SOUNDING ROCKETS
Analysis, simulation and optimization of a solid propellant motor

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ABSTRACT

The aim of the present paper is to expose the performance simulation of a small sounding rocket that operates with a solid propellant motor. In order to contribute in the understanding of solid propellant rocket engines, this work analyses the basic functioning of this type of motor and simulates its performance in the experimental rocketry field.
In a solid propellant rocket, the combustion is clearly determined by the grain shape, which establishes the burning surface as a function of time. It affects directly the chamber pressure and, along with burn rate, it determines thrust. So, as well as propellant characteristics, the burning surface is an essential parameter and the engineer can modify it in order to obtain the desired performance. The simplicity of the solid motor and the large number of possible grain burning shapes make solid propellants an interesting option for a rocket engine design.
Therefore, the focus of this study is to build a complete model of solid combustion, with a specific burning shape, that simulates the rocket performance. The motor dimensions have been based on an *Aerotech* engine used in experimental rocketry. It has served as a model for the first calculations and its real performance has proved to be a good comparison tool for the results obtained. The simulation has been carried out in *Hopsan*, a multi-domain software developed at the Linköping University.
Once the simulation is completely built, an optimization of the engine has been done in order to improve the performance and to increase the altitude reached by the rocket.
Finally, the results obtained with the current burning shape are compared to the ones that would have been gathered with another grain pattern. Hence, the comparison of different burning shapes gives an idea of how performance parameters such as thrust, chamber pressure or burn rate change when varying the way of combustion.

KEYWORDS: *solid propellant, grain shape, sounding rocket*

NOMENCLATURE

A - Cone area
A_e - Nozzle exit area
A_t - Nozzle throat area
a - Burn rate coefficient
c^* - Characteristic velocity
c_v - Heat capacity
D - Grain external diameter
D_e - Nozzle exit diameter
D_t - Nozzle throat diameter
d - Grain internal diameter
F - Force in the vessel
f_i - Fraction of fuel component
g - Gravity field constant
h - Theoretical altitude reached
L - Grain length
M' - Molecular weight
M_c - Mass of the case
M_e - Empty mass of the rocket
M_{ed} - Exit design Mach value
M_i - Lift-off mass
M_m - Mass of the motor
M_p - Mass of the propellant

M_{pl} - Payload mass
M_v - Dry mass of vehicle
n - Burn rate exponent
P_c - Pressure in the chamber
r' - Burn rate
r_1 - Cone upper radius
r_2 - Cone lower radius
s - Cone generator line
T_c - Adiabatic flame temperature
t - Case thickness
V - Rocket velocity
Θ - Burning angle of the grain
ε - Expansion ratio
ρ - Fuel density
ρ_i - Density of fuel component
σ - Normal stress in the vessel
γ - Ratio of specific heat

1 INTRODUCTION

A sounding rocket is a small vehicle used in scientific and research missions. It takes its name from the nautical term "sound", which means to take measurements. This kind of vehicles are designed to perform scientific testing and experiments in a range of altitudes from about 50 to 1500 kilometres so, unlike general rockets, they do not enter into orbit. Their purpose is to reach the desired altitude, where the carried instruments can perform the experiment, and then come back to ground. [1]
The sounding rocket design usually consists of a solid propellant motor and a payload that includes all the instruments needed for the research. The payload is located on top of the rocket, behind the nose cone, and it can include instruments for the experiment, telemetry equipment -that sends all the information to the ground station- and altitude control. More advanced sounding rockets can also operate with liquid propellant, but this work will focus on the ones that burn solid.
Their applications in science are extensive and diverse but they are mostly used for scientific research in the upper layers of the atmosphere and outer space. They are also widely used in experiments under microgravity conditions, to investigate combustion behaviours or to test solidification of metals in low-gravity situations. [2]

1.1 State of the Art

Due to the characteristics of sounding rockets and their requirements, it is possible to simulate their design and performance using software for recreational rocketry. The most commonly used are *OpenRocket*, *Rasaero*, *Rocksim* and *SpaceCAD*. The first two are free software with user-friendly interface and include templates of the different parts of the rocket; hence, they are a good choice for beginners. *Rocksim* and *SpaceCAD* are proprietary software, being both more complete tools able to design complex systems. There is a *Rocksim* Pro version, which includes additional functionality to simulate trajectories and splash patterns.

2 THEORETICAL ASSUMPTIONS

2.1 Basic functioning of a solid propellant motor

A solid propellant rocket is a simple propulsion system that consists of a high pressure vessel that contains all the solid components needed. The fuel and oxidizer are intimately mixed together and cast into a solid mass, called grain. The propellant grain usually has a hole down the centre of the chamber, which is called perforation, and may be shaped in various ways.
The functioning starts with an ignition system; whose firing causes the beginning of a chemical reaction over the solid surface in the perforation. Once ignited, a simple solid rocket motor cannot be shut off, as it contains all the propellants needed for combustion all together in the chamber where they are burned. After the ignition, the propellant grain burns on the entire inner surface of the perforation, until the end of the propellant. The heated gases generated during the solid combustion pressurize the inside of the chamber and are finally expelled through a nozzle, which accelerates them producing the reaction force needed to move, known as thrust. [3]
If the geometrical variations are smooth, the process reaches an almost stationary state in which fluid variables remain stable in time inside the chamber. The burning surface evolution determines the propellant mass flow consumed and, therefore, the chamber pressure. Finally, the propellant consumption results in a decrease of the mass flow at the entrance and the chamber discharge in a non-stationary process.

2.2 Components and systems

The architecture of a solid propellant rocket is quite simple and has just a few components, depending on the application. In simple designs, the rocket is able to operate safely with just a few components, shown in Fig.1.

- *Propellant grain*. It contains granular fuel and an oxidizer mixed together in a matrix binder and their composition can vary depending on the application. However, typical fuels are metals as aluminium and oxidizers are usually ammonium perchlorate or ammonium nitrate.
- *Igniter*. The igniter is the element in charge of raising the grain surface temperature so it increases the chamber pressure to self-sustaining levels. The system must be designed such that unwanted ignition cannot occur at any failure.
- *Insulation*. It is the element that protects the case from hot combustion products, avoiding over-heating at the end of the grains. It is built in ceramics or plastic materials, so it does not react with the propellant components.
- *Case*. It is a cylindrical container capable of resisting high pressures and temperatures whose function is to embody the combustion chamber. The dimensioning of this element is of great importance as it holds the grain and has to withstand the pressure achieved during the combustion. It can be constructed in metal, like steel, aluminium or titanium alloy, or in a fibre reinforced compound material.
- *Cylinder core*. It consists of a hole typically located in the centre of the chamber and it is the place where the ignition starts. The most typical configuration is a unique hole that can be shaped in various ways but for some grain patterns, there can be multiple holes located in different parts of the chamber.
- *Liner*. Its purpose is to inhibit grain burning and insulate the case when the flame front arrives. It is usually a propellant binder and can be mixed with the propellant. The most common binder used in solid propellant rockets is polybutadiene.
- *Nozzle*. It is the most important element in the motor, as it expands and accelerates the heated gases that produce thrust. A correct dimensioning of the nozzle is essential in a preliminary design of a rocket, as it will determine the performance. The general type of nozzle used in rockets is a converging-diverging Laval nozzle in which the flow is compressed in a convergent section until it reaches the throat, when it is expanded in the divergent section. The nozzle must be built in a material that endures the high temperatures reached and a usual construction is graphite epoxy with a carbon throat.

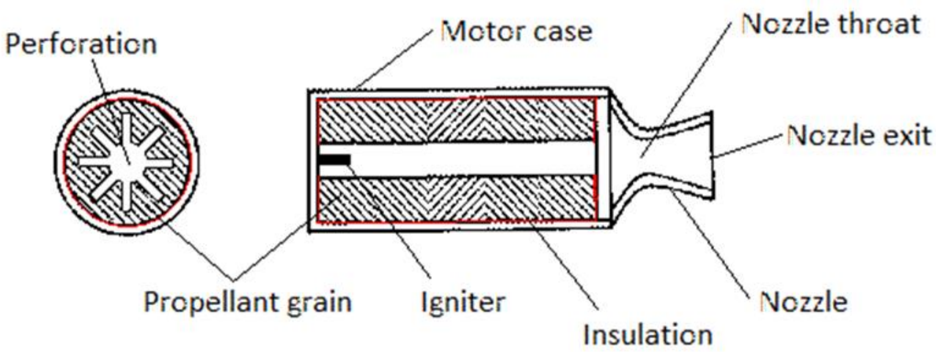

Figure 1: Solid propellant basic structure. Modified from [4]

In general terms, the parameter that determines the efficiency of the expansion in the nozzle is the expansion ratio. It links the exit and the throat area of the nozzle and it has a constant value in fixed geometry nozzles.

$$\varepsilon = \frac{A_e}{A_t} \tag{1}$$

When the nozzle flow cannot be variable, the optimum expansion ratio of a rocket nozzle which operates inside the atmosphere is the one that obtains an exit pressure equal to the atmospheric pressure, which means that the jet stream is adapted and the nozzle is critical for a determined altitude.

2.3 Characteristics of solid propellants

Solid propellants possess unique properties, such as the capability to self-sustain the burning process, generate thermal energy and produce propulsive mass at the same time. Some types of solid propellants are even able to initiate burning without any power input. That eliminates the necessity of having an igniter, which means less rocket weight. On the contrary, it makes the rocket more likely to unwanted ignitions that can be dangerous.
Among their advantages, there is the fact of being much easier to store and handle than liquid propellants and their density makes them a good choice when compact size is needed. Their simplicity is also a determining factor whenever large amounts of thrust are needed at relatively low cost.
However, they have lower specific impulses, so their efficiency is not as good as expected. Another clear disadvantage is their intolerance to cracks and voids, which would increase the burning surface area if existing. That would increase local temperature, system pressure and heat flux to the surface.
In conclusion, solid systems are relatively inflexible, but they give good performance at reduced complexity. [5]

2.4 Burning shape and combustion

In a mission that involves a solid propellant rocket great burning areas are needed in order to achieve enough pressure levels in the chamber. The most efficient shape for that purpose is a cylindrical chamber with a perforation in the centre, so the grain burns in a lateral combustion.
The initial shape of the core or perforation is what will determine the burning shape and it can have a wide variety of sections such as circular, star, cross, wagon-wheel, dendrite or anchor shape. The core is where the ignition starts and the burning area evolution will rule chamber pressure and thrust, which may be progressive, neutral or regressive depending upon whether it increases, remains constant or decreases in time.
A grain that has an increasing burning surface with time is called progressive. The main disadvantage of this kind of combustion is that the mass decreases as the thrust increases so there is a sharp increase of the spacecraft loads with time, which conducts to an inefficient structure. If the grain is inhibited at the ends of the chamber and it burns from the sides, the burning will be regressive. The last and most common burning design for spacecraft is the neutral one as it generates a constant chamber pressure and thrust. That is reflected in greater efficiency in delivery of total impulse, as the nozzle operates more efficiently when the chamber pressure is constant. [6]

2.5 Burn rate

Burn rate is the recession velocity of the solid propellant, which burns by layers with a flame front in a direction perpendicular to the burning surface. If the propellant recession is uniform, the layers also have a uniform thickness and burn rate can be estimated by a simple equation. The most significant factors that influence propellant burn rate are the combustion chamber pressure, the initial temperature of the reactants and the velocity of the combustion gases. [3]
Vielle's Law establishes the dependence of burn rate with chamber pressure, affected by a burn rate coefficient and a pressure exponent characteristic of every propellant, shown in Eq. 2. These two parameters must be determined empirically for a particular propellant and cannot be theoretically predicted.

$$r' = aP_c^{\ n} \qquad (2)$$

Burn rate is quite sensitive to the value of the pressure exponent, producing large changes in burn rate with relatively small changes in chamber pressure. A high pressure exponent is undesirable because it would produce high burn rates that disgorge in greater chamber pressures that are not probably tolerated by the motor. [4]
On the other hand, burn rate coefficient is function of the initial temperature of the propellant. If this parameter is increased, so does burn rate. The propellant sensitivity to the initial temperature must be taken into account, especially when the rocket is launched in extreme conditions.
It is of great importance to avoid erosive burning to have a uniform burn rate. In the presence of a cross flow, burn rate increases, producing a decrease in the characteristic thickness of the propellant.

This induces a modification in the burn rate equation that will depend on the Mach number of the cross flow. It will modify the thrust curve and the performance given will differ from the expected.

3 ROCKET ANALYSIS

3.1 Propellant specifications

The propellant selected for the simulation is the ammonium perchlorate composite (APCP), a compound propellant that has both fuel and oxidizer mixed with a binder, usually of a rubbery nature. The propellant consists of ammonium perchlorate as the oxidant and hydroxyl-terminated polybutadiene (HTPB) as an elastomer binder. It usually includes aluminium, which along with the binder, serves as the fuel. Normally, it is used in aerospace missions because it is easy to handle and store and has good propulsive characteristics.
The composition of the compound varies depending on the application, burn characteristics required and nozzle constraints. For this purpose, the composition has been established in AP 80%, Al 2%, HTPB 18%; typical proportion in high-power rocketry. General properties are shown in Table 1.

Table 1: General properties of propellant [5]

Parameter	Symbol	Value
Ratio of specific heats	γ	1.25
Molecular weight	M'	23.7 g/mol
Heat capacity	C_v	1534 J/kg·K
Adiabatic flame temperature	T_c	2780 K
Characteristic velocity	c^*	1470 m/s
Burn rate coefficient	a	0.123
Burn rate exponent	n	0.287

The average density of the compound has been estimated knowing the density of each product and their proportion in the mix, shown in Table 2.

Table 2: Density and mass fractions of propellant components

Product	Percentage (%)	Density (kg/m³)
Ammonium perchlorate (AP)	80	1950
Aluminium (Al)	2	2700
HTPB	18	930

$$\rho = \frac{1}{\frac{f_{AP}}{\rho_{AP}}+\frac{f_{Al}}{\rho_{Al}}+\frac{f_{HTPB}}{\rho_{HTPB}}} \quad (3)$$

"Eq. 3" gives a density value of 1636 kg/m³.

3.2 Motor dimensions

The input parameters for the grain and motor dimensions are based on the *Aerotech* reloadable motor M2000R-P and gathered in Table 3.

Table 3: Grain and motor dimensions [7]

Parameter	Symbol	Value (mm)
Grain external diameter	D	85.6
Grain internal diameter	d	28.575
Grain length	L	609.6
Nozzle throat diameter	D_t	21.44
Nozzle exit diameter	D_e	44.45

3.3 Burning area analysis

In order to study all the combustion types described in the previous section, we have established a kind of burning surface in which the propellant burns in a progressive, neutral and regressive way. It is shown in Fig. 2.

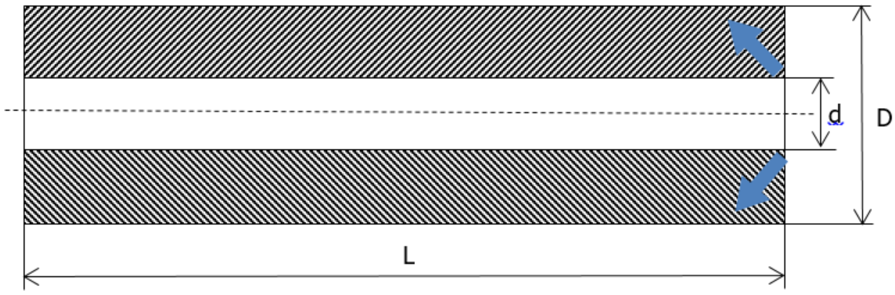

Figure 2: Burning surface configuration

As the original grain has a perforation in the centre, we will consider an ignition that starts in the inside corners of the core and the grain burns with an angle that can be changed as wanted. Then the burning surface of the combustion will be determined by the lateral area of a truncated cone. This area can be calculated as the semi sum of the perimeters of the two bases multiplied by the generator line as in Eq. 4.

$$A = \frac{2\pi r_1 + 2\pi r_2}{2} \cdot s = \pi(r_1 + r_2) \cdot s \qquad (4)$$

3.4 Case dimensioning

When a cylinder is under a uniform pressure, normal stresses appear in two directions. The ones that take action in the axial axis of the cylinder are called longitudinal stresses and the ones acting in the perpendicular direction are tangential stresses. The first ones are less important as they will have a lower value at the same chamber pressure. So if the case withstands the tangential stress it will do the same with the longitudinal one. As general assumptions, tensile and compressive stresses are supposed to be constant and uniformly distributed in the thickness of the wall. It is also assumed that all loads, stresses and strains are symmetrical about the cylinder's axis. [8]
To dimension the case, it is necessary to establish the thickness of the walls as it determines the behaviour of the container to stress. In thin-walled cylinders the stress distribution is uniform while the same does not happen in heavy-walled cylinders. A cylinder is considered of thin walls when the ratio between the external diameter and the thickness is greater than 10. On the contrary, the heavy-walled cylinder has a ratio smaller than that value. Our calculations will focus on a thin-walled cylinder as the chamber case and the thickness is set to 4 mm, as it is a typical value in similar designs. It will be supposed that the case is built into an aluminium alloy, which is capable to hold a maximum stress of 70 MPa. [9]
The vessel is subjected to a force that is function of the chamber pressure. It is applied along the cylinder and its maximum value is given by the external diameter of the grain. As the pressure is uniform along the chamber, the force that holds the case is equal to the previous one and depends on the normal stress (Eq. 5).

$$F = D \cdot L \cdot p = 2 \cdot t \cdot L \cdot \sigma \qquad (5)$$

Combining the equations, we obtain the maximum chamber pressure bearable by the case as a function of the case dimensions and the maximum stress (Eq. 6).

$$p_{max} = \frac{2t}{D}\sigma_{max} \qquad (6)$$

The maximum pressure the case can withstand has a value of 6.54 MPa, which has to be kept in mind as a restriction when optimizing the system.

3.5 Simulation model

The entire model has been built in *Hopsan*, a simulation program that allows the creation of different components and their subsequent connection to the system, so the simulation can be performed. It is a free multi-domain system simulation tool developed at the division of fluid and mechatronic systems at Linköping University. *Hopsan* is an open source software, being the source code available at the University's website. [9]

The model consists of the following subsystems integrated together as it is shown in Fig. 3.

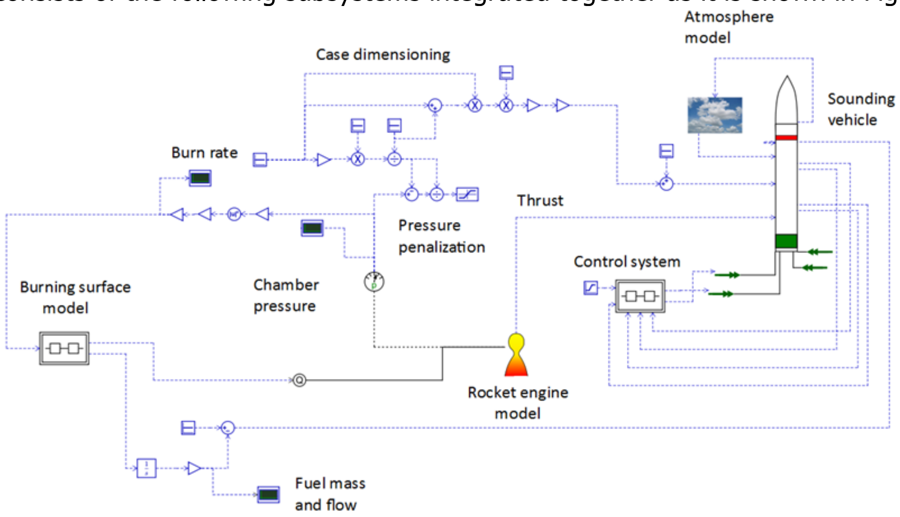

Figure 3: General model built in *Hopsan*

The burning surface model shown in Fig. 4 is a subsystem that contains all the necessary elements for the correct simulation of the burning evolution. It has been constructed on the basis of the burning area analysis that has been done before and using available components in *Hopsan*.

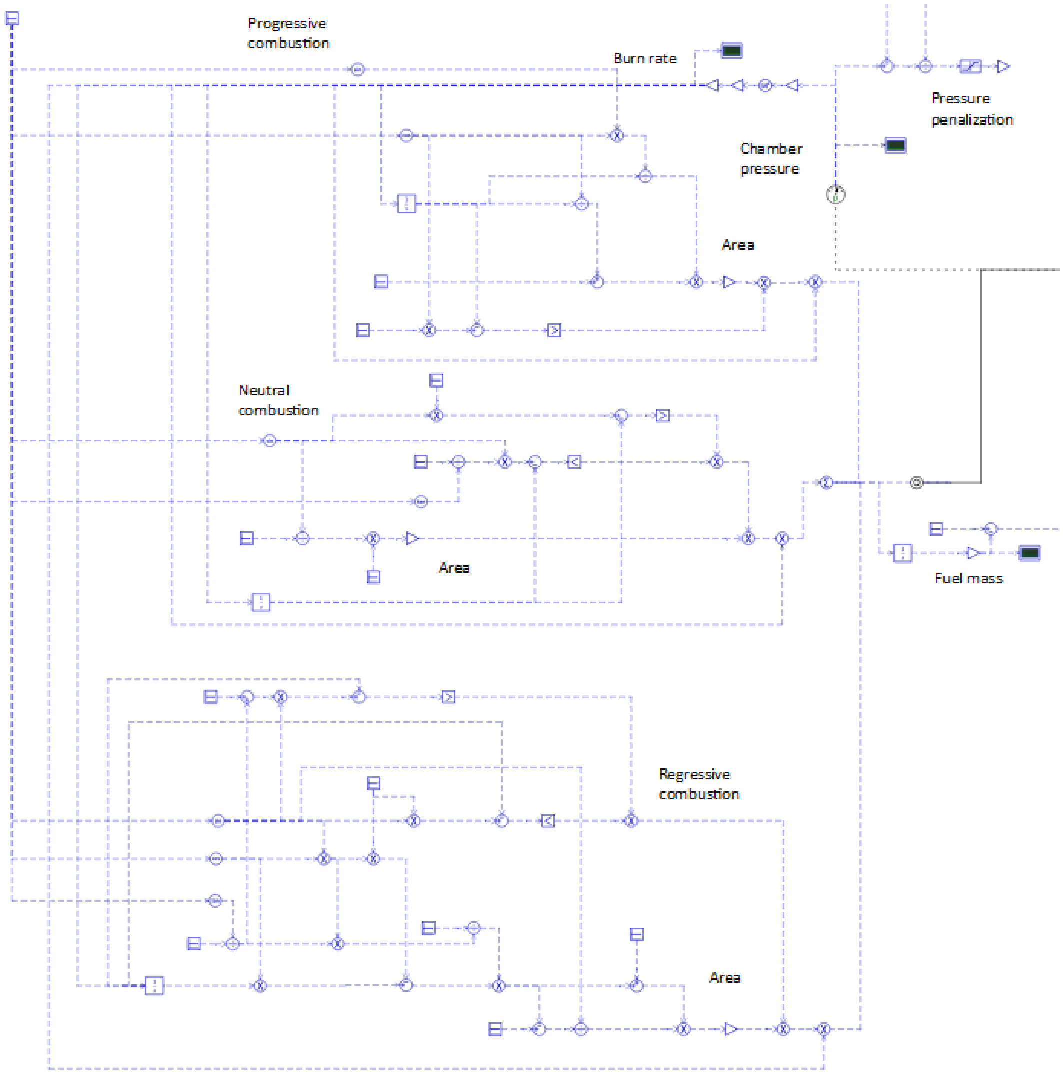

Figure 4: Model of the solid combustion

4 RESULTS

The combustion parameters obtained after the simulation are shown next and the results will be discussed and compared to the ones expected in similar models, in order to give an overall understanding of the system.
The initial angle for the conical combustion, established to 0.14 rad, induces a maximum chamber pressure of 1.95 MPa and a burn rate of 0.018 m/s, both shown in Fig. 5 and Fig. 6. These values are quite similar to the ones expected in a solid propellant as APCP.

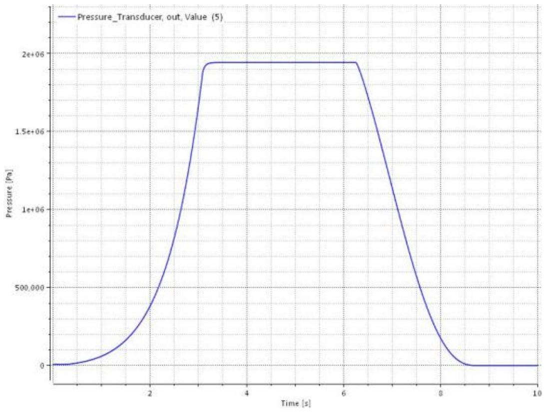

Figure 5: Simulation results for chamber pressure

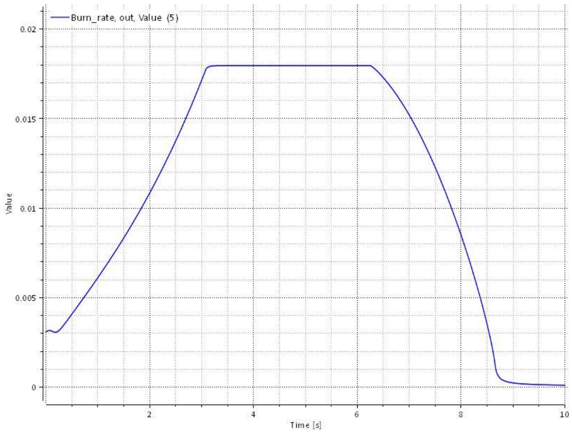

Figure 6: Simulation results for burn rate

The flow mass analysis shows in Fig.7 the evolution of the propellant burning and establishes that the propellant mass needed for the combustion is about 5.1 kg.

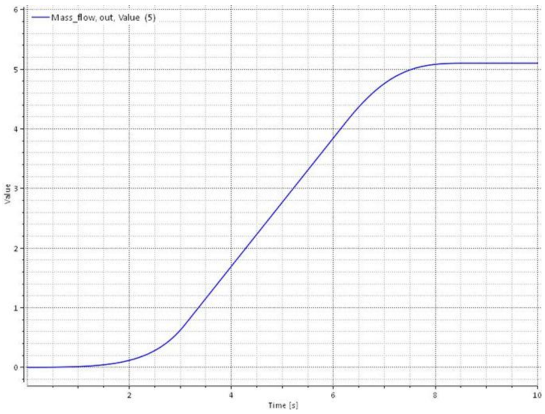

Figure 7: Simulation results for mass flow

4.1 Inventory of weights

When defining a mission, the inventory of weights is analysed in order to determine the minimum weights of the vehicle [3]. For a solid propellant rocket, Eq. 7 gives the basic calculation.

$$M_i = M_v + M_{pl} + M_c + M_p \tag{7}$$

The values of the specific weights that define the rocket are summarized in Table 4.

Table 4: Inventory of weights

Weight parameter	Symbol	Value (kg)
Lift-off mass	M_i	20.9
Vehicle mass	M_v	12
Propellant mass	M_p	5.1
Case mass	M_c	1.8
Payload mass	M_{pl}	2

4.2 Performance results

The flight simulation presents a maximum altitude of about 1.8 kilometres and a maximum thrust of 971 N. The net thrust, and therefore the trajectory, has a delay in its performance. It is because the rocket does not have enough force to launch until the chamber reaches a determined pressure. Therefore, the sounding rocket design must include a launch pad that holds the vehicle until it has enough force to take off. Both thrust and trajectory are gathered in Fig. 8 and Fig. 9.

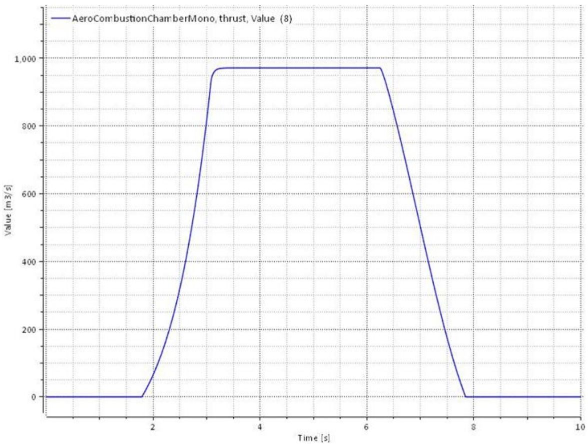

Figure 8: Simulation results for thrust

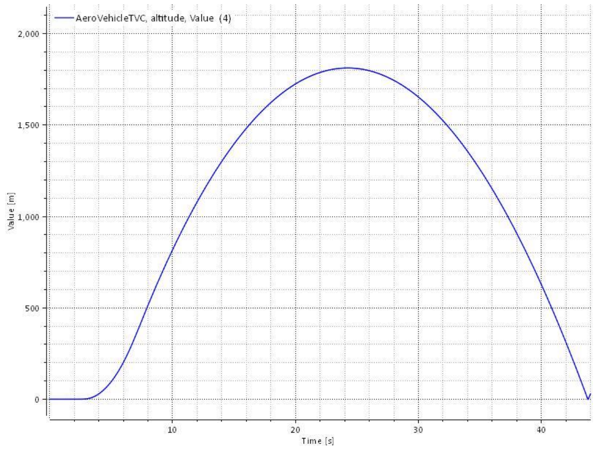

Figure 9: Simulation results for altitude as a function of time

4.3 Theoretical altitude

Ideally, in the absence of air drag, the theoretical altitude a propellant mass is able to reach can be estimated with the conservation of energy equation. Under the presence of just conservative forces, the entire energy of the propellant would be used to raise the vehicle and all the kinetic energy of the propellant could be changed into potential energy (Eq. 8).

$$\frac{1}{2}M_pV^2 = M_egh \qquad (8)$$

A vehicle of 12 kg and a payload of 2 kg combined with the other parameters give a theoretical altitude of 13 km. This value compared with the one obtained in the simulation shows how the presence of air drag and the inefficiency of the energy conversion lead to a lower altitude.

4.4 Optimization

Once the simulation is completely built and the results obtained have been discussed, an optimization is done in order to improve the performance. The purpose is to maximize the model's altitude by the optimization of the design values. Furthermore, when optimizing, it must be considered that the optimized chamber pressure cannot exceed the maximum pressure calculated before, so that penalization has been included in the model.
The parameters chosen to optimize the model are the exit design Mach value for the nozzle, Med, the nozzle throat area, At, and the burning angle of the grain, θ. The initial and the optimized values of the parameters are shown in Table 5.

Table 5: Optimized parameters for the simulation

Parameter	Initial value	Optimized value
Θ	0.14	0.131
M_{ed}	2.72	3.07
A_t (cm^2)	3.61	1.67

The resulting performance shows how the optimization tends to increase the pressure to the maximum bearable by shortening burning time. The increasing of the exit design Mach and the decreasing of the nozzle throat area leads to a higher value of the expansion ratio, which changes from 4.3 to 6.7. A greater value of this parameter means a more efficient nozzle expansion.

The combustion now reaches a chamber pressure of 6.5 MPa, quite close to the maximum, and a burn rate of 0.026 m/s, shown in Fig. 10 and Fig 11. The burning time is reduced to about 3 seconds and the same happens to the ignition delay, as the grain now burns faster.

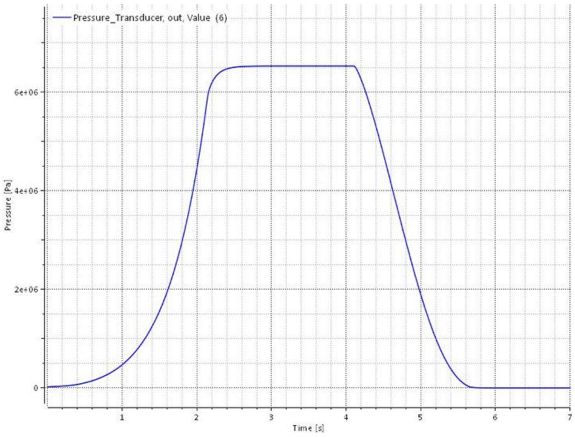

Figure 10: Optimization results for chamber pressure

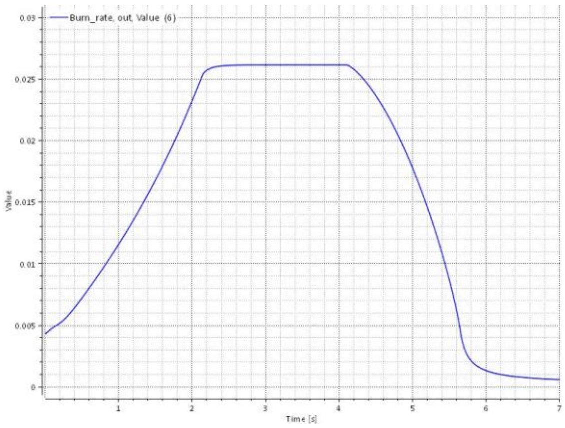

Figure 11: Optimization results for burn rate

For the optimized parameters, the results for the simulation show a maximum altitude of almost 2.84 kilometres with a peak thrust of 1714 N, as seen in Fig. 12 and Fig. 13.

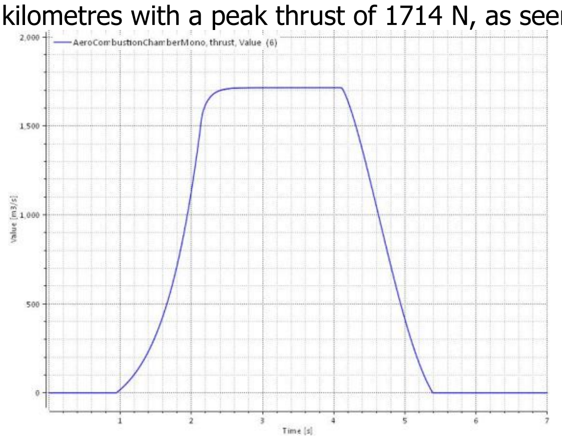

Figure 12: Optimization results for thrust

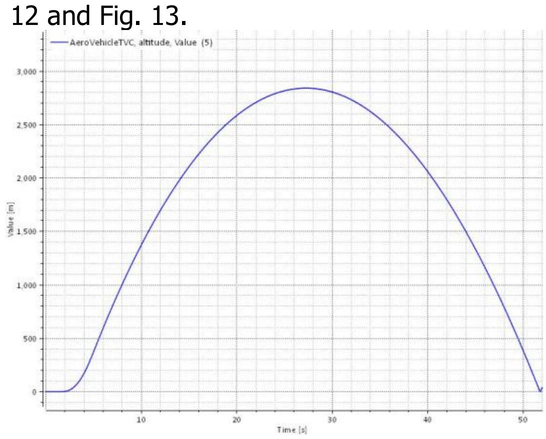

Figure 13: Optimization results for altitude as a function of time

4.5 Case thickness

The previous results show the optimization of the rocket when the case thickness has been established to 4 mm. However, if the case is re-dimensioned, it is possible to compare the different performance obtained when optimizing (Table 6).
If the case thickness is increased, it induces an increasing of the maximum chamber pressure bearable, which is reflected in a greater thrust. Nevertheless, more thickness means also more weight in the rocket, so it penalizes the altitude. As it can be seen in the table, the pressure increase does not offset the weight penalization and the reached altitude with a case of 6 mm in thickness is a bit lower than in the initial configuration. The same happens with a smaller thickness. The case would weigh less now, but the chamber pressure, and consequently thrust, would be also lower so the maximum altitude would decrease.

Table 6: Parameters at different thicknesses

Parameter	Thickness (mm)		
	2	4	6
ϴ	0.119	0.131	0.118
M_{ed}	2.69	3.07	3.33
A_t (cm^2)	3.03	1.67	1.39
P_{max} (MPa)	3.27	6.54	9.76
F_{max} (N)	1404	1714	2212
h_{max} (m)	2570	2840	2827

5 CONCLUSIONS

The model built in *Hopsan* is able to simulate the combustion selected for the design and the performance obtained is similar to the one expected.

As it was said, grain pattern is an essential parameter when defining a solid combustion and it will determine rocket performance. Knowing the shape of the perforation and where the ignition starts, it is possible to simulate almost every combustion in *Hopsan* software.

Chamber pressure and burn rate are also very important parameters and they depend on the propellant chosen for the system and chamber dimensions. So, a good selection of these parameters is a determining task in a rocket design.

Finally, the simulation carried out concludes in the fast that the case thickness has also an important influence in the rocket performance. There is a value that maximizes the altitude reached by the rocket during its flight. The optimized value is the one that gets a compromise solution between a high chamber pressure and a lighter system, so the pressure offsets the weight gained.

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