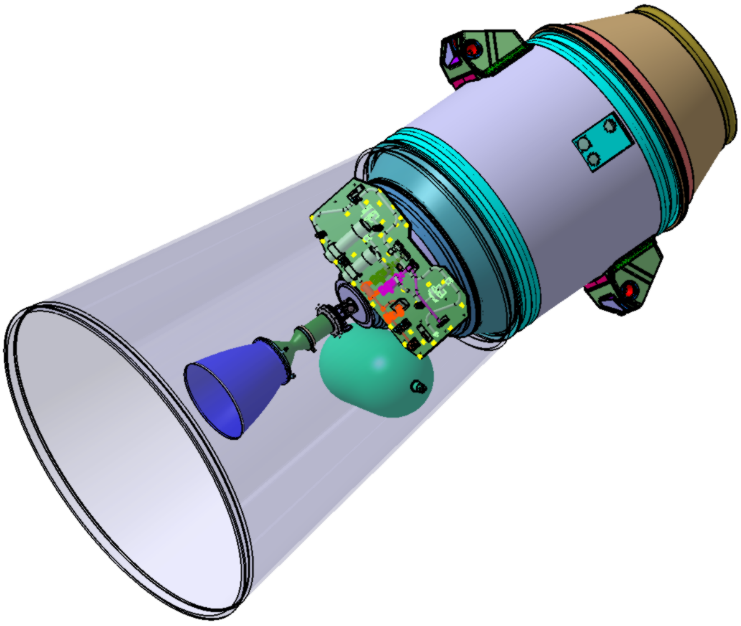


Figure 6.21: 3D drawing of the upper stage, 3-stage launcher

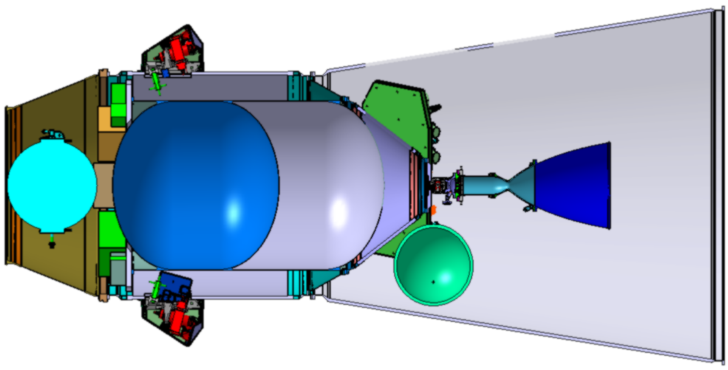


Figure 6.22: Section cut of the upper stage, 3-stage launcher

- equipped tanks (propellant tank, helium vessel, hydrazine vessel)
- fluid masses (helium, hydrazine, reserve propellant)
- structures (stage structure, interstage skirt, equipments)
- avionics
- RACS [20]
- separation system
- subsystems (thermal subsystems, neutralization system, feed lines supports, electrical system)

The mass breakdown table is represented on the following table. Only the main items presented before are included in this table. They are the relevant components for an upper-stage, the subsystems are defined but not explained in detail.

- The mass of the 3kN thrust AESTUS engine is 20kg.
- The calculated mass of the common bulkhead tank thanks to the Structural Calculation Mass Tool is 53.5kg.
- The 70liters of helium are stored in one vessel of 15kg. This mass is evaluated by scaling the existing vessel.
- The hydrazine vessel is the one used for the VEGA launcher, its mass is 8.45kg.
- The helium mass evaluated by the Helium budget is 4kg.
- The Hydrazine mass is the one considered for the VEGA launcher and it is 38kg.
- Regarding the propellant budget, 163kg are considered as reserve propellant.
- The structural mass is calculated with the Structural Calculations Tool. The stage structure is equal to 29.6kg and the interstage skirt is equal to 46.4kg. The upper cone, the avionics platform, the CTAR and the ETF masses have been evaluated by scaling from existing systems.
- The avionics mass considered is that of the VEGA launcher, 96kg.
- The RACS system is that of the VEGA launcher. The total mass is 41.75kg.
- The separation system is evaluated by scaling from existing systems, 14.2kg.
- The subsystem mass is scaled regarding the VEGA launcher, 57 kg have been considered for the total subsystem mass.

Finally, by summing up all these components, the total upper stage dry mass is equal to **713.7 kg**, with 60.6 kg of jettisonable mass.

NAME OF PRODUCT	JETTISONABLE MASS [kg]	MASS [kg]
Equipped tank		**76.95**
Equipped propellant tanks		53.5
Helium vessel		15
Hydrazine vessel		8.45
Fluid mass		**705**
Helium		4
Hydrazine		38
Reserve propellant		163
Nominal propellant		500
Structure	**46.4**	**109.1**
Stage structure		29.6
Interstage skirt	46.4	
Additional structural mass		79.5
Upper cone structure		*33*
Avionics platform		*16*
CTAR		*11.3*
Engine thrust frame		*19.2*
Avionics		**96**
RACS		**41.75**
Pressure transducers		0.5
Pyrotechnic valves		1
Feed line and drain valve		0.65
Turbing kit		2
Pipe support brackets + bolts		2.7
Thruster cluster module		27
Aerothermal cover		7.9
Separation system	**14.2**	
Pyrojacks	7.5	
Pyro lines	5.4	
RMV	1.3	
Propulsion system		**67.1**
Engine		20
Pressure control assembly		12
Feed lines		7
Pressurization lines		6.5
Passivation system		3.4
Actuators		15
Fill and drain coupling		3.2
Subsystems		**57.2**
Thermal subsystems		18.4
Neutralisation system		7.5
Supports		20.7
Electrical system		11.6
TOTAL STAGE DRY MASS	**60.6**	**653.1**
FAIRING	**180**	
TOTAL UPPER STAGE MASS	**240.6**	**1153.1**

Table 6.19: Mass breakdown of the upper stage for the 3-stage launcher

	Stage 1: Z23	Stage 2: Z9	Stage 3
Isp [s]	289	295	321
Thrust [kN]	1196	225	3
Propellant mass [kg]	23,906	10,115	500
Dry mass [kg]	1,845	833	653.1
Interstage mass [kg]			60.6
Length [m]	7454	3547	3600
Diameter [m]	1.9	1.9	1.2

Table 6.20: The three-stage launcher characteristics

6.3.6 Small launcher simulation

Simulations have been run with the designed upper stage. The characteristics of the Zefiro 23 and Zefiro 9 have been implemented on the Matlab code. The propellant mass of the third stage has been considered equal to 500kg which is the nominal propellant mass. Regarding the mass breakdown, the structural mass has been considered equal to 653.1kg and the interstage mass is equal to 60.6kg. The fairing is equal to 180 kg.

With this launcher configuration, the Matlab simulations show that it is possible to place 330kg into the elliptical transfer orbit 200km perigee - 700km apogee. As there is a need of around 35kg of propellant in order to realize the circularization, the payload which can be placed into the circular orbit with a radius of 700km, is around **295kg**.

The simulation is represented in the following figures. Figure 6.23 shows the variation of altitude, velocity and flight path angle as functions of time and the altitude as a function of velocity, from the launch pad to the elliptical orbit. Each color corresponds to one stage:

- the curve red is the first stage
- the blue curve is the second stage
- the green curve is the upper stage

Figure 6.24 represents the variations of the altitude, velocity and flight path angle as functions of time and the altitude as a function of velocity, from the launch pad to the circular orbit. Each color corresponds to a certain part of the trajectory:

- the blue curve is from the launch pad to the elliptical orbit at burnout of the upper stage
- the red curve corresponds to the trajectory in orbit when the thrust is equal to zero
- the green curve corresponds to the trajectory in the circular orbit

The upper stage re-ignition happens when the spacecraft reaches the orbit apogee. Figure 6.25 represents the variations of the mass, thrust, dynamic pressure, lateral and longitudinal velocity as functions of time. The lateral velocity is the red curve and the longitudinal velocity is the blue one.

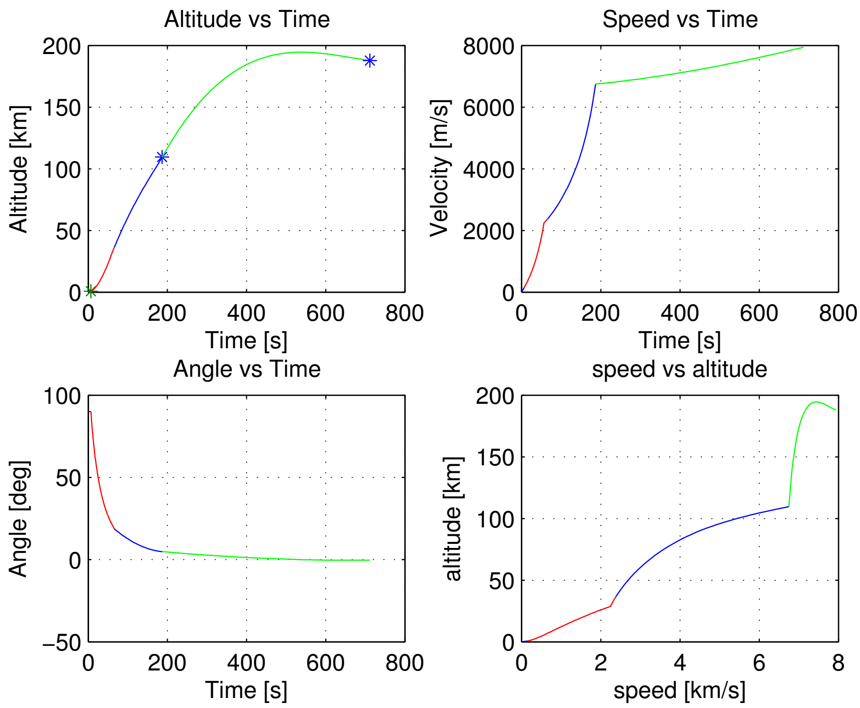


Figure 6.23: 3-stage launcher simulation from launch pad to the elliptical orbit

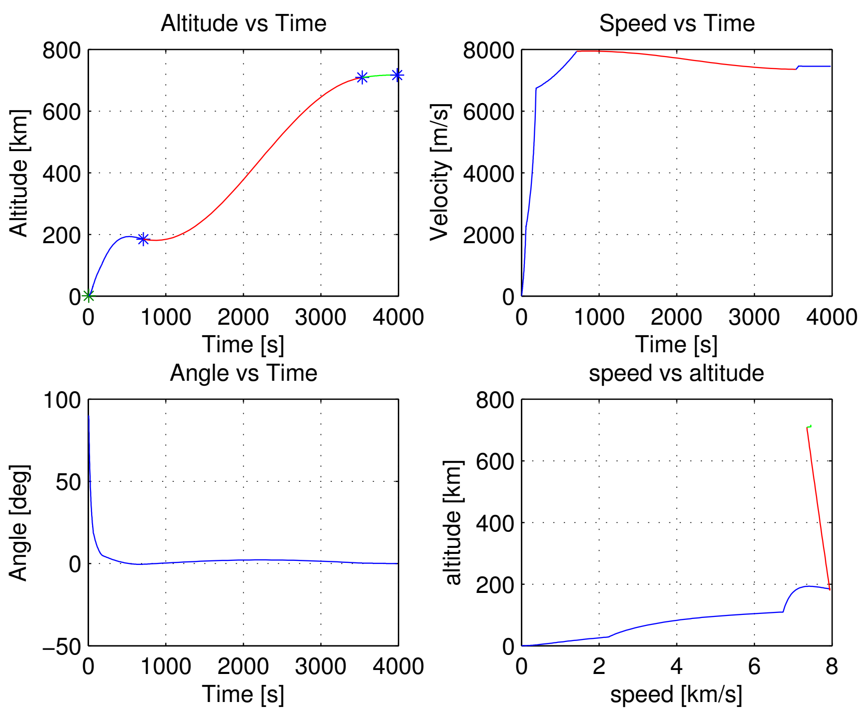


Figure 6.24: 3-stage launcher simulation from launch pad to the circular orbit

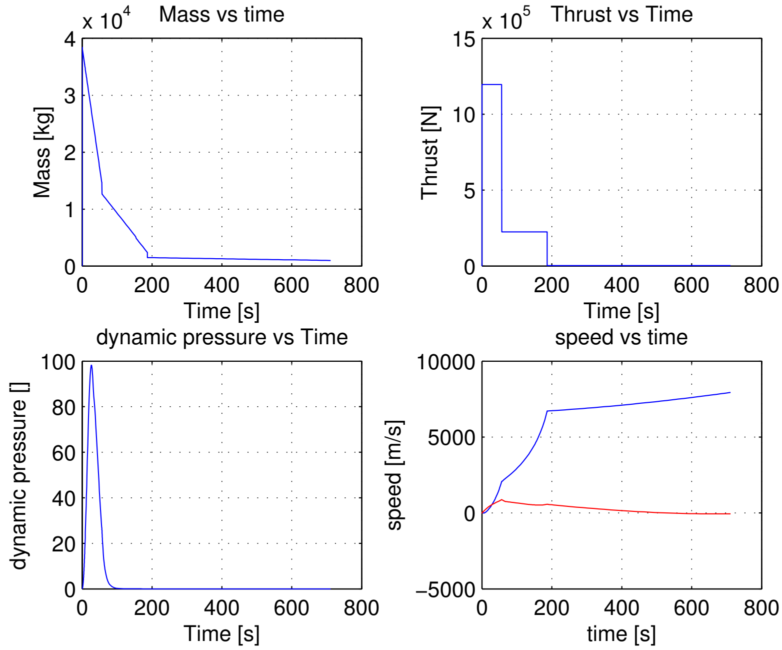


Figure 6.25: 3-stage launcher simulation, other characteristics

6.4 2-stage Launcher Solution

The purpose is to propose a solution with two liquid stages. The first stage with liquid propulsion is from an existing stage, the upper stage is designed in the course of the study with an engine developed in the company. Two existing liquid stages have been studied: the first stage of the Soyuz launcher, and the second stage of the Zenit launcher. Another engine has been studied as well. It is the engine derived form the Zenit launcher which is under investigation as explained previously. The results with these two boosters are presented in the following section.

6.4.1 Initial values, constants and inputs

As explained previously in the method part, existing liquid stages with enough thrust to lift off are too big for the purpose of the small launcher. That is why it has been decided to reduce their masses by decreasing the amount of propellant and so decreasing as well the structural mass. Concerning the first stage, the thrust and the specific impulse are considered constant and the structural index varies as a function of propellant mass. The curve, which represents this variation, is the optimum curve presented in the existing stages section. The structural mass of the upper stage is calculated from the propellant mass and the thrust of the upper stage. The specific impulse is constant and equal to 320.9.

The initial velocity is set to zero as the required orbit has an inclination of 90 degrees. The drag coefficient is constant and equal to 0.3. The reference area is equal to $6.6052m^2$ for the SOYUZ first stage, which corresponds to a diameter of 2.9 meters, it is equal to $11.9459m^2$ for the ZENIT first stage, which corresponds to a diameter of 3.9 meters, and it is equal to $7.0686m^2$ for the new Zenit engine, which corresponds to a diameter of 3 meters. The last diameter is an assumption which seems reasonable.

The payload is constant and equal to 250 kg.

The inputs are

- the beginning and the end of the gravity turn program: t_1 and t_2

- the propellant mass in the first stage: $m_{p,1}$
- the propellant mass in the third stage: $m_{p,2}$
- the thrust in the third stage: F_2

The optimization program has 5 parameters: two parameters from the gravity turn program, two parameters from the propellant mass in the three stages and one parameter for the thrust in the upper stage.

$$v_1 = \begin{pmatrix} t_1 & t_2 & m_{p,1} & m_{p,2} & F_2 \end{pmatrix} \tag{6.5}$$

6.4.2 Optimal launcher

The characteristics of the three studied stages are summarized in Table 6.21.

Entity	Soyuz, stage 1	Zenit, stage 2	Zenit, booster
Isp [s]	319	349	330
Thrust [kN]	990	912	849
m_p [kg]	93,311	80,600	
m_{dry} [kg]	6454	8900	
m_{engine} [kg]	1,400	1,500	1080
D [m]	2.9	3.9	3

Table 6.21: Chosen liquid stages characteristics

Solution 1: Soyuz stage + Upper stage

The configuration which has been studied is to have the first stage of the Soyuz launcher in the first stage of the small launcher and a liquid upper stage. As explained in the method, the purpose was, by keeping the thrust and the specific impulse constant, to optimize the propellant mass in the first stage and the the propellant mass and the thrust in the upper stage. In this optimization the payload is fixed and equal to 250kg. The optimization is done on the gravity turn program, the amount of propellant in the first and second stage and the thrust in the upper stage. The thrust and the specific impulse in the first stage are fixed, the structural mass was calculated thanks to the optimum curve of the structural index and the curve representing the engine mass as function of the thrust. The specific impulse in the upper stage is fixed, the structural mass was calculated from the functions depending on the thrust and the propellant mass. Several possibilities are studied depending on the amount of propellant in the first stage. The criteria for the best solution is the highest payload ratio.

From several simulations, it has been noticed that this configuration can not put 250kg of payload in the desired orbit. Therefore, other simulations have been run decreasing the amount of payload. A solution has been found with 150kg of payload. The launcher characteristics with this solution are written in the table below.

Propellant mass [kg]		Dry mass [kg]		Thrust [kN]		Payload [kg]	Payload ratio [%]
1	2	1	2	1	2		
58,887	2,383	5,280	1,209	990	27	150	0.221

Table 6.22: Liquid, Solution 1, Soyuz stage and liquid upper stage

The simulations with the Soyuz launcher do not give good enough results.

Solution 2: Zenit stage + Upper stage

Another configuration has been studied with the second stage of the Zenit launcher. The purpose is to optimize the amount of propellant in the first stage and the upper stage and the thrust in the upper stage with a payload mass fixed and equal to 250kg. The optimization is done on the gravity turn program the amount of propellant in the first and second stage and the thrust in the upper stage. The thrust and the specific impulse in the first stage are fixed, the structural mass is calculated thanks to the optimum curve of the structural index. The specific impulse in the upper stage is fixed, the structural mass is calculated from functions depending on the thrust and the propellant mass. As before, several possibilities are studied depending on the amount of propellant in the first stage, the selection criteria is to get the highest payload ratio. The results are shown in the following table.

Propellant mass [kg]		Dry mass [kg]		Thrust [kN]		Payload [kg]	Payload ratio [%]
1	2	1	2	1	2		
43,023	1,284	5,064	770	912	12.3	250	0.496
41,536	1,507	4,955	840	912	15	250	0.509
40,620	1,850	4,943	944	912	18.5	250	0.514
39,745	2,329	4,930	1084	912	23.3	250	0.517

Table 6.23: Liquid, Solution 2, Zenit stage and liquid upper stage

When decreasing the propellant mass in the first stage, the upper stage needs to have better performance and therefore to increase its thrust and propellant mass. The purpose was to see at what point the payload ratio started decreasing and so to have the best solution. As can be seen in Table 6.23, in the last row the payload ratio is the highest which was possible to get, which means that the best possible small launcher configuration is to have a first stage with around40,000kg of propellant, and an upper stage with around 23kN thrust and 2,300kg of propellant mass.

However, in this configuration the total thrust takes into account the steering engines of the Zenit launcher. But, the thrust from these engines is non-propulsive as it is not longitudinal thrust but it is used for attitude control. Therefore, another study has been done by using the engine under investigation derived from the Zenit main engine.

Solution 3: Zenit booster + Upper stage

The same simulations as explained previously, have been done by using the new Zenit engine. This engine is derived from the main engine of the Zenit second stage. The purpose is to developed an engine usable in a first stage. In the simulations, the thrust and specific impulse of the first stage were constant whereas the structural mass was calculated thanks to the thrust and the amount of propellant. Concerning the upper stage, the specific impulse was constant, the structural mass was calculated thanks to the thrust and the propellant mass. The variables are the amount of propellant in the 2 stages, the thrust in the upper stage, and the times of the gravity turn program. A solution has been found with 150kg of payload. The results are shown in the table below.

Propellant mass [kg]		Dry mass [kg]		Thrust [kN]		Payload [kg]	Payload ratio [%]
1	2	1	2	1	2		
47,000	1,989	5,268	991	849	18.4	150	0.271

Table 6.24: Liquid, Solution 3, Zenit booster and liquid upper stage

6.4.3 The retained solution

The best solution among the possible configurations exposed previously is chosen based on the payload ratio. So the configuration, which gives the highest payload ratio, is the one which uses the Soyuz stage as a first stage. However, as explained previously, in the simulations, the thrust of the Soyuz stage takes into account the thrust from the steering engines, so the considered thrust is too large compared to reality. It has been decided to choose the new Zenit engine which is still under investigation even if from the simulations this configuration could place into orbit only 150kg.
Concerning the upper stage, the engine under investigation in the company, which is the closest one from the simulation results, is a 22kN thrust engine. The exit nozzle of this engine is not optimized, and so the specific impulse is a bit smaller than the one considered in the simulations, and it is equal to 316s. The characteristics of this small launcher are summarized in Table 6.25.

	Zenit booster	Upper stage
I_{sp} $[s]$	330	316
Thrust $[kN]$	849	22
m_p $[kg]$	47,000	2,000

Table 6.25: The 2-stage launcher solution

The thrust and the amount of propellant of the upper stage are fixed, the structure of the upper stage can be designed in more detail for this solution. The structure of the first stage using the Zenit booster will be designed as well.

6.4.4 The Zenit new design

The first stage of the 2-stage small launcher uses the Zenit engine which is under investigation. This stage needs to be entirely designed. This section explains the new design of the Zenit stage with the functional budget and the structural mass calculations.

The engine

The engine of the Zenit stage is composed of the booster engine which is derived from the main engine of the Zenit second stage. The characteristics of this engine, namely mixture ratio, engine mass, length, diameter, thrust and specific impulse, are written in the table below.

	Zenit booster RD-120K
Thrust $[kN]$	849
Isp $[s]$	330
Engine mass $[kg]$	1080
MR $[-]$	2.6
Mass flow $[kg/s]$	20.8
Diameter $[mm]$	1400
Length $[mm]$	2400

Table 6.26: The Zenit booster characteristics

The functional design has been done using the Excel tool presented in chapter 3, section 4. This section summarizes the results.

Propellant budget

The manual inputs for the propellant budget are summarized below. The Excel sheet illustration is shown in Figure 6.26.

		LOX	Kerosene
Number of boosts	1		
Propellant mixture ratio (MR)	2,6		
Module propulsive mixture ratio (MRu)	2,56		
Nominally used propulsive propellant mass (main. prop. + att. control) [kg]	47000	33944	13056
Performance reserve (input either as % or comp. with MRu)	2,0%	679	261
Transient phase engine consumption [kg] (conservative)		70	12
Lines priming [kg]		0,0	0,0
∑ prop. mass + perf. reserve + trans. phase (kg)	**48022**	34693	13329
Geometrical residuals [kg]		0	0
Project design margin (of nominally used prop. mass)	1%	339	131
Vaporized mass (in tanks) [kg] (NTO/MMH: input=0)		0	0
Loaded Mass [kg]	**48492**	35033	13459
Loading process inaccuracy (of loaded mass)	0,7%	245,23	94,21
		35278	13553
Total loaded propellant mass [kg]		**48831**	

Figure 6.26: Propellant budget results for the first stage of the 2-stage launcher

- The number of boosts is equal to 1
- The propellant mixture ratio is preset by the engine characteristics and so MR=2.6
- The nominally used propellant is adapted from the simulations and so $m_{prop} = 47000kg$
- The value of a performance reserve is evaluated by an expert and it has been considered to be 2% of the nominally used propellant
- The propellant mass for the geometrical residuals is calculated considering that 10 liters of each propellant is lost in geometrical residuals.
- The project design margin is set up to 1%
- The value of 0.7% is used for the loading process inaccuracy and this value is estimated by experts

With those inputs and the associated calculations, the total loaded propellant for LOX is ca. **35278kg**, and for Kerosene ca. **13553kg**. So the total loaded propellant is **48831kg**.

Tank volume

With the results from the propellant budget, the tank volume can be calculated as explained in chapter 3.5. The manual inputs needed to do the calculations are summarized below. The Excel sheet illustration is shown in Figure 6.27.

- Both propellant densities are required: $\rho_{LOX} = 1142kg/m^3$ and $\rho_{Kerosene} = 803kg/m^3$.
- A volume of 15L for both propellants has been considered for the internal equipment. This value is validated by experts.
- The ullage volume has been considered to be 50L for both propellants.
- The project design margin is fixed to 5%.

	[%] LOX	[%] Kerosene	LOX	Kerosene
Total loaded propellant mass [kg]			35278	13553
Temperature propellant at lift-off [K] (information only)			*90,1*	*285*
Density propellants [kg/m³]			1141,97	802,685
Total propellant loaded volume [m³]			**30,892**	**16,885**
	LOX	*Kerosene*		
Ullage volume needed [input: % or m³]	2%	2%		
			0,618	0,338
Project margin (foaming vol., internal tank equ., design margin, tank volume var.)	5%		1,545	0,844
			33,055	**18,067**
Tank Volume [m³]			**51,0**	

Figure 6.27: Tank volume results for the first stage of the 2-stage launcher

With those inputs and the associated calculations, the total tank volume for LOX is ca. $\mathbf{33}m^3$, and for Kerosene ca. $\mathbf{18}m^3$. This results in a total tank of $\mathbf{51}m^3$.

Helium budget
With the results from the tank volume calculations, the required helium mass can be computed. The manual inputs needed to do the calculations are summarized below. The Excel sheet illustration is shown in Figure 6.28.

		LOX	Kerosene
Engine mass flow [kg/s]	20,8		
Propellant Flowrate [kg/s]		15,0	5,8
Density Propellant [kg/m³]		1141,97	802,685
Prop. Volumen Flowrate [m³/s]		0,013	0,007
Vol. Flowrate Pressure Gas [m³/s]		0,013	0,007
Tank Pressure [bar]		4	4
Ullage Temperature [K]		175	175
R Pressure Gas [J/(mol*K)]		2076,3	2076,2
Density Pressure Gas [kg/m³]		1,101	1,101
Mass-Flowrate Pressure Gas [kg/s]		0,014	0,008
Propellant to be fed out [kg]	48492	35033	13459
Pressurization duration [s]		2332,1	2329,5
Pressure Gas need [kg]	**52**	33,8	18,5
Command system consumption [kg]		0,5	
Project Margin [kg]	5%	2,6	
Total Need Pressure Gas [kg]		**55,4**	

Figure 6.28: Pressure gas budget results for the first stage of the 2-stage launcher

- The engine mass flow is an engine characteristic: $\dot{m}_{engine} = 20.8kg/m^3$.
- The tank pressure in both tanks is considered equal to 4bar.
- The ullage temperature in both tanks is assumed for 175K.

- The Helium consumption of the command system is set to 0.5kg, this has been chosen according to an expert's statement.

With those inputs and the associated calculations, the helium mass needed for the pressurization is **55.4kg**.

With the previous results, the maximal expected operating pressure is calculated. The Excel sheet illustration is shown in Figure 6.29.

	LOX	Kerosene
Maximum He regulated pressure [bar]	6,00	6,00
Helium mass needed	55	
He consumption variations (5% incl. leakage) [kg]	2,77	
Helium need for propellant expulsion only [bar]	300,54	
Helium budget non-consumable [bar]	36	
He consumption deviations [bar]	10,00	
He budget uncertainty & margin [bar]	10,00	
Total HP need for He storage [bar]	356,54	
He pressure decrease due to temp. compensation [bar]	10,0	
He pressure rise due to temp. compensation [bar]	10,0	
MEOP He vessel [bar]	**376,5**	

Figure 6.29: MEOP Helium results for the first stage of the 2-stage launcher

- The maximum regulated pressure is for both tanks 6bar. This pressure represents the maximum propellant tank pressure during the mission plus margins.
- From today's upper stage, a value of 10bar is considered for the consumptions deviations.
- 10bar is considered for the uncertainty and margins
- The Helium pressure decrease or rise after the last pressurization on ground due to temperature compensations requires a pressure amount of 10bar.

The Helium vessel volume has to be adjusted in order to get a MEOP less than 410bar. In this study, a volume of **900L** has been considered, that means the MEOP is equal to ca. **376bar**. From existing Helium vessel, the vessel mass has been scaled and the result is **80kg**.

Structural mass calculation results

The structural mass calculation tool has been modified for the purpose of this study. This section summarizes the results of the first stage design. As the propellant mass is important, the common bulkhead with a load bearing tank configuration is clearly the lightest one. It is the configuration which is usually used for the first stage, so only the results concerning this configuration are presented in the following part.

The load bearing common bulkhead tank architecture
Independent from the tank architecture, inputs are required for the structural mass calculations. These inputs are summarized below.

- The maximal axial and lateral acceleration are assumed by an expert: $k_{ax} = 5g$ and $k_{lat} = 1g$.
- The LOX propellant is placed on the top of the common bulkhead configuration. The propellant mass, tank volume and ullage pressure are the ones used in the functional budget: $m_{LOX} = 35449kg$, $V_{LOX} = 33215L$, $p_{ull,LOX} = 4bar$.
- The Kerosene propellant is placed on the bottom of the common bulkhead configuration. The propellant mass, tank volume and ullage pressure are the ones used in the functional budget: $m_{Kerosene} = 13675kg$, $V_{Kerosene} = 18229L$, $p_{ull,Kerosene} = 4.5bar$.
- The characteristics of the structural material have to be implemented. Aluminum 2219 and the CFRP M40/6376 sandwich are used. For the aluminum, the required inputs are: yield stress $\sigma_{Alu} = 358N/mm^2$, modulus of elasticity $E = 72000N/mm^2$ and density $\rho_{Alu} = 2851kg/m^3$. Concerning the composite materials, the following inputs are needed: modulus of elasticity $E_{CFRP} = 96386N/mm^2$, core density $\rho_{core} = 50kg/m^3$ and skin density $\rho_{skin} = 2296kg/m^3$.
- The payload mass takes into account the mass of the upper stage and the payload mass.

As the tanks are part of the structure, the tank radius is the stage radius. The radius which enables to get the lightest tank structural mass is around 1500mm, so the chosen stage radius is 1500mm. No structure is considered above the tank as the top of the tank is directly connected to the interstage structure with the upper stage. A structure below the tank is considered in order to protect the helium vessels and equipment. The length of this structure is equal to 1800mm and the diameter is equal to 3000mm.

The structural mass calculation results are shown in Figure 6.30. The total structural mass of the first stage with the new Zenit engine is **1306kg**. This mass does not include the subsystems. A mass break-down and simulations with the complete launcher have been done once again to see the performances.

6.4.5 The upper stage design

The upper stage needs to be designed in detail. This section presents the functional budget results and the structural mass calculations results. First the engine used is presented.

The engine

It has been decided to consider the 22kN thrust engine for the upper stage. The characteristics of the engine are summarized in the table below.

Entity	Value
Oxidizer	NTO
Fuel	MMH
Mixtur Ratio $[-]$	1.9
Mass flow $[kg/s]$	10
Nozzle diameter $[mm]$	438
Engine length $[mm]$	823

Table 6.27: The 22kN thrust engine characteristics

Used Material	Tank 1	thickness t [mm]	m [kg]	with Contingency
ALU 2219	Upper Dome	1,3	31,8	1
	due to strength (yield)	1,2		
ALU (scaled from ESC-A)	Upper Y-Ring		61,5	
	shell thickness: pressure	3,5		
	shell thickness: buckling	1,7		
	due to buckling	0,0		
ALU 2219	Cylinder Structure with t_max	3,5	309,8	1
ALU 2219	Common Bulkhead	1,9	47,0	1
	due to strength (yield)	1,9		
ALU (scaled from ESC-A)	Bulkhead Y-Ring		61,5	
	Total Mass		**511,4**	
	Tank 2	thickness t [mm]	m [kg]	
	Tank 2 Cylinder			
	shell thickness: pressure	2,8		
	shell thickness: buckling	1,7		
ALU 2219	Cylinder Structure with t_max	2,82	195,7	1
ALU 2219	Lower Tank Dome	1,5	37,0	1
	due to strength (yield)	1,5		
ALU (scaled from ESC-A)	Lower Y-Ring		61,5	
	Total Mass		**294,2**	
	Helium Vessel	thickness t [mm]	m [kg]	
	Total Mass		**200,0**	
	Engine Thrust Frame	thickness t [mm]	m [kg]	
ALU (scaled from ESC-B)	ETF Structure		160,19	1,5
	Total Mass		**160,2**	
	Skirt	thickness t [mm]	m [kg]	
Sandwich	*2 x t_face sheet: strength (scaled)*	1,0		
	2 x t_face sheet: buckling (approximation)	0,7		
	thickness core (estimate)	15,0		
	ISS Structure	1,3	117,9	1,2
ALU (scaled from AVUM)	ISS upper ring+lower ring		8,8	
ALU (scaled from AVUM)	Rivet		9,2	
	Bolts, Screws		4,0	
	Total Mass		**139,9**	
	Total Structural Mass [kg]		**1305,6**	

	Total Tank Mass [kg]		**805,6**	
	Total Structural Mass without tanks [kg]		**500,0**	

Figure 6.30: Structural mass, Common bulkhead load bearing tank, first stage, 2-stage launcher

Functional budget results

The functional design has been done using the Excel tool presented in chapter 3.5. This section summarizes the results.

Propellant budget
The manual inputs for the propellant budget are summarized below. The Excel sheet illustration is shown in Figure 6.31.

		NTO	MMH
Number of boosts	2		
Propellant mixture ratio (MR)	1,9		
Module propulsive mixture ratio (MRu)	1,90		
Nominally used propulsive propellant mass (main. prop. + att. control) [kg]	2000	1310	690
Performance reserve (input either as % or comp. with MRu)	1,5%	20	10
Transient phase engine consumption [kg] (conservative)		20	8
Lines priming: m_lines [kg]		5,0	4,0
m_1: Enhanced nom. propellant mass [kg]	**2067**	1355	712
Geometrical residuals [kg]		15	9
Project design margin (of nominally used prop. mass) [%]	4%	52	28
m_2: Loaded propellant mass [kg]	**2171**	1422	749
Loading process inaccuracy: I (of loaded mass) [%]	0,7%	10	5
		1432	754
Total loaded propellant mass [kg]		**2186**	

Figure 6.31: Propellant budget results for the upper stage of the 2-stage launcher

- The number of boosts is equal to 2, as a re-ignition is done to circularize the orbit
- The propellant mixture ratio is preset by the engine characteristics and so MR=1.9
- The nominally used propellant is adapted from the simulations and so $m_{prop} = 3000kg$
- The value of a performance reserve is evaluated by an expert and it has been considered 1.5% of the nominally used propellant
- The propellant mass for the geometrical residuals is calculated considering 10 liters of each propellant is lost in geometrical residuals.
- The project design margin is set up to 4%
- The value of 0.7% is used for the loading process inaccuracy and this value is estimated by experts

With those inputs and the associated calculations, the total loaded propellant for NTO is ca. **1432kg**, and for MMH ca. **740kg**. So the total loaded propellant is **2186kg**.

Tank volume
With the results from the propellant budget, the tank volume can be calculated as explained in the method in chapter 3.5. The manual inputs needed to do the calculations are summarized below. The Excel sheet illustration is shown in figure 6.32.

- Both propellant densities are required: $\rho_{NTO} = 1428kg/m^3$ and $\rho_{MMH} = 875kg/m^3$.

	[%]		NTO	MMH
Total loaded propellant mass [kg]			1432	754
Temperature propellant at lift-off [K] (for information only)			*300*	*300*
Density propellants [kg/m³]			1428	875
Total loaded propellant volume: V_load [m³]			**1,003**	**0,862**
	NTO	*MMH*		
Ullage volume needed: V_ull [input: % or m³]			0,050	0,050
			0,050	0,050
Internal equipments			0,015	0,015
Project margin M_proj [%]	5%		0,050	0,043
			1,118	**0,970**
Tank Volume [m³]			**2,1**	

Figure 6.32: Tank volume results for the upper stage of the 2-stage launcher

- A volume of 15L for both propellants has been considered for the internal equipment. This value is validated by experts.
- The ullage volume has been considered to be 50L for both propellants.
- The project design margin is fixed to 5%.

With those inputs and the associated calculations, the total tank volume for NTO is ca. $\mathbf{1.1}m^3$, and for MMH ca. $\mathbf{0.9}m^3$. This results in a total tank of $\mathbf{2.1}m^3$.

Helium budget
With the results from the tank volume calculations, the required helium mass can be computed. The manual inputs needed to do the calculations are summarized below. The Excel sheet illustration is shown in Figure 6.33.

- The engine mass flow is an engine characteristic: $\dot{m}_{engine} = 10kg/m^3$.
- The tank pressure in both tanks is considered equal to 21bar. This value is evaluated from today's upper stage.
- The ullage temperature in both tanks is assumed for 250K.
- The mass needed to pressurize the tank from 5bar to 21bar has been evaluated by experts and is equal to 0.3kg.
- The Helium consumption of the command system is decided to be 0.3kg, this has been chosen according to an expert's statement.

With those inputs and the associated calculations, the helium mass needed for the pressurization is **8.1kg**.

With the previous results, the maximal expected operating pressure is calculated. An Excel sheet illustration is shown in Figure 6.34.

- The maximum regulated pressure is for both tanks 25bar. This pressure represents the maximum propellant tank pressure during the mission plus margins.
- From today's upper stage, a value of 10bar is considered for the consumptions deviations.
- 10bar are considered for the uncertainty and margins

		NTO	MMH
Engine mass flow: m-dot_engine [kg/s]	10		
Propellant flowrate: m-dot [kg/s]		6,6	3,4
Density propellant: ρ [kg/m³]		1428	875
Prop. volume flowrate: v-dot [m³/s]		0,005	0,004
Vol. flowrate pressure gas: v-dot_He [m³/s]		0,005	0,004
Tank pressure: p [bar]		21	21,0
Ullage temperature press. gas: T_ull [K]		250	250
Specific gas const. pressure gas: R_He [J/(mol*K)]		2075,7	2075,8
Density pressure gas: ρ_He [kg/m³]		4,047	4,047
Mass-flowrate pressure gas: m-dot_He [kg/s]		0,019	0,016
Propellant to be fed out [kg]	2067	1355	712
Pressurization duration [s]		206,8	206,5
Pressure Gas need [kg]	7	3,8	3,3
Tank pressurization (5bar - 20bar)		0,3	
Command system consumption [kg]		0,3	
Project margin: M_proj [kg]	5%	0,4	
Total need pressure gas: m_He,tot [kg]		**8,1**	

Figure 6.33: Pressure gas budget results for the upper stage of the 2-stage launcher

	NTO	MMH
Maximum He regulated pressure [bar]	25,00	25,00
Needed Helium mass m_He, tot [bar]	8	
He consumption variations (5% incl. leakage) [kg]	0,41	
Helium need for propellant expulsion only [bar]	265,62	
Helium budget non-consumable [bar]	55	
He consumption deviations [bar]	10,00	
He budget uncertainty & margin [bar]	10,00	
Total press. need for He storage. P_He,stor [bar]	340,62	
He pressure decrease due to temp. compensation [bar]	10,0	
He pressure rise due to temp. compensation [bar]	10,0	
MEOP He vessel [bar]	**360,6**	

Figure 6.34: MEOP Helium results for the upper stage of the 2-stage launcher

- the Helium pressure decrease or rise after the last pressurization on ground due to temperature compensations requires a pressure amount of 10bar.

The Helium vessel volume has to be adjusted in order to get a MEOP lower than 410bar. In this study, a volume of **200L** has been considered, that means the MEOP is equal to ca. **360bar**. From existing Helium vessel, the vessel mass has been scaled and the result is **50kg**.

Structural mass calculation results

The Structural Mass Calculation Tool has been modified for the purpose of this study. This chapter summarizes the results of the upper stage design. Considering the advice of mechanical engineers about the upper stage tank configuration, the load-bearing common bulkhead configuration has not been studied as this configuration does not permit any direct access between the lower part and upper part of the stage. Therefore, two designs have been studied, the non load-bearing common bulkhead configuration and the spherical-cylindrical configuration.
The functional budget analysis has determined the required propellant mass and the tank volumes for the 2-stage small launcher.

Common inputs for the two possible configurations
Independant from the tank architecture, inputs are required for the structural mass calculations. These inputs are summarized below.

- The maximal axial and lateral acceleration are assumed by an expert: $k_{as} = 5g$ and $k_{lat} = 1g$.
- The NTO propellant is placed on the top in the common bulkhead configuration and in the sphere in the spherical-cylindrical configuration. The propellant mass, tank volume and ullage pressure are from the functional budget: $m_{NTO} = 1432kg$, $V_{NTO} = 1118L$, $p_{ull,NTO,CB} = 21bar$ in the common bulkhead configuration and $p_{ull,NTO,Sph} = 21.5bar$in the spherical configuration.
- The MMH propellant is placed on the bottom in the common bulkhead configuration and in the cylinder in the spherical-cylindrical configuration. The propellant mass, tank volume and ullage pressure are from the functional budget: $m_{MMH} = 754kg$, $V_{NTO} = 970L$, $p_{ull,MMH,CB} = 21.5bar$ in the common bulkhead configuration and $p_{ull,MMH,Sph} = 21bar$ in the spherical configuration.
- The characteristics of the structural material have to be implemented. Aluminum 2219 and the CFRP M40/6376 sandwich are used. For the aluminum, the required inputs are: yield stress $\sigma_{Alu} = 358N/mm^2$, modulus of elasticity $E = 72000N/mm^2$ and density $\rho_{Alu} = 2851kg/m^3$. Concerning the composite materials, the following inputs are needed: modulus of elasticity $E_{CFRP} = 96386N/mm^2$, core density $\rho_{core} = 50kg/m^3$ and skin density $\rho_{skin} = 2296kg/m^3$.
- The payload mass is fixed to $250kg$.
- The fairing mass is calculated from the AVUM fairing. As the considered payload is six times smaller than the AVUM payload, so the payload mass is considered 3 times smaller. As the AVUM fairing mass is equal to $560kg$, in this study it has been considered a fairing of $180kg$.

The non load bearing common bulkhead tank architecture
Regarding the tank volume, the tank diameter which gives the smaller tank mass is around 1200mm, so this diameter of 1200mm has been chosen for the tanks. The space between the tank shells and the structure is equal to 150mm, and so the stage diameter is equal to 1500mm. The length of the cylinder and domes, as well as the surfaces of the different parts of the tanks are summarized in Table 6.28. For the calculations of the axial fluxes, the diameter and length of the different sections are required. The diameter of all the sections is 1500mm. The length of section 1, which is the length of the fairing, is 3200mm. Section 2, which is the length of the intermediate structure, is 1800mm long and section 3, which is the length of the interstage skirt, is 3500mm. These dimensions have been chosen regarding the space needed for all the components and tank length. The axial fluxes are shown in Figure 6.35.

	Tank 1	Tank 2
Cylinder radius $[mm]$	600	600
Cylinder length $[mm]$	423	857
Dome length $[mm]$	424	424
Cylinder surface $[m^2]$	1.59	3.23
Dome surface $[m^2]$	1.37	1.37

Table 6.28: Tank geometrical characteristics, not load bearing common bulkhead configuration

load for loadb. tanks:

Sections	Length [mm]	Diam. [mm]	Axial Fluxes: Fax [N]	M [Nmm]	n_Fax	n_M	n_max	considered masses
length - fairing top to sec	3200	1700	21092	32083887	4	14	**18**	***P/L +fairing +wind***
length - sec 1 to sec 2	1800	1500	131356	88025853	28	50	**78**	***+ prop+helium storage***
length - sec 3 to sec 4	3500	1500	135280	236767836	29	134	**163**	***+ engine***

load for not loadb. tanks:

Sections	Length [mm]	Diam. [mm]	Fax [N]	M [Nmm]	n_Fax	n_M	n_max	considered masses
length - sec 1 to sec 2	2553	1200	2186	27377369	1	24	**25**	***prop***

Figure 6.35: Axial fluxes, non load-bearing Common Bulkhead configuration, upper stage, 2-stage launcher

The structural mass calculation results are shown in Figures 6.36 and 6.37. The calculation method is the one explained in chapter 3, section 5. The total structural mass of the upper stage is **454.6kg**. This mass does not include the fairing, payload, avionics and subsystsem masses.

The spherical-cylindrical tank architecture

From the propellant volume, the tank dimensions are calculated. The geometries of the cylinder and sphere, as well as the surfaces of the different parts of the tanks, are summarized in Table 6.29.

	Tank 1 (cylinder)	Tank 2 (sphere)
Cylinder radius $[mm]$	463	
Cylinder length $[mm]$	1288	
Cap length $[mm]$	149	
Cylinder surface $[m^2]$	3.74	
Cap surface $[m^2]$	0.43	
Sphere radius $[mm]$		793
Sphere surface $[m^2]$		7.04

Table 6.29: Tank geometrical characteristics, spherical-cylindrical configuration

For the calculations of the axial fluxes, the diameter and the length of the different sections are required. The diameter of all the sections is 1800mm. This diameter is chosen in order to have a space of 100 mm between the tank shells and the structure. The length of section 1, which is the fairing length, is 3200mm. Section 2 is the length of the intermediate structure and is 1400mm long. Section 3, which is the length of the interstage skirt, is 3100mm long. These dimensions have been chosen regarding the space needed for all the components and tank length. The axial fluxes are shown in Figure 6.38.

The structural mass calculation results are shown in Figure 6.39 and 6.40. The calculation method is the one presented in chapter 3, section 5.

The total structural mass of the upper stage is **524.7kg**. This mass does not include fairing,

Used Material	Upper Cone structure	thickness t [mm]	m [kg]	with Contingency
Cone 3936	Upper Cone / Payload adapter		45	
	Avionic Platfrom		22	
	Total Mass		**67,0**	
	Tank 1	thickness t [mm]	m [kg]	
ALU 2219	Upper Dome	2,0	7,8	1
	due to strength (yield)	2,0		
ALU (scaled from ESC-A)	Upper Y-Ring		20,6	
	shell thickness: pressure	*4,1*		
	shell thickness: buckling	*0,6*		
	due to buckling	0,0		
ALU 2219	Cylinder Structure with t_max	4,1	18,5	1
ALU 2219	Common Bulkhead	2,1	8,0	1
	due to strength (yield)	2,1		
ALU (scaled from ESC-A)	Bulkhead Y-Ring		20,6	
	Total Mass		**75,7**	
	Tank 2	thickness t [mm]	m [kg]	
	Tank 2 Cylinder			
	shell thickness: pressure	*4,1*		
	shell thickness: buckling	*0,6*		
ALU 2219	Cylinder Structure with t_max	4,13	38,0	1
ALU 2219	Lower Tank Dome	2,1	8,1	1
	due to strength (yield)	2,1		
ALU (scaled from ESC-A)	Lower Y-Ring		20,6	
	Tank 2 Sphere:			
	shell thickness: pressure	*3,04*		
	due to strength (yield)	*3,04*		
ALU 2219	Sphere Structure	0,00	0,0	1,0
	Total Mass		**66,8**	
	Helium Vessel	thickness t [mm]	m [kg]	
	Total Mass		**50,0**	

Figure 6.36: Structural mass part 1, non load-bearing Common Bulkhead configuration, upper stage, 2-stage launcher

	Intermediate Structure (1 to 2)	thickness t [mm]	m [kg]	
	2 x t_face sheet: strength (scaled)	0,2		
	2 x t_face sheet: buckling (approximation)	0,2		
Sandwich	thickness core (estimate)	10		
	Structure sec 1 to 2	1,3	35,5	1,2
scaled from AVUM	Upper ring + Lower ring		4,4	
	Bolts + screws		4,0	
	Total Mass		**43,9**	
	Engine Thrust Frame	thickness t [mm]	m [kg]	
ALU (scaled from ESC-B)	ETF Structure		32,71	1,5
	Total Mass		**32,7**	
	CTAR	thickness t [mm]	m [kg]	
scaled	CTAR		14,15	
	Total Mass		**14,2**	
	Interstage Skirt (2 to 3)	thickness t [mm]	m [kg]	
Sandwich	*2 x t_face sheet: strength (scaled)*	0,4		
	2 x t_face sheet: buckling (approximation)	0,3		
	thickness core (estimate)	10,0		
	ISS Structure	1,3	91,4	1,2
ALU (scaled from AVUM)	ISS upper ring+lower ring		4,4	
ALU (scaled from AVUM)	Rivet		4,6	
	Bolts, Screws		4,0	
	Total Mass		**104,3**	
	Total Structural Mass [kg]		**454,6**	
	Total Tank Mass [kg]		**142,5**	
	Total Structural Mass without tanks [kg]		**312,1**	

Figure 6.37: Structural mass part 2, non load-bearing Common Bulkhead configuration, upper stage, 2-stage launcher

load for loadb. tanks:

Sections	Length [mm]	Diam. [mm]	Axial Fluxes: Fax [N]	M [Nmm]	n_Fax	n_M	n_max	considered masses
length - fairing top to sec	3200	1700	21092	32083887	4	14	**18**	*P/L +fairing +wind*
length - sec 1 to sec 2	1400	1800	131356	75594305	23	30	**53**	*+ prop+helium storage*
length - sec 3 to sec 4	3100	1800	135280	207337205	24	81	**105**	*+ engine*
load for not loadb. tanks:								
length - sec 1 to sec 2	1586	1586	2186	17003296	0	9	**9**	*prop*

Figure 6.38: Axial fluxes, Spherical-Cylindrical configuration, upper stage, 2-stage launcher

Used Material	Upper Cone structure	thickness t [mm]	m [kg]	with Contingency
Cone 3936	Upper Cone / Payload adapter		45	
	Avionic Platfrom		22	
	Total Mass		**67,0**	
	Tank 1	thickness t [mm]	m [kg]	
ALU 2219	Upper Dome	1,5	1,9	*1*
	due to strength (yield)	1,5		
ALU (scaled from ESC-A)	Upper Y-Ring		0,0	
	shell thickness: pressure	*3,5*		
	shell thickness: buckling	*0,6*		
	due to buckling	*12,1*		
ALU 2219	Cylinder Structure with t_max	12,1	129,2	*1*
ALU 2219	Common Bulkhead	1,6	2,0	*1*
	due to strength (yield)	1,6		
ALU (scaled from ESC-A)	Bulkhead Y-Ring		0,0	
	Total Mass		**133,0**	
	Tank 2	thickness t [mm]	m [kg]	
	Tank 2 Cylinder			
	shell thickness: pressure	*0,0*		
	shell thickness: buckling	*0,6*		
ALU 2219	Cylinder Structure with t_max	0,00	0,0	*1*
ALU 2219	Lower Tank Dome	1,3	0,0	*1*
	due to strength (yield)	0,0		
ALU (scaled from ESC-A)	Lower Y-Ring		0,0	
	Tank 2 Sphere:			
	shell thickness: pressure	*3,04*		
	due to strength (yield)	*3,04*		
ALU 2219	Sphere Structure	3,04	61,0	*1,0*
	Total Mass		**61,0**	
	Helium Vessel	thickness t [mm]	m [kg]	
	Total Mass		**50,0**	

Figure 6.39: Structural mass part 1, Spherical-Cylindrical configuration, upper stage, 2-stage launcher

payload, avionics and subsystems masses.

The solution

The tank configuration, which gives the lightest structure, is a non load-bearing common bulkhead configuration. This configuration has a lighter tank structure and stage structure than the Spherical-Cylindrical tank configuration. Therefore, this configuration is chosen. The upper stage with the non load-bearing common bulkhead configuration has been chosen and studied in more detail. A mass breakdown has been done and simulations with the complete launcher have been done again in order to see the performance of this launcher.

6.4.6 The 2-stage launcher

With the results from the simulations and the systems engineering tools, the configuration of the two-stage launcher is defined. In this part, the architecture of the launcher is described and more

	Intermediate Structure (1 to 2)	thickness t [mm]	m [kg]	
	2 x t_face sheet: strength (scaled)	0,1		
	2 x t_face sheet: buckling (approximation)	0,2		
Sandwich	thickness core (estimate)	10		
	Structure sec 1 to 2	1,3	33,1	*1,2*
scaled from AVUM	Upper ring + Lower ring		5,3	
	Bolts + screws		4,0	
	Total Mass		**42,4**	
	Engine Thrust Frame	thickness t [mm]	m [kg]	
ALU (scaled from ESC-B)	ETF Structure		20,64	*1,5*
	Total Mass		**20,6**	
	CTAR	thickness t [mm]	m [kg]	
scaled	CTAR		49,51	
	Total Mass		**49,5**	
	Interstage Skirt (2 to 3)	thickness t [mm]	m [kg]	
Sandwich	*2 x t_face sheet: strength (scaled)*	*0,3*		
	2 x t_face sheet: buckling (approximation)	*0,3*		
	thickness core (estimate)	10,0		
	ISS Structure	1,3	86,3	*1,2*
ALU (scaled from AVUM)	ISS upper ring+lower ring		5,3	
ALU (scaled from AVUM)	Rivet		5,5	
	Bolts, Screws		4,0	
	Total Mass		**101,1**	
	Total Structural Mass [kg]		**524,7**	
	Total Tank Mass [kg]		**194,0**	
	Total Structural Mass without tanks [kg]		**330,7**	

Figure 6.40: Structural mass part 2, Spherical-Cylindrical configuration, upper stage, 2-stage launcher

particularly the architecture of the upper stage is defined thanks to CAD drawings. The total dry masses of the first stage and upper stage have been calculated as well. The upper-stage CAD drawing are especially generated for this study by the design department of Astrium GmbH in Bremen.

The architecture

The two-stage launcher is composed of a first stage using the Zenit booster which is still under investigation and an upper stage; both structures are especially designed for this study. A complete drawing of the 2-stage launcher is represented in Figure 6.41. A complete definition of the two stages is done in the following part.

Concerning the upper stage with the 20kN thrust engine, the common bulkhead as a non-load bearing propellant tank is chosen regarding the structural mass. The stage diameter is 1500mm and the cylindrical tank diameter is 1200mm in order to have a space of 150mm between the tank shells and the stage structure. The interstage between the first stage and the stage structure is a conic shape; the big diameter is equal to 3000mm, which is the diameter of the first stage. The fairing is placed on the top of the upper stage. The fairing length is equal to 3200mm and the maximal diameter of the fairing is 1700mm. The fairing geometry is chosen regarding the fairings of current upper stages and more precisely regarding the one from the VEGA launcher.

The figure 6.42 and 6.43 illustrates the upper stage concept with all components. The technical drawing of the upper stage can be found in Annex *coming soon*. The upper cone is designed to be the upper interface to the payload and its design is based on an existing payload adapter. The big diameter of the cone is equal to the stage diameter and the small diameter is equal to 800mm. Additionally, an avionics platform is placed at the top of the stage, it enables the attachment of all the avionics equipment and a vessel as well which is placed in the middle of the platform. The attitude control system is the one used on the VEGA launcher: RACS (Roll and

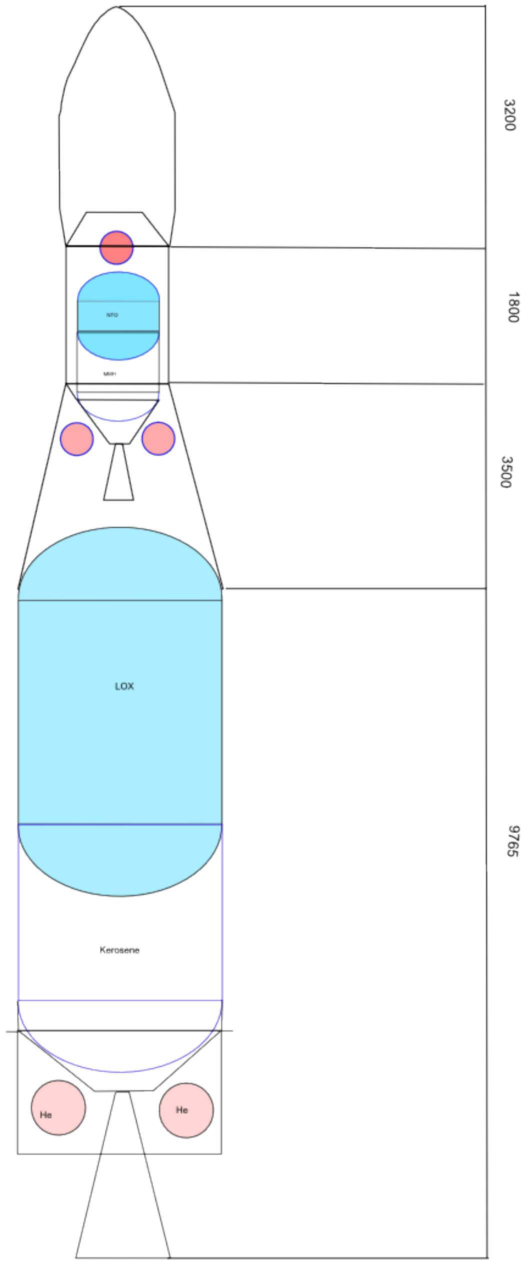


Figure 6.41: The two-stage launcher architecture

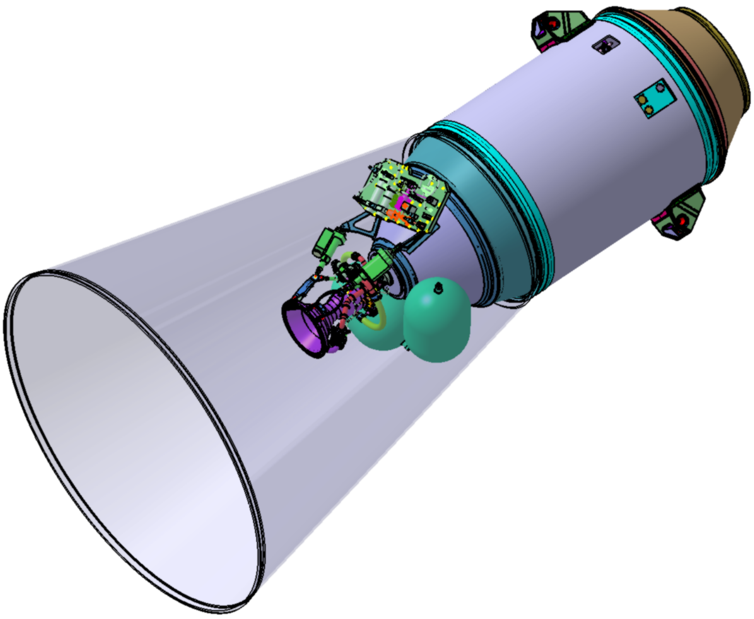


Figure 6.42: 3D drawing of the upper stage, 2-stage launcher

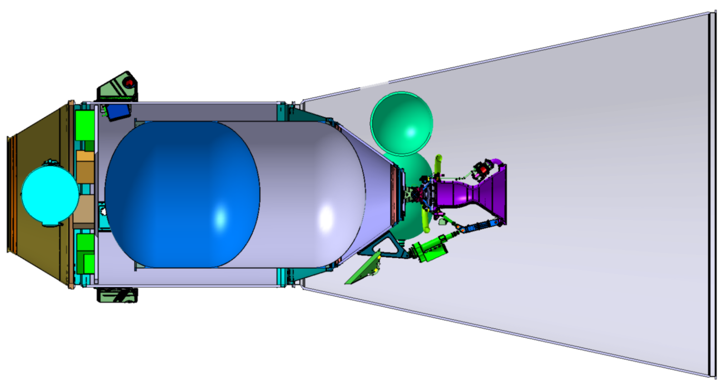


Figure 6.43: Section cut drawing of the upper stage, 2-stage launcher

Attitude Control System). This system requires a hydrazine vessel with 38kg of hydrazine. This vessel has a diameter of 482mm and is placed in the middle of the avionics platform. The two thruster cluster modules of the RACS are diametrically opposite placed at the top of the stage.
At the stage's interstage structure, the common bulkhead NTO/MMH tank is attached thanks to the CTAR. The engine thrust frame is directly attached to the lower cylinder and the 22kN engine is attached to the ETF. The pressure control system is attached to the ETF as well as the helium vessels. The helium vessel configuration is cylindrical with two spherical domes. Two helium vessels with a diameter of 480mm are required. The interstage structure represents the connection with the first stage, it is placed directly at the top of the first stage tank.
The total length of the upper stage, including the cone and the interstage structure, is 5800mm.

Concerning the first stage with the Zenit booster, the common bulkhead as a load bearing propellant tank is chosen regarding the structural mass. The stage diameter is equal to 3000mm which is the tank diameter as well. A skirt is placed below the tanks in order to protect the helium vessels and other equipment as the pressure control system. This skirt length is equal to 1800mm. Two helium vessels are required; they have a cylindrical shape with two spherical domes. The helium vessel diameter is 400mm. The engine thrust frame is directly attached to the structure and enables connection to the Zenit engine.
The total length of the stage is equal to 9765mm

The total length of the launcher including the fairing is **18265mm**.

The mass breakdown

After the definition of the first and upper stage, the total mass of each stage has to be evaluated. The Structural Calculation Software Tool has enabled the calculation of the tank mass and the main structural component masses, but there is still the need to evaluate the mass of the remaining equipment. Therefore, a mass breakdown has been established for both stages. The different subsystem masses have been chosen regarding existing studies and validated by an expert.

First stage mass breakdown
The following items have been considered in the mass break down:

- propulsion system
- equipped tanks (propellant tank, helium vessel)
- fluid masses (helium, reserve propellant)
- structures (skirt)
- subsystems (thermal subsystems, neutralization system, electrical system)

The mass breakdown table is represented in Table 6.30. Only the main items presented before are including in this table. They are the relevant components for an upper-stage, the subsystems are defined but not explained in detail.

- The mass of the Zenit engine is 1080kg.
- The calculated mass of the common bulkhead tank thanks to the Structural Calculation Mass Tool is 805kg.
- The 900 liters of helium are stored in two vessels of 100kg each. This mass is evaluated by scaling existing vessels.

- The helium mass evaluated by the Helium budget is 70kg.
- Regarding the propellant budget, 1831kg are considered as reserve propellant.
- The structural mass is calculated with the Structural Calculations Tool. The skirt is equal to 219kg. The ETF mass has been evaluated by scaling from existing systems.
- 161 kg have been considered for the total subsystem mass.

Finally, by summing up all these components, the total first stage dry mass is equal to **4707 kg**.

NAME OF PRODUCT	**MASS** $[kg]$
Equipped tank	**1043**
Equipped propellant tanks	805
Helium vessel	200
Fluid mass	**48939**
Helium	70
Reserve propellant	1831
Nominal propellant	47000
Structure	**379**
Stage structure	219
Additional structural mass	160
Engine thrust frame	*160*
Propulsion System	**1223**
Engine	1080
Pressure control assembly	50
Feed lines	25
Pressurization lines	18
Actuators	40
Fill and drain coupling	10
Subsystems	**161**
Thermal subsystems	60
Neutralisation system	11
Supports	70
Electrical system	20
TOTAL STAGE DRY MASS	**4707**
TOTAL STAGE MASS	**51707**

Table 6.30: Mass breakdown of the first stage for the 2-stage launcher

Upper stage mass breakdown
The following items have been considered in the mass break down:

- propulsion system
- equipped tanks (propellant tank, helium vessel, hydrazine vessel)
- fluid masses (helium, hydrazine, reserve propellant)
- structures (stage structure, interstage skirt, equipments)
- avionics
- RACS
- separation system

- subsystems (thermal subsystems, neutralization system, feed lines, electrical system)

The mass breakdown table is represented in Table 6.31. Only the main items presented before are included in this table. They are the relevant components for an upper-stage, the subsystems are defined but not explained in detail.

- The mass of the 22kN thrust engine is 80kg.
- The calculated mass of the common bulkhead tank, thanks to the Structural Calculation Mass Tool, is 142.2kg.
- The 200 liters of helium are stored in two vessels of 25kg each. This mass is evaluated by scaling existing vessels.
- The hydrazine vessel is the one used for the VEGA launcher, its mass is 8.45kg.
- The helium mass evaluated by the Helium budget is 12kg.
- The Hydrazine mass is the one considered for the VEGA launcher and it is 38kg.
- Regarding the propellant budget, 186kg are considered as reserve propellant.
- The structural mass is calculated with the Structural Calculations Tool. The stage structure is equal to 47.8kg and the interstage skirt is equal to 91.1kg. The upper cone, the avionics platform, the CTAR and the ETF masses have been evaluated by scaling from existing systems.
- The avionic mass considered is the one of the VEGA launcher, 96kg.
- The RACS system is the one of the VEGA launcher. The total mass is 41.75kg.
- The separation system is evaluated by scaling from existing systems, 28kg.
- The subsystem masses are scaled regarding the VEGA launcher, 74 kg has been considered for the total subsystem mass.

Finally, by summing up all these components, the total upper stage dry mass is equal to **1071.2 kg**, with 119 kg of jettisonable mass.

6.4.7 Small launcher simulation

Simulations have been done with the complete launcher. The characteristics of the two stages have been implemented in the MATLAB code. The propellant mass in the first stage has been considered equal to 47000kg, and 2000kg in the upper stage, which are the required nominal propellant masses. Regarding the mass breakdown, the structural mass is equal to 4707kg for the first stage and 952 for the upper stage. The interstage mass is equal to 119kg and the fairing mass is equal to 180kg. The launcher characteristics are summarized in Table 6.32

With this launcher configuration, the MATLAB simulation shows that it is possible to place 270kg of payload into the elliptical transfer orbit 200km perigee - 700km apogee. As there is a need of ca. 35kg of propellant to realize the circularization, the payload which can be placed into the circular orbit with a radius of 700km is around **235kg**.

The simulation is represented in the following figures. Figure 6.44 shows the variation of the altitude, velocity and flight path angle as functions of time and the altitude as a function of velocity, from the launch pad to the elliptical orbit. Each color corresponds to one stage:

NAME OF PRODUCT	JETTISONABLE MASS [kg]	MASS [kg]
Equipped tank		**200.2**
Equipped propellant tanks		142.2
Helium vessel		50
Hydrazine vessel		8
Fluid mass		**2236**
Helium		12
Hydrazine		38
Reserve propellant		186
Nominal propellant		2000
Structure	**91.1**	**116.8**
Stage structure		47.8
Interstage skirt	91.1	
Additional structural mass		114
Upper cone structure		*45*
Avionics platform		*22*
CTAR		*14*
Engine thrust frame		*33*
Avionics		**96**
RACS		**41.75**
Pressure transducers		0.5
Pyrotechnic valves		1
Feed line and drain valve		0.65
Turbing kit		2
Pipe support brackets + bolts		2.7
Thruster cluster module		27
Aerothermal cover		7.9
Separation system	**28**	
Pyrojacks	15.6	
Pyro lines	9.4	
RMV	3	
Propulsion System		**142.1**
Engine		80
Pressure control assembly		15
Feed lines		11
Pressurization lines		9.4
Passivation system		4.5
Actuators		18
Fill and drain coupling		4.2
Subsystems		**74.3**
Thermal subsystems		22.7
Neutralisation system		10
Supports		26.5
Electrical system/Hardware		15.1
TOTAL STAGE DRY MASS	**119.1**	**952.15**
FAIRING	**180**	
TOTAL UPPER STAGE MASS	**299.1**	**3202.15**

Table 6.31: Mass breakdown of the upper stage for the 2-stage launcher

	Stage 1: Zenit	Stage 2
Isp $[s]$	330	216
Thrust $[kN]$	849	22
Propellant mass $[kg]$	47,000	2,000
Dry mass $[kg]$	4,707	952
Interstage mass $[kg]$		119
Length $[mm]$	9,765	5,300
Diameter $[mm]$	3,000	1,500

Table 6.32: The two stage launcher characteristics

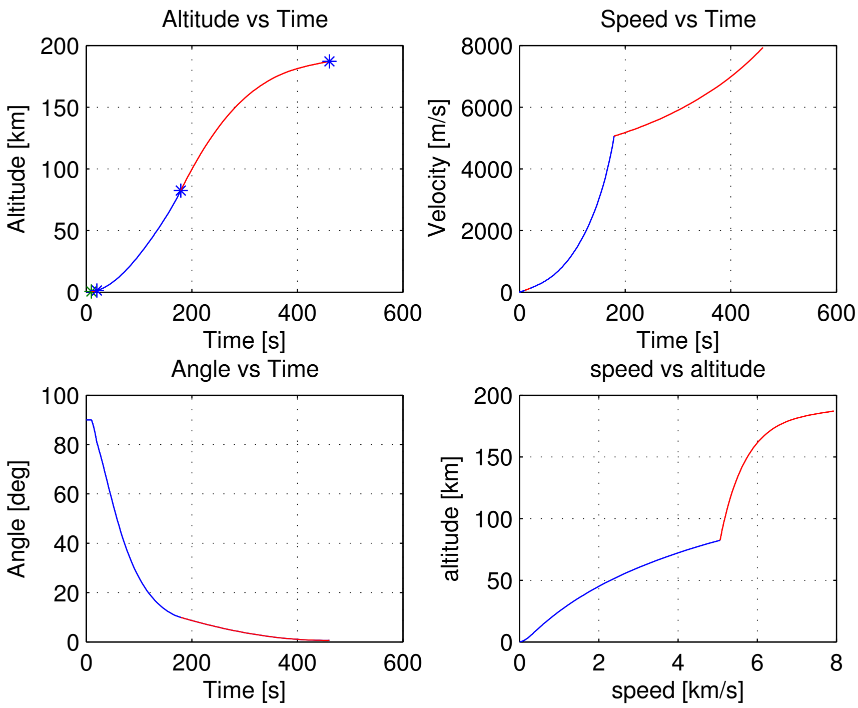


Figure 6.44: 2-stage launcher simulation from launch pad to the elliptical orbit

- the blue curve is the first stage
- the red curve is the upper stage

Figure 6.45 represents the variation of altitude, velocity and flight path angle as functions of time and the altitude as a function of velocity, from the launch pad to the circular orbit. Each color corresponds to a certain part of the trajectory:

- the blue curve is from the launch pad to the elliptical orbit at burnout of the upper stage
- the red curve corresponds to the trajectory in orbit when the thrust is equal to zero
- the green curve corresponds to the trajectory in the circular orbit

The upper stage re-ignition happens when the spacecraft reaches the orbit apogee. Figure 6.46 represents the variation of mass, thrust, dynamic pressure, lateral and longitudinal velocity as functions of time. The lateral velocity is the red curve and the longitudinal velocity is the blue one.

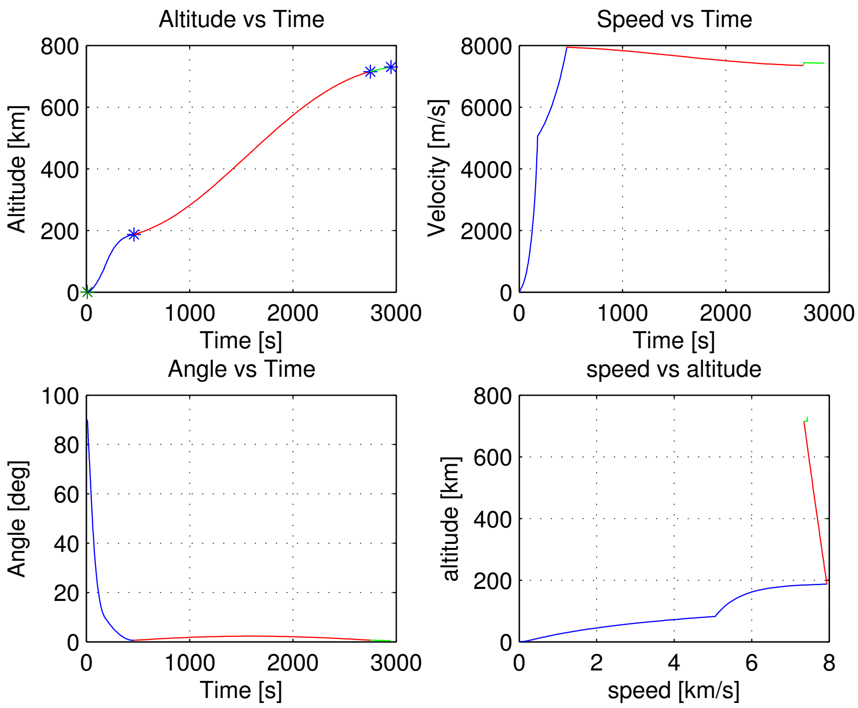


Figure 6.45: 2-stage launcher simulation from launch pad to the circular orbit

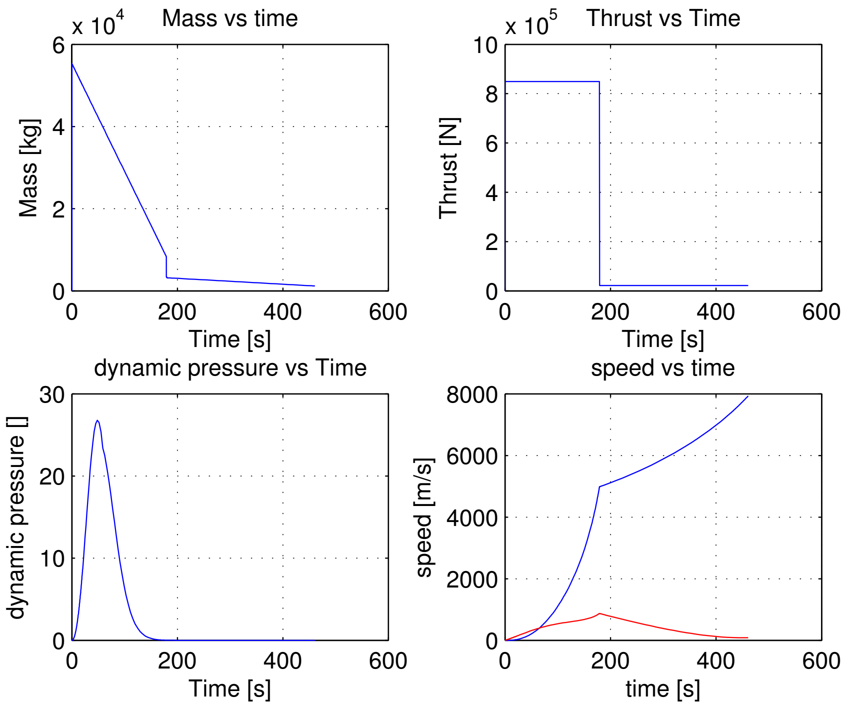


Figure 6.46: 2-stage launcher simulation, other characteristics

Chapter 7

Conclusion

The purpose of this Master Thesis was to perform the conceptual lay-out of a small launcher. The small launcher has to be able to place into orbit 250kg of payload in a circular orbit with a radius of 700km and an inclination of 90 degrees. In order to reduce development costs, the small launcher must use existing stages for the first and/or second stage, whereas the upper stage is entirely designed using an engine under investigation at the company.
This preliminary study included two different aspects: one aspect concerning the launcher performance and the other one concerning the stage design. The launcher performance needed to be studied in order to define the launcher characteristics and to know exactly the amount of payload that the small launcher can carry and place into orbit. A complete new tool has been developed to study the launcher performance. This tool is coded using MATLAB. By setting the launcher characteristics, the tool enables simulation of rocket trajectories. Based on the Nelder-Mead algorithm, the tool sizes, as well as to optimize, the launcher staging and the gravity turn trajectory. The study concerning the stage design is based on a previous tool written in Excel. This tool has been changed and improved in the purpose of this study. It enables the evaluation of the propellant budget, helium budget and structural mass calculations.
Two small launchers have been highlighted from the study: a two-stage launcher using two liquid stages, and a three-stage launcher using two solid stages and a liquid upper stage. After research on existing launchers and preliminary design of the upper stage, the performance tool has been used to choose the different characteristics of the launchers. The purpose was to choose the existing stages which give the best performances and to define the upper stage characteristics. Then, the upper stage for each launcher has been designed in details.

Concerning the three-stage small launcher, research has been done on existing stages using solid propulsion focusing only on boosters with not too high thrust, as the purpose was to define a small launcher. From the research, correlations have been settled between the thrust, specific impulse, structural index and propellant mass in the stages. The launcher characteristics are so defined thanks to its propellant mass. A first rough design has been done for the upper stage in order to get the variations of the structural index as a function of the engine thrust and the amount of propellant which are the two variables which needed to be chosen in the purpose of this study. The specific impulse of the upper stage has always been considered constant as the engine under investigation at the company has the same specific impulse. By using the performance tool with these curves directly implemented in the code, an optimum launcher could be found. The variables were the amount of propellant in the three stages, the thrust in the upper stage and the gravity turn parameters. So, by setting the launcher characteristics and using the optimization program, the purpose was to find the optimum launcher characteristics. From these simulations it has been seen a tendency about the amount of propellant in the different stages, and from this tendency existing stages have been chosen to be study in more detail. The chosen existing stages

are the Italian boosters, Zefiro 23 and Zefiro 9, and the brazilian boosters, S-50 and S-43. By setting these launcher characteristics, simulations have been done in order to find the upper stage characteristics which give the best performances. The variables were so the propellant mass and thrust in the upper stage and the gravity turn parameters. Then, the launcher which gives the best performances has been chosen and the upper stage has been designed in detail. The chosen small launcher uses the Zefiro 23 as a first stage, the Zefiro 9 as a second stage and an upper stage with a 3kN thrust engine and 500kg of propellant mass.
Then, the upper stage has been designed in detail. In order to perform the design study, functional budget concerning the propellant, the pressure gas and the structure have been prepared by using and improving an existing excel tool. From the propellant budget, 603 kg of propellant are required, which corresponds to a volume of $0.7m^3$. From the helium budget, 70 liters of helium are needed, pressurized at 340 bars. Three tank configurations have been studied and the common bulkhead as non load bearing propellant tank has been chosen regarding the structural mass, and according to expert's advices, the direct access between the lower and upper part of the stage. From structural calculations, the propellant tank mass is equal to 53.5 kg, the helium vessel to 15kg and the structure mass to 155.5kg. Then, considering all the important equipments needed in an upper stage, namely the attitude control system, the pressurization system, the avionics, the separation system and the subsystems, a total dry mass of around 710kg is calculated in a total mass breakdown.
Finally, using the performance tool, a final performance assessment has been done on the three-stage launcher. This configuration enables to place into the circular orbit of 700km radius, a payload of 295kg which fulfill all the preset requirements.
This launcher uses the two Italian boosters Zefiro 23 and Zefiro 9 which have been developed for the VEGA launcher. This study has not taken into account any changes on these boosters which would necessarily be required for the development of the small launcher. Furthermore, the Brazil is currently developing boosters. So far, this study shows the Italian boosters give better results than the brazilian boosters but, the Italians profit from the european aerospace knowledge. Therefore, in the future, the Brazil might be able to develop boosters which will have better performances and will be more competitive regarding certainly the production costs.

Concerning the two-stage launcher, research has been done on existing stages using liquid propulsion focusing on stages with not too high thrust as well. From the research and preliminary simulations, it has been noticed that the existing stages using liquid propulsion with thrust enough to lift-off from Earth were too big for the purpose of this study. So, it has been decided to modify the existing liquid stages. Three existing stages have been studied: the Soyuz first stage, the Zenit second stage and the Zenit new booster which is an engine derived from the Zenit second stage engine. The stage modification was on the amount of propellant and so on the structure as well. From the research, the correlation between the structural index and the propellant mass has been found. But, the structural index calculation does not take into account the engine mass. Indeed, as the changes are only done on the amount of propellant, the engine is kept and so its characteristics, thrust and specific impulse, and mass as well. By using the performance tool, the purpose was to find both the amount of propellant in the first stage and the upper stage characteristics which give the best performances. The variables were the amount of propellant in the two stages, the thrust in the upper stage and the gravity turn parameters. From these simulations, the new Zenit booster engine has been chosen regarding its performances. The chosen two-stage launcher uses the new Zenit booster as a first stage with 47000kg of propellant, and an upper stage with 22kN thrust and 2000kg of propellant mass.
Then, the first stage and the upper stage have been designed in detail. The design study has been performed using the existing excel tool in order to get the functional budgets. About the upper stage, from the propellant budget, 2186kg of propellant are required, which correspond to a volume

of $2.1m^3$. From the helium budget, 200 liters of helium are needed, pressurized at 360bars. The common bulkhead as non load bearing propellant tank configuration has been chosen regarding the structural mass, and according expert's advices, the direct access between the lower and upper part of the stage. From structural calculations, the propellant tank mass is equal to 142.2kg, the helium vessel to 50kg and the structural mass to 253kg. Then, considering the equipments needed in an upper stage, a total dry mass of 1065kg is calculated in a total mass breakdown. The same study has been done for the first stage. From the propellant budget, 48831kg of propellant are required, which correspond to $51m^3$. From the helium budget, 900 liters of helium are needed, pressurized at 376bars. The common bulkhead as load bearing propellant tank configuration has been chosen regarding the structural mass. From structural calculations, the propellant tank mass is equal to 805kg, the helium vessel to 200kg and the structural mass to 379kg. Then considering the equipments, a total dry mass of 4707kg is calculated in a total mass breakdown.
Finally, using the performance tool, a final performance assessment has been done on the two-stage launcher. This configuration enables to place into the circular orbit of 700km radius, a payload of 235kg, which fulfill all the preset requirements.
This launcher uses the new Zenit booster which is still under investigation. This engine is derived from the engine used in the Zenit second stage. It has been chosen because the development of this new engine is done in the purpose to be used in a first stage. So, this engine fill the requirements of this launcher, and as its development is done in order to be used in a first stage launcher no change would be needed on it for the future development of the small launcher. Only the stage structure needs to be designed.

Further development can be executed on the basis of this thesis. First, the developed performance tool is a simple tool which enables fast rocket trajectory simulations. It is based on several assumptions in order to have a simple tool but some of the assumptions can be evaluated again. For instance, a study focused on the gravity turn trajectory would enables to have a better understanding of rocket trajectory optimization. Furthermore, the Nelder-Mead algorithm has been chosen regarding its simplicity to be understandable and its quick implementation in the code, but it may not be the best control theory algorithm. Now, as the characteristics of two small launchers have been identified, technology development can be initiated. More detailed design and performance calculations can be started.

As a conclusion for this study, two small launchers have been identified using existing technologies: a three-stage launcher using the Zefiro 23 and Zefiro 9 as first and second stages and a 3kN thrust upper stage, a two-stage launcher using the under-investigation Zenit engine in the first stage and a 22kN thrust upper stage. Both launchers enable to place into a circular orbit with a radius of 700km a payload of nearly 300kg for the three-stage configuration and 235kg for the two-stage configuration.

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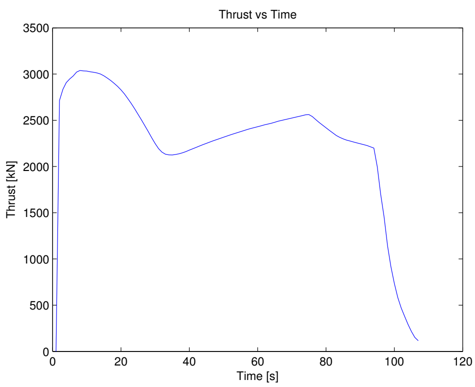
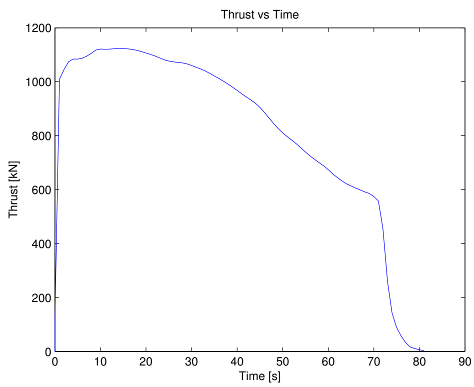
Appendix A

Thrust profile

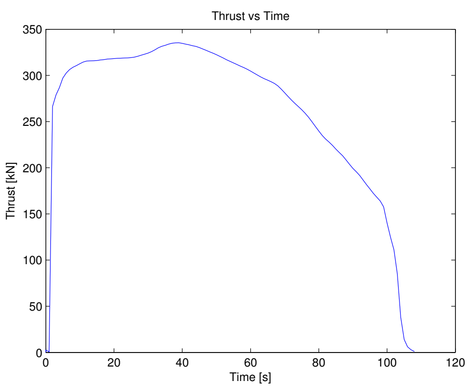
A.1 VEGA boosters thrust profile



(a) P80 thrust profile



(b) Z23 thrust profile



(c) Z9 thrust profile

Figure A.1: Thrust profile of the different VEGA launcher boosters

A.2 Ariane 5 boosters thrust profile

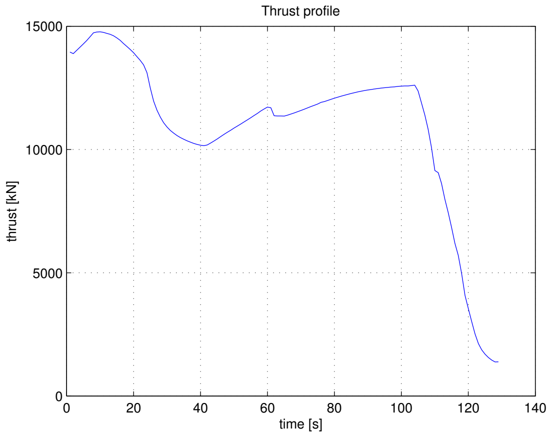


Figure A.2: Thrust profile of the solid booster

Appendix B

Functional Budget Calculation Tool

Functional Sizing of an Upper Stage

- Propellant Budget -

Choose with '0' or '1' the propellants: 0

LOX/ LH2: input = 1

NTO/ MMH: input = 0

inputs

		NTO	MMH
Number of boosts	1		
Propellant mixture ratio (MR)	2,09		
Module propulsive mixture ratio (MRu)	2,09		
Nominally used propulsive propellant mass (main. prop. + att. control) [kg]	500	338	162
Performance reserve (input either as % or comp. with MRu)	2,5%	8	4
Transient phase engine consumption [kg] (conservative)		32	12
Lines priming: m_lines [kg]		4,5	3,6
m_1: Enhanced nom. propellant mass [kg]	**564**	383	181
Number of Chill down after ballistic phase duration > 1 hour)	0	0,0	0,0
Number of Chill down (after ballistic phase duration < 1 hour)	0	0,0	0,0
m_2: Operational propellant consumption [kg]	**564**	383	181
LH2 reconditioning maneuvre [kg] (NTO/MMH: input=0)		0	0,0
Geometrical residuals [kg]		14,28	8,75
Thermal residuals [kg] (NTO/MMH: input=0) [kg]		0	0
Project design margin (of nominally used prop. mass) [%]	5%	17	8
Vaporized mass (in tanks) [kg] (NTO/MMH: input=0)		0	0
m_3: Loaded propellant mass [kg]	**612**	414	198
Loading process inaccuracy: I (of loaded mass) [%]	0,7%	2,90	1,39
		416,73	199,69
Total loaded propellant mass [kg]		**616**	

Assumptions:	
MRe engine mixture ratio deviation (σ_MRe/MRe)	0,0%
MRu usable propellant mixture ratio deviation (σ_MRu/MRu)	0,0%

propellant need (kg)	**ignition**	**shut down**
NTO	10,5	21,0
MMH	4,5	7,5

Figure B.1: Propellant budget excel sheet

Functional Sizing of an Upper Stage

- Tank Volume Propellant tanks -

inputs

	[%]		NTO	MMH
Total loaded propellant mass [kg]			417	200
Temperature propellant at lift-off [K] (for information only)			*300*	*300*
Density propellants [kg/m³]			1428	875
Total loaded propellant volume: V_load [m³]			**0,292**	**0,228**
	NTO	*MMH*		
Ullage volume needed: V_ull [input: % or m³]			0,100	0,100
			0,100	0,100
Project margin M_proj [%]	5%		0,015	0,011
			0,406	**0,340**
Tank Volume [m³]			**0,7**	

Densities [kg/m³]:		***T_ref [K]***
LOX	1141,97	90,1
LH2	71,13	20,2
NTO	1427,99	300
MMH	875	300

see also separate sheets!

Figure B.2: Tank volume excel sheet

Functional Sizing of an Upper Stage

- Pressure Gas Budget -

inputs

		NTO	MMH	
Engine mass flow: m-dot_engine [kg/s]	16,68			
Propellant flowrate: m-dot [kg/s]		11,3	5,4	
Density propellant: ρ [kg/m³]		1428	875	
Prop. volume flowrate: v-dot [m³/s]		0,008	0,006	
Vol. flowrate pressure gas: v-dot_He [m³/s]		0,008	0,006	
Tank pressure: p [bar]		20	20,0	
Ullage temperature press. gas: T_ull [K]		250	250	
Specific gas const. pressure gas: R_He [J/(mol*K)]		2075,7	2075,8	see separate sheet: He c_v,kappa
Density pressure gas: ρ_He [kg/m³]		3,854	3,854	
Mass-flowrate pressure gas: m-dot_He [kg/s]		0,030	0,024	
Ullage LH2 vol. before reignition: V_ull,LH2,reign [m³]	*p_start [bar]*	0	0,00	
LH2 re-pressurisation from T=22K: p_start [kg]	0	0	0,0	
Propellant to be fed out [kg]	564	383	181	
Pressurization duration [s]		33,9	33,6	
Pressure Gas need [kg]	2	1,0	0,8	
Engine consumption: TPO DSP [kg]		0		
Command system consumption [kg]		0,3		
Project margin: M_proj [kg]	5%	0,1		
Total need pressure gas: m_He,tot [kg]		**2,2**		

Figure B.3: Pressure budget excel sheet

Functional Sizing of an Upper Stage

- Pressure Gas: Compression -

(rough approach)

inputs

	NTO	MMH
Maximum He regulated pressure [bar]	25,00	25,00
Needed Helium mass m_He, tot [bar]	2	
He consumption variations (5% incl. leakage) [kg]	0,11	
Helium need for propellant expulsion only [bar]	30,83	
Helium budget non-consumable [bar]	55	
He consumption deviations [bar]	10,00	
He budget uncertainty & margin [bar]	10,00	
Total press. need for He storage. P_He,stor [bar]	105,83	
He pressure decrease due to temp. compensation [bar]	10,0	
He pressure rise due to temp. compensation [bar]	10,0	
MEOP He vessel [bar]	**125,8**	

Constants for Helium:	
general gas constant R [J*mol]	8,314472
molar mass He [g/mol]	4,003
temperature [K]	250
compressibility factor He []	1,2

Volume He vessel: V_He,vess [L]	**475**

Figure B.4: Helium pressure excel sheet

Appendix C

MATLAB function files

```matlab
%% Atmospheric Model
%This function returns the density depending on the altitude
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%
function rho=dens(H)
if H<11000
    T=(15-0.0065*H)+273;
else
    T=-56.5+273;
end
if H<100000
    rho=1.29*exp(-0.00001*1.29*9.81*H*(14.4+273)/(T));
else
    rho=0;
end
end
```

Figure C.1: Atmospheric model function

```matlab
%% Mass function
%This function calculates the mass of the rocket depending on the time
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%
function m=mass(t)
load('rocketdata.mat')
global tcirc
m0 = [];

%2-stage rocket
if N==2
      T (1)=(tb(1));
      T (2)=(tfair);
      T (3)=(tb(1)+tb(2));

    m0(1)=mpl+mp(1)+mp(2)+ms(1)+ms(2)+mfair;
    m0(2)=mpl+mp(2)+ms(2)+mfair;
    m0(3)=mpl+mp(2)-mdot(2)*(T(2)-T(1))+ms(2);

    if t<0,
        m = m0(1);
        error('Negative time value');
    else
        m = (t<=T(1)).*(m0(1)-mdot(1).*t);
        m = m + (t>T(1)).*(t<=T(2)).*(m0(2)-mdot(2).*(t-T(1)));
        m = m + (t>T(2)).*(t<=T(3)).*(m0(3)-mdot(2).*(t-T(2)));
        m = m + (tb(2)~=0).*(t>T(3)).*mpl + (tb(2)==0).*(t>T(3)).*m0(3);
    end
end
```

Figure C.2: Mass function

```matlab
%% Gravity Model
%This function returns the local gravity g depending on the altitude H
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%
 function g = gravity(H)
g0 = 9.80665;       % Earth mean gravity field [m.s-2]
RE = 6371e3;        % Earth radius [m]
g = g0*(RE/(RE+H))^2;
end
```

Figure C.3: Local gravity function

```matlab
%% Area fuction
%This function calculates the area of the rocket depending on the time
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%
function A=area(t)

load('rocketdata')

%2-stage rocket
if N== 2
    if t<=tb(1)
        A=A(1);
    else
        if t<=tb(2)
            A=A(2);
        else
            A=0;
        end
    end
end

%3-stage rocket
if N== 3
    if t<=tb(1)
        A=A(1);
    else
        if t<=tb(2)
            A=A(2);
        else
            if t<=tb(3)
                A=A(3);
            else
                A=0;
            end

        end
    end
end


end
```

Figure C.4: Reference area function

```
%% Equations
%This function describes the system of equations taken in account before
%and after the pitch program
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%
function dydt = rock1(t,y)
%y(1) Velocity,
%y(2) gamma,
%y(3) Longitude (m),
%y(4) Altitude H (m),

load('rocketdata.mat')
dydt=[];

dydt(1) = Thr(t)/mass(t)-0.5*dens(y(4))*(y(1))^2*area(t)*Cp/mass(t)...
    -(gravity(y(4))-(y(1)*zval(cos(y(2))))^2/(RE+y(4)))*zval(sin(y(2)))...
    -y(1)*zval(cos(y(2)))*y(1)*zval(sin(y(2)))/(RE+y(4))*zval(cos(y(2)));
if (t<tb(1) && dydt(1)<0)
    dydt(1)=0;
end
if y(1)~=0
    dydt(2) = -(gravity(y(4))-(y(1)*zval(cos(y(2))))^2/(RE+y(4)))*zval(cos(y(2)))/y(1)...
        +y(1)*zval(cos(y(2)))*y(1)*zval(sin(y(2)))/(RE+y(4))*zval(sin(y(2)))/y(1);
else
    dydt(2)=0;
end
dydt(3) = y(1)*zval(cos(y(2)));
dydt(4) = y(1)*zval(sin(y(2)));

dydt = dydt';
end
```

Figure C.5: Newton's equations of motion

```
%% Equations
%This function describes the system of equations taken in account during
%the pitch program
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%%
function dydt = rock2(t,y)
%y(1) Velocity,
%y(2) gamma,
%y(3) Longitude (m),
%y(4) Altitude H (m),

load('rocketdata.mat')
global rate
dydt=[];

dydt(1) = Thr(t)/mass(t)-0.5*dens(y(4))*(y(1))^2*A(1)*Cp/mass(t)...
    -(gravity(y(4))-(y(1)*zval(cos(y(2))))^2/(RE+y(4)))*zval(sin(y(2)))...
    -y(1)*zval(cos(y(2)))*y(1)*zval(sin(y(2)))/(RE+y(4))*zval(cos(y(2)));
if (t<tb(1) && dydt(1)<0)
    dydt(1)=0;
end
if y(1)~=0
    dydt(2) = -(gravity(y(4))-(y(1)*zval(cos(y(2))))^2/(RE+y(4)))*zval(cos(y(2)))/y(1)...
        +y(1)*zval(cos(y(2)))*y(1)*zval(sin(y(2)))/(RE+y(4))*zval(sin(y(2)))/y(1) -rate;
else
    dydt(2)=0;
end
dydt(3) = y(1)*zval(cos(y(2)));
dydt(4) = y(1)*zval(sin(y(2)));

dydt = dydt';
end
```

Figure C.6: Newton's equations of motion during the gravity turn

Appendix D

MATLAB Main file

```
%% Rocket trajectory simulation:

%Before running a simulation, verify the RocketData fils in order to check
%if there are the good datas

%%
clear all
RocketData
save('rocketdata')
load('rocketdata.mat')

%global variables
global tcirc
tcirc=5000;

%if simu=0, simulation from 0 to elliptical orbit
%if simu=1, Simulation from 0 to circularization
simu=1;

% gravity turn parameters
tg=[10.0538 17.1012];

%options to control and reduce the errors in the numerical resolution
options = odeset('RelTol', 1e-5, 'AbsTol', 1e-4);
options1 = odeset('RelTol', 1e-5, 'AbsTol', 1e-4, 'events', @stop);

if N==3
    %t defines the different time intervals used in the simulation
    t = [0 tg(1) tg(2) (tb(1)+tb(2)+tb(3))];

    % From lift-off to the first nudge
    [T1,Y1]=ode45(@rock1,[t(1) t(2)],[0 pi/2 0 200],options);

    % Nudge over time tgt(1) to tgt(2) during the first stage
    Y20=Y1(length(Y1(:,1)),:);
    [T2 Y2]=ode45(@rock2,[t(2) t(3)],Y20,options);

    % From end of nudge to eliptical orbit
    Y30=Y2(length(Y2(:,1)),:);
    [T3 Y3]=ode45(@rock1,[t(3) t(4)],Y30,options);
```

Figure D.1: Main file (1/3)

```
%calculations of the elliptical orbit characteristics
V=Y3(end,1);
H=Y3(end,4);
gam=Y3(end,2);
Vx=V*cos(gam);
mu=g0*RE*RE;

E=V*V/2-mu/(RE+H);
M=Vx*(H+RE);
r1=-(mu-sqrt(mu^2+2*E*M^2))/(2*E);
r2=-(mu+sqrt(mu^2+2*E*M^2))/(2*E);
perigee=(r1-RE)/1000
apogee=(r2-RE)/1000
val=[tg(1) tg(2) rate]
%r=(perigee-200)^2+(apogee-700)^2
if apogee>695

    if simu==1

        % on orbit, TE = time when H=700km
        Y40=Y3(length(Y3(:,1)),:);
        [T4 Y4 TE YE IE]=ode45(@rock1,[t(4) 4000],Y40,options1);

        %on orbit to apogee
        Y40=Y3(length(Y3(:,1)),:);
        [T4 Y4]=ode45(@rock1,[t(4) TE],Y40,options);

       % circularisation at TE
        tcirc=TE;
        Y50=Y4(length(Y4(:,1)),:);
        [T5 Y5]=ode45(@rock1,[TE (TE+200)],Y50,options);

        %circular orbit characteristics
        V=Y5(end,1);
        H=Y5(end,4);
        gam=Y5(end,2);
        Vx=V*cos(gam);
        mu=g0*RE*RE;

        E=V*V/2-mu/(RE+H);
        M=Vx*(H+RE);
        r1=-(mu-sqrt(mu^2+2*E*M^2))/(2*E);
        r2=-(mu+sqrt(mu^2+2*E*M^2))/(2*E);
        perigee_C=(r1-RE)/1000
        apogee_C=(r2-RE)/1000
```

Figure D.2: Main file (2/3)

```
        end
    else
        simu=0;

    end

end

 T=[T1;T2;T3];
 Y=[Y1;Y2;Y3];
 thr=[]; %Thrust
 m=[]; % mass
 q=[]; %dynamic pressure
 for i=1:(length(T)-1)
    thr(i+1)=Thr(T(i+1));
    q(i+1)=0.5*dens(Y(i+1,4))*Y(i+1,1)*Y(i+1,1);
    m(i+1)=mass(T(i+1));
    d(i+1)=dens(Y(i+1,4));
    speedx(i+1)=Y(i+1,1)*cos(Y(i+1,2));
    speedy(i+1)=Y(i+1,1)*sin(Y(i+1,2));
end
```

Figure D.3: Main file (3/3)

Appendix E

Fluid schematic

BDVM	burst disk passivation MMH
BDVN	burst disk passivation NTO
BDM	burst disk MMH
BDN	burst disk NTO
BM	branching manifold with filter
CRVH	fill and drain valve Helium
CRVM	fill and drain valve MMH
CRVN	fill and drain valve NTO
CVM	check valve MMH
CVN	check valve NTO
ETF	engine thrust frame
FCV	flow control valve
FDV1	pressurant fill and drain valve
FDV2	propellant fill and drain valve
FDV3	feedline fill and drain valve
FH	helium filter
HVO	Helium Venting Orifice
ISPR	I stage pressure regulator
IISPR	II stage pressure regulator
LVM	Latch valve MMH
LVN	Latch valve NTO
MEOP	maximum expected operating pressure
PO	priming orifice
PT	redundant pressure transducers
PV1	priming pyrovalve
PV2	mainflow pyrovalve
PV3	depletion pressure transducers
PVAM	pressure valve assembly (MMH)
PVAN	pressure valve assembly (NTO)
PVH	pyrovalve Helium
PVM	pyrovalve MMH
PVN	pyrovalve NTO
RDCM	vent valve MMH
RDCN	vent valve NTO
TCA	thrust chamber assembly
TCM	thruster cluster module
TH	helium tank
TM	MMH tank

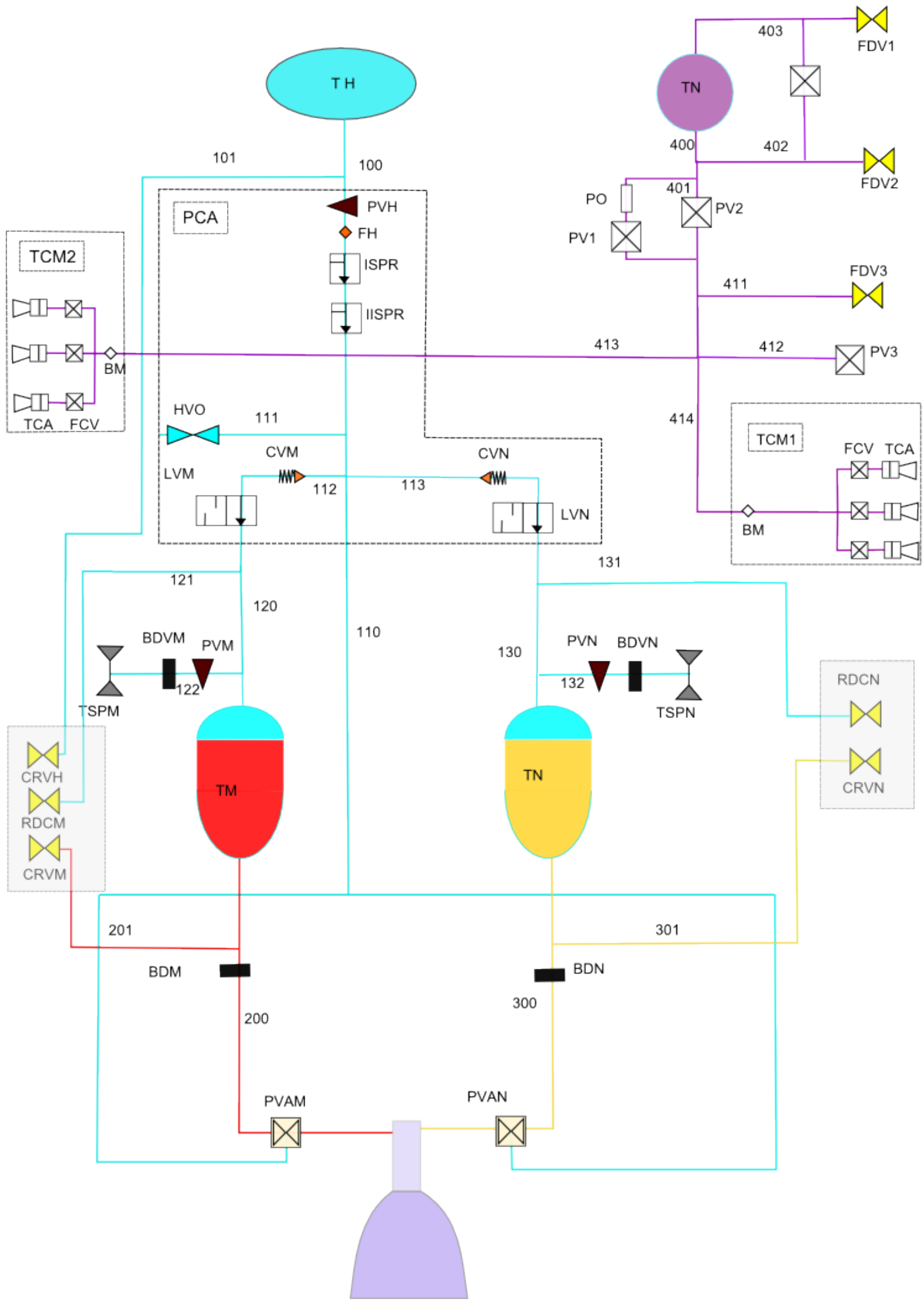


Figure E.1: Upper stage fluid schematic

Appendix F

Technical drawings

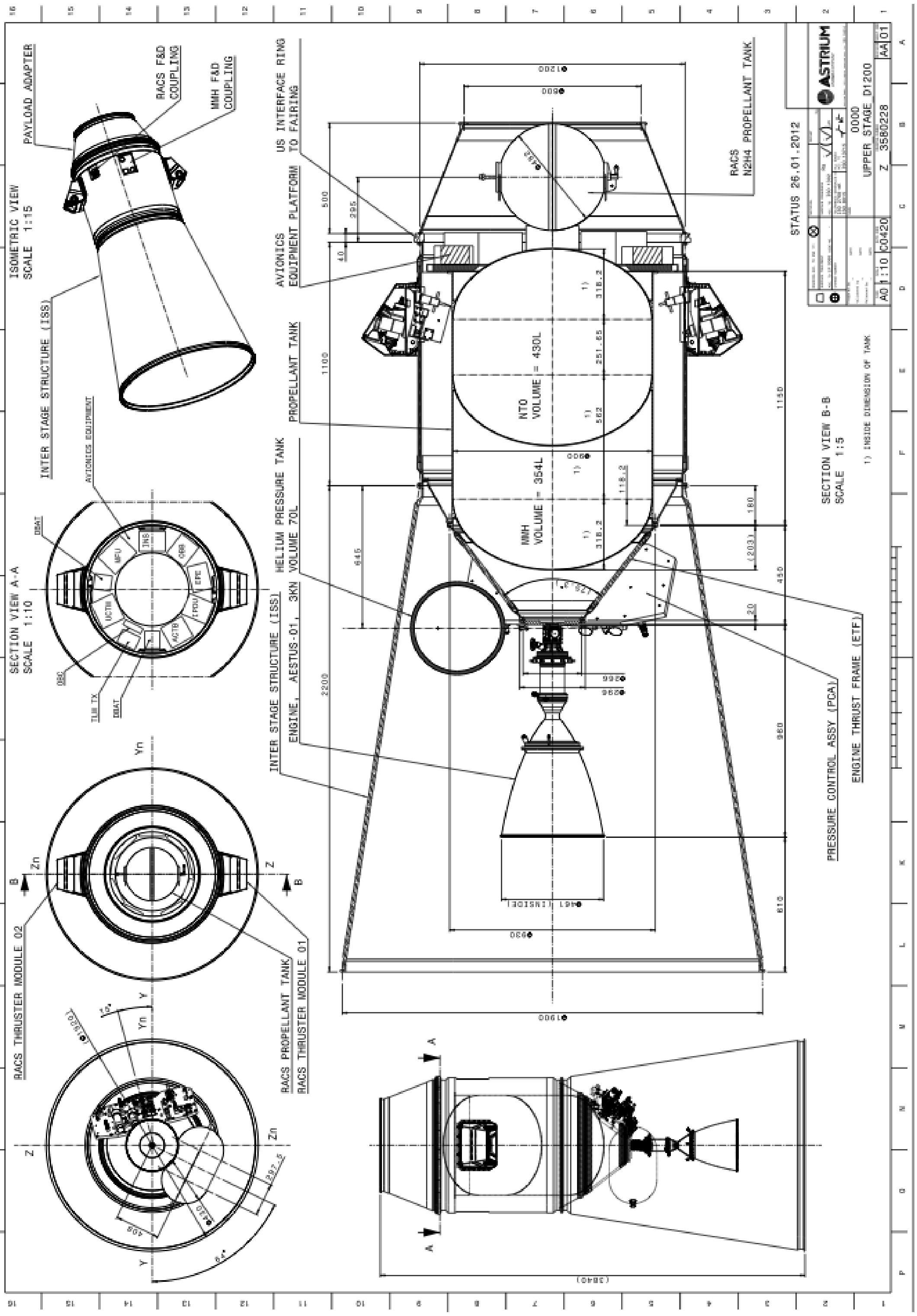


Figure F.1: Upper stage technical drawing, 3-stage launcher

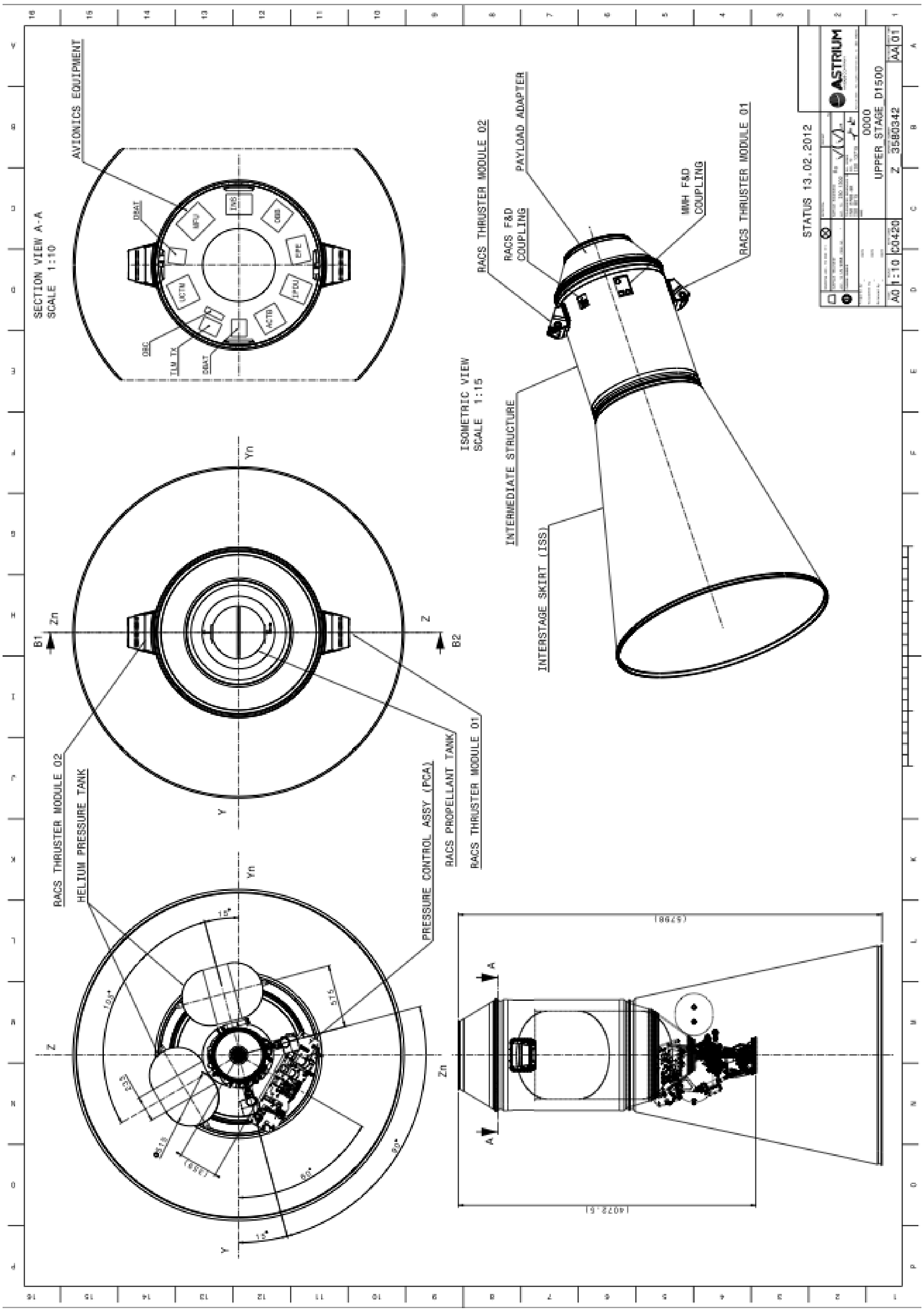


Figure F.2: Upper stage technical drawing 1, 2-stage launcher

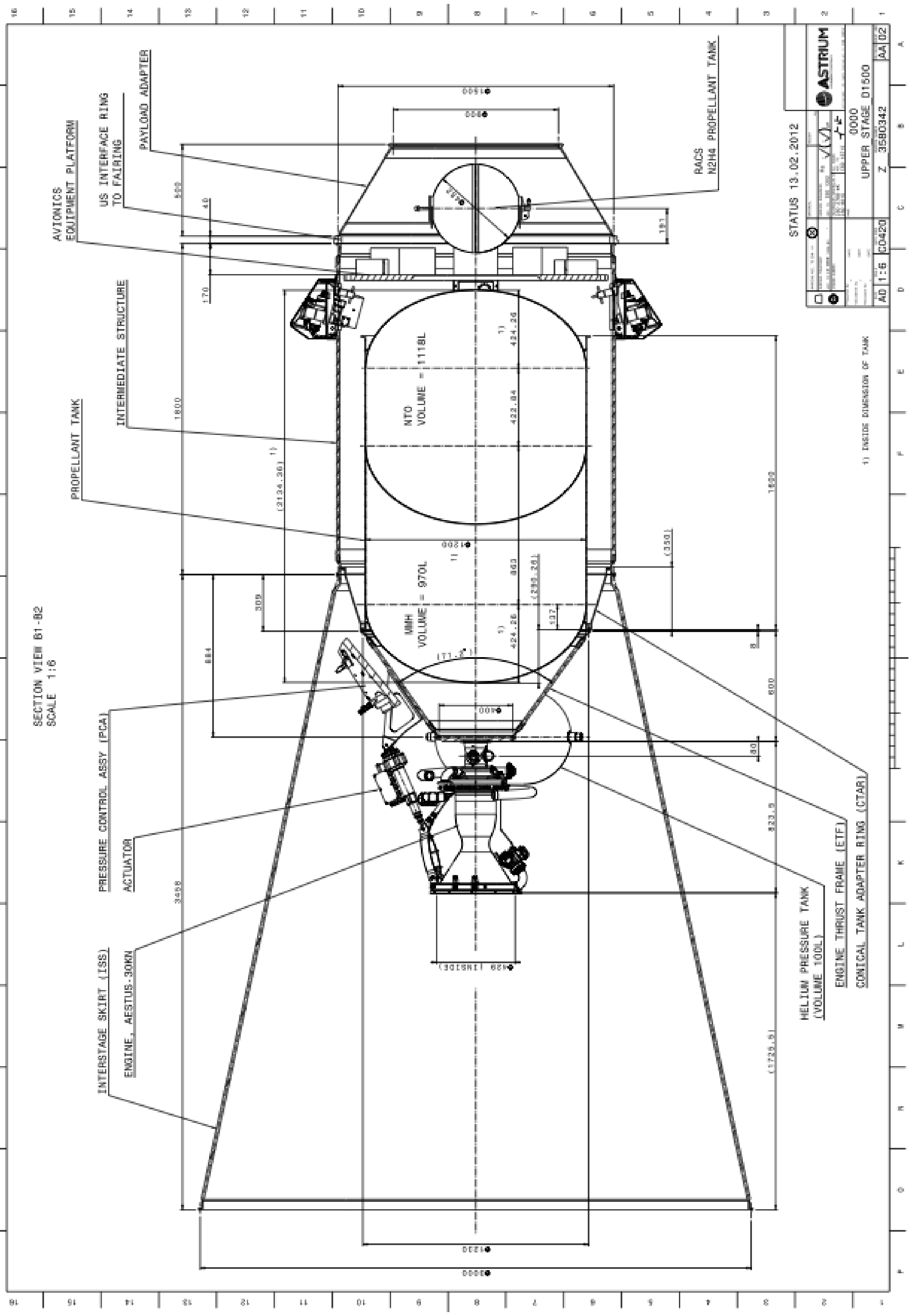


Figure F.3: Upper stage technical drawing 2, 2-stage launcher