Escola d'Enginyeria de Telecomunicació i Aeroespacial de Castelldefels
UNIVERSITAT POLITÈCNICA DE CATALUNYA

TREBALL DE FI DE CARRERA

TÍTOL DEL TFC: Design and implementation of a second stage nozzle for a low cost mini-launcher

TITULACIÓ: Enginyeria Tècnica Aeronàutica, especialitat Aeronavegació

AUTORS: Ernest Arias Novo

DIRECTOR: Joshua Tristancho Martínez

DATA: 25 de Marzo de 2011

Título: Design and implementation of a second stage nozzle for a low cost mini-launcher
Autores: Ernest Arias Novo
Director: Joshua Tristancho Martínez
Fecha: 25 de Marzo de 2011

Resumen

Este estudio está dirigido al diseño y construcción de una segunda etapa de combustible sólido basado en APCP (*Ammonium Perchlorate Composite Propellant*) como parte de una beca de la EETAC.

Usaremos el simulador Moon2.0 para diseñar los parámetros básicos de la tobera. Se deberán hacer una serie de pruebas de materiales para validar la tecnología basada en materiales compuestos como Carbon FOAM, graphite y cerámicos.

En este trabajo final de carrera (TFC) intentamos comprobar si este diseño es útil para un proyecto de una lanzadera de bajo coste como es el Wiki-Launcher de menos de 100 kg de masa que pone en órbita LEO (Low Earth Orbit) un femto-satélite de menos de 100 gramos de masa

Palabras clave: Tobera, Low cost, Mini-launcher, Femto-satellite, Carbon FOAM, Graphite, Ceramic

Title: Design and implementation of a second stage nozzle for a low cost mini-launcher
Authors: Ernest Arias Novo
Director: Joshua Tristancho Martínez
Date: March, 25th 2011

Overview

This study is addressed to design and build a nozzle for a second stage of a solid propellant APCP (Ammonium Perchlorate Composite Propellant) based as a part of an EETAC university grand.

We will use the simulator Moon2.0 to design basics nozzle parameters. A series of material tests should be done in order to validate the technology based on composite materials such as Carbon FOAM, graphite and ceramics.

In this final bachelor work, we try to check if this design is useful for a project like the low cost Wiki-Launcher less than 100 kg mass that puts in LEO (Low Earth Orbit) orbit a femto-satellite of less than 100 grams mass.

Keywords: Nozzle, Low cost, Mini-launcher, Femto-satellite, Carbon FOAM, Graphite, Ceramic

INDEX

LIST OF FIGURES

LIST OF TABLES

INTRODUCTION

We define the space payload paradigm as the engineering process of designing a space mission around its payload and not around space industry [1]

Nowadays to put a satellite on orbit is really difficult. The use of a big reusable launcher makes the cost of the “renting” very expensive.

Following the space payload paradigm, in this work we present the idea of to use a launcher especially designed for the satellite. This launcher is controlled by the satellite and it has a scalable design. Because this, the reduction in size makes very significant the weight and the complexity of the subsystems.

A mini-launcher is a small launcher that can put on orbit femto and pico satellites on orbit. The main idea is to reduce the costs and avoid the space monopoly established.

The second stage design will follow the same path. The objective is to develop a reliable, safe and low cost system that will meet the requirements to put on LEO orbit the Wikisat satellite. To achieve this purpose we will study some technologies and materials that can provide us a good performance with a low cost. The second stage is used to reach the orbital speed when the launcher is at the apogee. It has a very short burn time and very high acceleration.

In chapter 1 I will introduce a research of technologies for low cost nozzles. This includes the high performance solid propellant concept with some examples and an explanation of advanced composite materials and its properties.

Chapter 2 talks about Wikisat space program: the relation between Wikisat and the N-prize, the presentation of our femto-satellite and mini-launcher and the subsystems of both.

In chapter 3 I will explain all the second stage design. I will begin with the propellant study, very important for the development of all the second stage, followed by the nozzle study and finally the structure and igniter study. We will see how all the components have their main task in all the second stage development.

In chapter 4 I will talk about the fabrication of all the components studied in chapter 3 doing more emphasis on the nozzle construction.

In chapter 5 there are a series of validation tests that their main objective is to check if all what we will fabricate is useful for our project. The purpose is to validate all the second stage elements.

The last chapter shows us briefly: which is the environmental impact of our second stage launching, the waste generated during the development process and the conclusions obtained of all the work done.

We used the Wikipedia as an encyclopedia but we checked its correctness.

Acknowledgements

First of all I want to be grateful to Joshua Tristancho for being so patient, for helping me every time I need it and for bringing me the opportunity of participating in this project with whom I have learned so much.

Thanks to Lluís Bonet for his brilliant work with his web blog and for being there all the time helping and taking care of all the people who was involved in the project.

Finally we want to thanks our sponsors and collaborators (see Figure 1).

Figure 1 WikiSat partners

ACRONYMS, ABBREVIATIONS AND DEFINITIONS

Al_2O_3	Aluminum Oxide (alumina)
AN	Ammonium Nitrate
ANCP	Ammonium Nitrate Composite Propellant
AP	Ammonium Perchlorate
APCP	Ammonium Perchlorate Composite Propellant
Aprox	Approximately
BP	Black Powder
CFOAM	Carbon FOAM
CMC	Ceramic Matrix Composite
CP	Composite Propellant
CTPB	Carboxy-terminated Polybutadiene
DB	Double Base
GAP	Glycidal Azide Polymer
GPS	Global Positioning System
GUIPEP	Graphical Interface to PEP
HMX	High Melting Explosive
HTPB	Hydroxy-terminated Polybutadiene
Isp	Specific impulse
LEO	Low Earth Orbit
MCU	Main Control Unit
MEKP	Methyl Ethyl Ketone Peroxide
MIPS	Microprocessor without Interlocked Pipeline Stages
MMC	Metal Matrix Composite
NP	Nitronium Perchlorate
PBAA	Polybutadiene Acrylic Acid
PBAN	Polybutadiene Acrylonitrile
PEPU	Polyether Polyurethane
PMC	Polymer Matrix Composite
PNC	Nitrocellulose Plasticized
PS	Polysulfide
PVC	Polyvinyl Chloride
RDX	Royal Demolition Explosive
SMRs	Solid Rocket Motors
ZS	Zinc Sulphur

CHAPTER 1. TECHNOLOGIES FOR LOW COST NOZZLES

In this chapter few technologies that we will used later are presented such as:

- High performance solid propellants
- Advanced composite materials

1.1 High performance solid propellants

Solid propellants were used since the earliest rockets were built. As we will be able to see in Chapter 3 there are different types of solid propellants classified in five families depending on its components.

High performance solid propellants refer to those that with a low cost and easy manufacture can reach a high performance in thrust and efficiency.

So in this context we assume that most of these propellants are Composite Propellants based. These solid propellants, containing separate fuel and oxidizer intimately mixed, replaced the simple double-base propellants to a considerable extend. Fuels for composite propellants are usually metallic powders. Oxidizers are typically crystalline materials, whereas the binder is a ruber-based material which holds the powder/crystal mixture together in a cohesive grain. Note that the binder itself also acts as a fuel and is oxidized during the combustion. In fact, low smoke formulations used in propulsion systems may incorporate very little metallic powder.

- **Oxidizer:** An oxidizing agent is a chemical compound that readily transfers oxygen atoms. High oxygen content is a desirable characteristic for an oxidizer and is one measure of its performance[1].

Table 1.1 Oxidizers

Compound	**Chemical formula**	**Oxygen Content Mass %**	**Density [kg/m3]**	**Remarks**
Ammonium Perchlorate	NH4CLO4	59.5	1,950	Moderate performance and cost
Ammonium Nitrate	NH4NO3	60.0	1,730	Moderate performance, low cost
Potassium Perchlorate	KCLO4	46.2	2,520	Low regression rate, moderate performance
Nitronium Perchlorate	NO2CLO4	66.0	2,200	Very reactive, unstable

All these compounds are detailed a little more bellow[2]:

[1] http://en.wikipedia.org/wiki/Oxidizing_agent

Ammonium perchlorate (AP): is by far the most common oxidizer in use today. It is characterized by high heat and is a good gas producer. It is moderately low cost, has a moderate performance and good processability.

Ammonium nitrate (AN): is used for slower burning rates. It is characterized by low heat, is a high gas producer, and is good for gas generator propellants. AN has the most applications and while is not as energetic as AP, it is less expensive and provides a clean exhaust. It may be the oxidizer for the future. It contains no toxic elements and therefore, together with a high energy non-polluting fuel, could provide a more 'environmentally friendly' solid propellant.

Potassium perchlorate: is used for fast burning rates. It is characterized by high heat, is a low gas producer.

Nitronium perchlorate (NP): is the most powerful oxidizer but mixtures with nearly all organic compounds are dangerous and explosives and are apt to explode spontaneously. Mixtures with other oxidized materials behave similarly. NP does not produce smoke when burned with fuels.

- **Binder:** The binder holds the entire formulation in a structurally sound grain which can withstand temperature variations as well as pressure and acceleration loads during flight. The aim is to use a binder with low density and energy of combustion.

Binders are usually polymers that can keep propellant's powders and crystals in place by forming a continuous matrix. Binder materials used in composite propellants for SMRs are summarized in Table 1.2.

Table 1.2 Binder materials

Binder designation	**SRM Applications**
Polysulfide (PS)	Older rockets
Polyether Polyurethane (PEPU)	Polaris Stage 1
Polybutadiene Acrylic Acid (PBAA)	Older rockets
Polybutadiene Acrylonitrile (PBAN)	Titan and Shuttle SRM
Nitrocellulose (Plasticized) PNC	Minuteman and Polaris
Carboxy-terminated Polybutadiene (CTPB)	Minuteman Stage 2, 3
Hydroxy-terminated Polybutadiene (HTPB)	IUS, Peacekeeper, Star 48
Glycidal Azide Polymer (GAP)	Proposed for tactical rockets
Nitramines (RDX / HMX)[3]	Not known

- **Fuel additives:** To help provide high energy, many propellants use special fuel additives such as powdered metals [2].

[2] http://www.astronautix.com/articles/comlants.htm
[3] http://www.globalsecurity.org/military/systems/munitions/explosives-nitramines.htm

The most common fuel today is powdered aluminum[4] that gives extra energy with a stable burning. Magnesium is being used in some modern "clean" formulas with reduced exhaust emissions. The beryllium fuels are the most energetic available but the exhaust products are toxic and has a higher cost. For this reason beryllium fuels are considered only for space applications.
Zirconium and titanium are both merely competitive in reference to the volumetric heating value.

The best fuels will be those ones with a high density and a high performance once are mixed with the oxidizer and the binder reaching an increase of the specific impulse and the propellant density.

Table 1.3 Fuels

Fuel	Chemical symbol	Molecular Mass [kg/kmol]	Density [kg/m3]	Remarks
Zirconium	Zr	91.22	6,400	Low performance but high density
Titanium	Ti	47.9	4,500	Low performance but high density
Magnesium	Mg	24.32	1,750	Clean propellant applications
Aluminum	Al	26.98	2,700	Low cost, common fuel
Aluminum Hydride	AlH3	30.0	1,420	Difficult to make
Beryllium	Be	9.01	2,300	Toxic exhaust products
Boron	B	10.81	2,400	Inefficient combustion
Beryllium Hydride	BeH2	11.03	650	Toxic exhaust products

The most commonly used composite propellants used nowadays are detailed bellow. Those are Ammonium Perchlorate Composite Propellant and Ammonium Nitrate Composite Propellant.

1.1.1 *APCP*

Ammonium Perchlorate Composite Propellant is a solid rocket propellant used in a wide range of fields and especially in aerospace propulsion applications.
APCP primarily consists of ammonium perchlorate and aluminum powder usually found in a percentage of 70/15. Both mixed with HTPB binder or PBAN in a percentage of 15. If the performance is not a problem it can be done a low smoke composition such as 80/18/2 of AP/HTPB/Al[5]

- Advantages:

[4] http://www.jtbaker.com/msds/englishhtml/a2712.htm
[5] http://en.wikipedia.org/wiki/APCP

High performance, fast burn rate
Twice the energy density of black powder
Vast amount of information and data available
Easy manufacture
Good for very large motors
No longer considered an explosive
Easier to light than ANCP
Low cost

- Disadvantages:

More expensive compared to Candy propellant or ANCP
Harder to get than candy propellant
Less environmentally friendly than ANCP or Candy

Table 1.4 APCP properties

PROPERTIES	
Molecular Formula	NH_4ClO_4
Molar mass	117.49 g/mol
Appearance	White granular
Density	1.95 g/cm^3
Melting Point	Exothermic decomposition before melting at >200 ºC
STRUCTURE	
Crystal structure	Orthorhombic (< 240ºC)
	Cubic (> 240ºC)

1.1.2 ANCP

Ammonium Nitrate Composite Propellant (ANCP) is a powerful rocket propellant that primarily consists of ammonium nitrate and magnesium powder usually found in a percentage of 60/20. ANCP is cheaper than APCP but it has a lower performance too.
In some cases, mixtures with fast burn rates turn to be explosive if the casing strength is poor or the pressure and rate are incorrectly calculated.

- **Advantages[6]:**

AN propellants have very low temperatures, very good for gas generators
Good performance
Fast burn rate possible
Steady, reliable burn rate
Non Corrosive
Very low cost

[6] http://ae-www.technion.ac.il/admin/serve.php?id=14460

- **Disadvantages[7]:**

More expensive compared to Candy propellant
Harder to get than candy/sugar propellant
Magnesium more expensive than aluminum powder
Harder to ignite
Very low rates require catalysts to get desired rates
Low Isp

Table 1.5 ANCP properties

PROPERTIES	
Molecular Formula	$(NH_4)(NO_3)$
Molar mass	80.052 g/mol
Appearance	white solid
Density	1.725 g/cm^3 (20 ℃)
Melting Point	169.6 ℃
Boiling point	approx. 210 ℃
STRUCTURE	
Crystal structure	Trigonal
EXPLOSIVE DATA	
Shock sensitivity	Very low
Friction sensitivity	Very low
Explosive sensitivity	5,270 m/s

1.2 Advanced composite materials

For the last 30 years [3], the use of high performance polymer-matrix fiber composites in aircraft structures has grown steadily, although not as dramatically as predicted at that time. This is despite the significant weight-saving and other advantages that these composites can provide.

The main reason for the slower-than-anticipated take-up is the high cost of aircraft components made of composites compared with similar structures made from metal, mainly aluminum, alloys. Other factors include the high cost of certification of new components and their relatively low resistance to mechanical damage, low through-thickness strength, and temperature limitations. Thus, metals will continue to be favored for many airframe applications.

The most important polymer-matrix fiber material is carbon fiber-reinforced epoxy. Although the raw material costs of this and similar composites will continue to be relatively high, with continuing developments in materials, design, and manufacturing technology, their advantages over metals are increasing.

[7] http://ae-www.technion.ac.il/admin/serve.php?id=14460

However, competition will be fierce with continuing developments in structural metals. In aluminum alloys developments include improved toughness and corrosion resistance in conventional alloys; new lightweight alloys (such as aluminum lithium); low-cost aerospace-grade castings; mechanical alloying (high temperature alloys); and super-plastic forming. For titanium they include use of powder performs casting, and super-plastic-forming / diffusion bonding.
Advanced joining techniques such as laser and friction welding, automated riveting techniques, and high-speed (numerically controlled) machining also make metallic structures more affordable.

We are considering these materials if they are suitable for low cost nozzles or not. In section 3.3.6 we will consider these issues.

1.2.1 *Drivers for improved airframe materials*

Weight saving through increased specific strength or stiffness is a major driver for the development of materials for airframes. However, there are many other incentives for the introduction of a new material.
In choosing new materials for airframe applications, it is essential to ensure that there are no compromises in the levels of safety achievable with conventional alloys. Retention of high levels of residual strength in the presence of typical damage for the particular material is a critical issue. Durability, the resistance to cyclic stress or environmental degradation and damage, through the service life is also a major factor in determining through-life support costs. The rate of damage growth and tolerance to damage determine the frequency and cost of inspections and the need for repairs throughout the life of the structure.

Table 1.6 Drivers for improved materials [3]

DRIVERS FOR IMPROVED MATERIALS FOR AEROSPACE APPLICATIONS	
Weight reduction	**Improved performance**
increased range	smoother, more aerodynamic form
reduced fuel cost	special aeroelastic properties
higher pay load	increased temperature capability
increased maneuverability	improved damage tolerance
Reduced Acquisition Cost	**Reduced Through-Life Support cost**
reduced fabrication cost	resistance to fatigue and corrosion
improved fly-to-buy ratio	resistance to mechanical damage
reduced assembly costs	

1.2.2 *High-performance fiber composite concepts*

The fiber composite approach can provide significant improvements in specific strength and stiffness over conventional metal alloys. The approach is to use strong, stiff fibers to reinforce a relatively weaker, less stiff matrix. Both the fiber and matrix can be a polymer, a metal, or a ceramic.

So, a fiber composite material consists of a filamentary phase embedded in a continuous matrix phase. The aspect ratio of the filaments may vary from 10 to infinity. Their scale, in relation to the bulk material, may range from the microscopic to gross macroscopic.

Composite constituents, fibers and matrices, can be conveniently classified according to their elastic module and ductility. Within the composite, the fibers may, in general, be in the form of continuous fibers, discontinuous fibers, or whiskers (very fine crystals) and may be aligned to varying degrees or randomly orientated.

1.2.3 *Fiber reinforcements*

Stiff fibers can be made from the light elements; carbon and boron, and the compounds silicone oxide, silicon carbide, and silicon nitride. Fibers can also be made from organic materials based on long-chain molecules of carbon, hydrogen, and nitrogen. Such fibers include aramid fibers. Fibers may be available in the form of single large-diameter filaments or as tows consisting of many thousands of filaments. In the early 90s a new form of carbon called carbon nanotubes was discovered. These are essentially sheets of hexagonal graphite basal plane rolled up into a tube, with a morphology determined by the way in which the sheet is rolled up. The tube walls may be made of single or double layers. Carbon in this way has exceptionally high strength and stiffness.
Whiskers are expensive and difficult to incorporate into composites with high degrees of orientation and alignment. So, they have not been exploited in any practical composites. Although nanotubes are also expensive and similarly difficult to process into composites, they have such attractive mechanical properties and potential for relatively cheap manufacture.

1.2.4 *Matrices*

The matrix, which may be a polymer, metal or ceramic, forms the shape of the component and serves to transfer load into and out of the fibers, separates the fibers to prevent failure of adjacent fibers when one fails and protects the fiber from the environment. The strength of the fiber/matrix interfacial bond is crucial in determining toughness of the composite. The interface, known as interphase, is regarded as the third phase surface. The interface is even more complex in some fibers, notably glass fibers, which are pre-coated with a sizing agent to improve bond strength, to improve environmental durability, or simply to reduce handling damage.
Properties of the composite that are significantly affected by the properties of the matrix include temperature and environmental resistance, longitudinal compression strength, transverse tensile strength and shear strength. The matrix may be brittle or tough.

1.2.4.1 Polymers

Most polymers used for the matrix can be classified in to different groups: thermosetting and thermoplastic. Thermosetting polymers are long-chain molecules that cure by cross-linking to form a fully three-dimensional network and cannot be melted and reformed. They have the great advantage that they allow fabrication of composites at relative low temperatures and pressures. Epoxies have excellent mechanical properties, low shrinkage and form adequate bonds to the fibers. Epoxy systems have service temperatures of 100 ºC to 150 ºC depending on the curing temperature.

Bismaleimide resins have excellent formability and mechanical properties similar to epoxies and can operate at higher temperatures above 180 ºC; however they are more costly.

High-temperature thermosetting polymers such as polyimides, curing at around 270 ºC, allow increases up to 300 ºC. However, they are even more expensive and much more difficult to process.

Thermosetting materials generally have relatively low failure strains. This result in poor resistance to through-thickness stresses and mechanical impact damage can cause delaminations in laminate composites.

Thermoplastic polymers, linear polymers that can be melted and reformed, are also suitable for use as matrices. High-performance thermoplastics suitable for aircraft applications include polymers such as polyetherketone, application approximately to 120 ºC, polyetherketone to 145 ºC, and polyimide to 270 ºC.
Thermoplastic polymers have much higher strains to failure because they can undergo extensive plastic deformations resulting in significantly improved impact resistance.

Because thermoplastic absorb little moisture, they have better hot/wet property retention than thermosetting composites. However they are generally more expensive and are more costly to fabricate because they require elevated temperature processing.

1.2.4.2 Metals

The light metals, magnesium, aluminum, and titanium alloys are used to form high-performance metal-matrix composites. These materials offer the possibility of higher temperature service capabilities, between 150 ºC and more than 700 ºC, and have several other advantages over polymer-matrix composites. However these advantages are offset sometimes by more costly, complex and limited fabrication techniques.

Metals often react chemically with and weaken fibers during manufacture or in service at elevated temperatures, so in translation of fiber properties is often poor. The tendency for a metal to react with the fiber is termed fiber/matrix compatibility. Generally ceramic fibers are most suited for reinforcing metals.

However, carbon fibers may be used with aluminum or magnesium matrices, provided that exposure to high temperature is minimized.

1.2.4.3 Ceramics

For much higher temperatures than can't be achieved with polymer or metal matrices, the options are to employ ceramic-matrix composites.

In the case of the high-modulus ceramic matrices, the fibers provide little stiffening; their purpose is to increase toughness.

Table 1.7 Comparison carbon epoxy with aluminum alloys [3]

COMPARISON CARBON EPOXY WITH CONVENTIONAL ALUMINUM ALLOYS	
Weight reduction	**Acquisition cost**
Saving 15-20% compared with aluminum alloys	Material cost increase
Cost reduction 45 € - 80 € per kg	Reduction due to high conversion rate
Reduction in number of joints	Reduction due to reduction in joints
Performance	Fabrication cost generally increases
Smoother, more aerodynamic form	
Improved aeroelastic properties	**Repair costs**
More resistant to acoustic environment	Fatigue resistant, reduction
More resistant to service environment	Corrosion immune, reduction
Improved fire containment	Fretting resistant, reduction
Improved crash resistance	Impact sensitive, increase
Improved stealth properties	Prone to delamination, increase

1.2.5 *Polymer Matrix Composites (PMCs)*

Polymer matrix composites are those whose matrix is a polymer and the fiber may be in most cases a metal or a ceramic[8].

Most common used composites are epoxy resin matrix systems with carbon, glass, aramid or boron fibers. These types of materials are popular due to their low cost, simple fabrication methods and high level of their mechanical properties.

8 http://www.substech.com/dokuwiki/doku.php?id=polymer_matrix_composites_introduction&DokuWiki=4395b30f1e80be24ce4a79457b23f724

Table 1.8 PMC advantages and disadvantages [3]

Advantages	Disadvantages
High tensile strength	Low thermal resistance
High stiffness	High coefficient of thermal expansion
High fracture stiffness	
Good abrasion resistance	
Good puncture resistance	
Good corrosion resistance	
Low cost	

There are two different groups of polymers used as matrix in composite materials. Those are thermosets and thermoplastics. The main difference between them is that when we increase the temperature of Thermoplastic polymers they melt and when the temperature decreases these polymers return to the previous state. This process can be assumed as reversible while with thermoset polymers don't.

Properties of Polymer Matrix Composites are determined by the orientation, the concentration, the properties of the fibers and by the properties of the matrix.

PMCs are used for manufacturing secondary load-bearing aerospace structures, boat bodies, automotive parts, bullet-proof vests and other armor parts.

1.2.6 *Non-polymeric composite systems*

1.2.6.1 Metal Matrix Composites (MMCs)

Matrix composites based mainly on continuous fibers and metal matrices. Some aircraft applications of these composites include engine components, such as fan and compressor blades, shafts, airframe components, such as spars and skins, and undercarriage components.

Carbon/aluminum and carbon/magnesium alloy composites are particularly attractive for satellite applications. These MMC's combine the high specific properties and low, thermal expansion coefficients exhibited by PMC's together with the advantages indicated in the following table.

Table 1.9 Comparison MMCs / PMCs [3]

ADVANTAGES	DISADVANTAGES
Higher temperature capability, particularly titanium and titanium aluminide	Limited and costly fabrication technology
Higher through-thickness strength, impact damage resistant	Difficult and inefficient joining technology
Higher compressive strength	Limited in temperature capability by fiber/matrix chemical incompatibility
Resistant to impact damage	Prone to thermal fatigue: fiber/matrix expansion mismatch problem
High electrical and thermal conductivity	Prone to corrosion, particularly with conduction fibers

However, MMC’s based on carbon fibers, although potentially low-cost, suffer several drawbacks for non-space applications. These include oxidation of carbon fibers from their exposed ends at elevated temperature and corrosion of metal matrix in wet environments.

The most exploited aluminum matrix MMC is boron/aluminum, based on boron filaments. This MMC is used in the Space Shuttle structure. On the other hand boron aluminum may be superseded by silicon carbide/aluminum, which has the advantage of much greater resistance. The increased resistance simplifies composite fabrication and improves fiber/matrix compatibility at elevated temperature.

Aluminum matrix MMC’s do not offer an increased temperature capability over PMC’s based on high-temperature matrices. So, aluminum MMC’s generally have no major advantage over PMC’s and are far more expensive.

In contrast, titanium alloy and titanium aluminide MMC’s have a large margin on temperature capability over PMC’s. They also have excellent mechanical properties but they cannot match PMC’s in terms of moderate temperature properties and are much more expensive.

Titanium based MMC’s are used to high-temperature applications in high speed transport and gas-turbine engines. Titanium MMC’s lend themselves very well to selective reinforcement and can also be used to reinforce titanium-skinned fan blades.

MMC’s capable of operation to temperatures over 800 ºC are also keenly sought for gas-turbine applications. Unfortunately, the use of available high-performance carbon or ceramic fibers is not feasible with high-temperature alloy matrices because of severe compatibility problems.

To sum up it can be said that costs of MMC’s are very high compared with PMC’s, and the range of sizes and shapes that can be produced is much more limited.

1.2.6.2 Particulate MMCs

Particulate MMCs should be mentioned because they may have extensive aerospace applications as structural materials. In these composites, aluminum or titanium alloy-matrices are reinforced with ceramic particles like silicon carbide or alumina. In these composites the reinforcement is not directional as with the fiber-reinforced MMCs, so the properties are essentially isotropic. Because of this the specific stiffness of these composites suffers a considerable increasing.

The primary fabrication techniques are rapid-liquid-metal processes and also have the considerable cost advantage of being formable by conventional metal-working techniques.

Particulate MMCs have high strength, acceptable fracture toughness, good resistance to fatigue crack propagation, high stiffness and wear resistance compared with conventional alloys.

1.2.6.3 Ceramic Matrix Composites (CMCs)

Ceramic-matrix composites (CMCs) offer the main long-term promise for high-temperature applications in gas turbine engines and for high-temperature airframe structures. The main requirement is for lightweight blades able to operate in environments around 1,400 ºC.

The main limitation is the unavailability of fibers with high-elastic module and strength, chemical stability, and oxidation resistance at elevated temperatures.

Table 1.10 CMCs [3]

SYSTEMS
Silicon carbide/glass; Silicon carbide/silicon nitride
Carbon/carbon; Carbon/glass
Alumina/glass
ADVANTAGES
High to very high temperature capability (500 – 1,500 ºC)
Resistant to moisture problems
Low conductivity
Low thermal expansion
Resistant to aggressive environments
DISADVANTAGES
Fabrication can be costly and difficult
Joining difficult
Relatively low toughness
Matrix microcracks at low strain levels

For suitable reinforcement of ceramic matrices, the fiber must have high oxidation resistance at high temperature, be chemically compatible with the matrix and must match it in its coefficient of thermal expansion.

CMC's are sometimes based on three dimensional fiber architectures because in many applications, the fibers are required to provide toughness rather than stiffness as required in other classes of composites.

Glass and glass-ceramic matrices are promising for applications at temperatures around 500 ºC because of their excellent mechanical properties and relative ease of fabrication. For higher-temperature applications more oxidation resistant fibers such as silicon carbide must be used.

Carbon/carbon composites have no significant chemical or thermal expansion compatibility problems. They are also prone to rapid attack at elevated temperature in an oxidizing environment. Even where oxidation is a problem, the composites can be used where short exposures to severe applications at temperatures over 2,000 ºC are experienced as in rocket nose-cones, nozzles and leading edges on hypersonic wings.

1.2.7 *Hybrid Metal/PMC Composites*

Structural metals, such as aluminum alloys and composites, including carbon/epoxy, have a variety of advantages and disadvantages for airframe applications. Metals tend to fatigue cracking but PMC's are not. PMC's are easily damaged by low-energy mechanical impacts but metals are not. Thus the potential exists to combine these materials in such a way as to get the best of both materials.

1.2.8 *CFOAM*

CFOAM[9] carbon foam is applicable to a broad spectrum of commercial, defense and aerospace markets. It is an enabling technology for a host of next-generation material systems and components replacing those currently based on more conventional materials such as balsa wood, intumescing mats, polymer matrices, metallic honeycombs, ceramic fibrous insulation, ceramic tile, Kevlar-based structures, polystyrene, plastics, fiberglass, rubber and various metals.

Carbon foams are currently being developed for a variety of uses including fire-resistant ship decking and bulkheads, noise and impact mitigation for aircraft, structural panels and firewalls for automobiles, lightweight personnel and vehicular armor, modular construction, and as part of spacecraft thermal management systems.

These foams will not off-gas at elevated temperatures and will not support ignition. Unlike those of most metals and ceramics, CFOAM carbon foam

[9] http://www.cfoam.com/whatis.htm

mechanical properties do not deteriorate with increased temperature if protected from oxidation, making carbon foams an attractive thermal protection material. Carbon foams are also tolerant to impact damage and can be repaired in-place using carbonaceous adhesives. For additional oxidation or impact protection CFOAM carbon foam can be integrated easily with dissimilar materials such as metals or polymer matrix composites (PMCs).

CFOAM carbon foam is a domestically produced material offering advantages such as low cost, enhanced structural properties, fire resistance, radar cross-section, corrosion susceptibility, and low weight.

Table 1.11 CFOAM properties[10]

Property	**Touchstone CFOAM**		**Unit**
	20	25	
Nominal Density	0.32	0.40	g/cc
Compressive Strength	8.3	>15	MPa
Compressive Modulus	620	830	MPa
Tensile Strength	>2.2	>3.5	MPa
Tensile Modulus	500	830	MPa
Shear Strength	1.7	2.1	MPa
Coefficient of Thermal Expansion	5.0	5.0	ppm/℃
Thermal Conductivity	0.25 to 25 Tailorable		W/m·K
Maximum Operational Use Temperature	600 Air 3,000 Inert		ºC
Electrical Resistivity	10^{-2} to 10^{7} Tailorable		ohm-cm
Fire Resistance	Results indicate CFOAM will pass all key fire tests including: radiant panel, smoke generation, toxicity, cone calorimeter, fire resistance, and room corner tests.		

[10] http://www.cfoam.com/pdf/CFOAMProductDataSheet.pdf

CHAPTER 2. WIKISAT SPACE PROGRAM

2.1 Wikisat Project

The Wikisat project is a try to demonstrate that it is possible to put a femto-satellite in a LEO orbit by a private made low-cost mini-launcher. This project is hold by the community. The motivation behind this is to participate in a contest that has similar specifications. This contest is called N-Prize. The current work is a small part of this community effort to develop open collaboration low-cost technology. Some satellite prototypes were done like demonstrators.

2.1.1 N-Prize contest

The N-Prize is a competition to stimulate innovation towards obtaining cheap access to space. According to the rules, the objective is to put into orbit around the Earth a satellite with a mass of between 9.99 and 19.99 grams, and to prove that it has completed at least 9 orbits. No part of any orbit may be lower than 99.99 km above the surface of the Earth[11].

N-Prize has stated that the cost of the launch must fall within a budget of £999.99. A prize of £9,999.99 will be awarded to the first group to achieve it.

The Wikisat group is formed by teachers, students and collaborators. We are developing a femto-satellite called WikiSat and a mini-launcher for this satellite called WikiLauncher to participate in the N-prize [12].

2.1.2 WikiSat: The femto-satellite

The satellite WikiSat showed in Figure 2, it will control the mission, not only in orbit but during the launch. The idea is that the payload has the control of the launcher and takes autonomous decisions when needed. This is one of the group statements related to the design. For precaution, the satellite will not have receiver. The WikiSat is being designed to fit N-Prize rules so even it is a femto-satellite; the mass is less than 20 grams.

The most restrictive parameter is not only the satellite mass, it is the launcher budget. We decided to use Single Fault Tolerant systems as a result of implementing our non-redundancy policy.

2.1.3 WikiLauncher: The mini-launcher

The Wiki-Launcher, see Figure 2, will be controlled by the WikiSat. There is no available prototype yet but the design is mainly decided. This design do not uses fins to avoid the horizontal flight. We want that this design is used to put

[11] http://www.n-prize.com/
[12] http://code.google.com/p/moon-20/

satellites in orbit and not as a missile. A GPS will be used in the mini-launcher but not in the satellite for tracking purposes.

Current simulations say that will be able to carry a 20 grams payload to an altitude of 250 km for few days with a launcher of 1.5 m of length and about 35 kg of mass, using ACPC (Ammonium Perchlorate Composite Propellant) as combustible. Performances can be improved if we add a third stage that works as an elevator and consist of a balloon and a launching ramp.

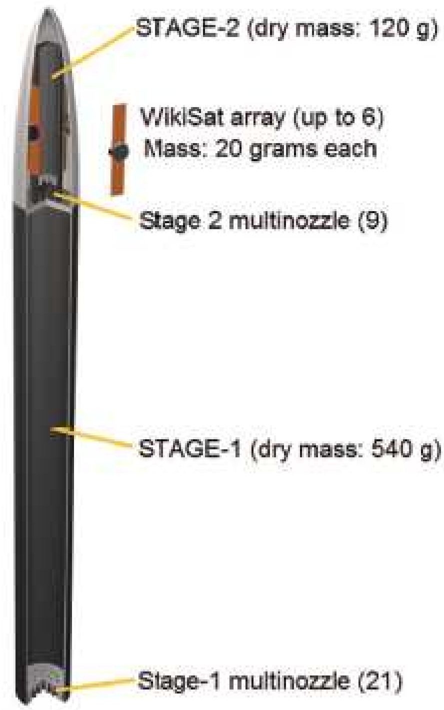


Figure 2 Wiki-launcher and the WikiSat

2.2 WikiSat Satellite

2.2.1 Performances

Orbit: LEO 250 km
Communications range: 200 to 1,000 km
Time in orbit: 8 days
Mass: 20 grams
MIPS: 20

2.2.2 Subsystems

As can be supposed, for mass restrictions the number of subsystems are. So it has the minimum required subsystems to accomplish its mission. These subsystems are:

- Power Supply subsystem
- Communication subsystem
- Structure subsystem
- Attitude determination subsystem
- Position determination subsystem
- Attitude control subsystem
- Tracking subsystem
- Payload subsystem

2.3 Prototypes

During the Wikisat program three femto-satellites were developed like demonstrators that are shown in Figure 3.

- The version one used a Texas Instrument eZ430 MCU + Transceiver, a simple inertial platform and a coin battery.
- The version two was a more sophisticated version includes an Arduino MCU, a Nordic transceiver with an Antenna Array, a 6DOF SparkFun inertial platform and a recycled laptop web cam.
- The final version was a more realistic design that take into account the thermal performances and dynamic loads during the launch.

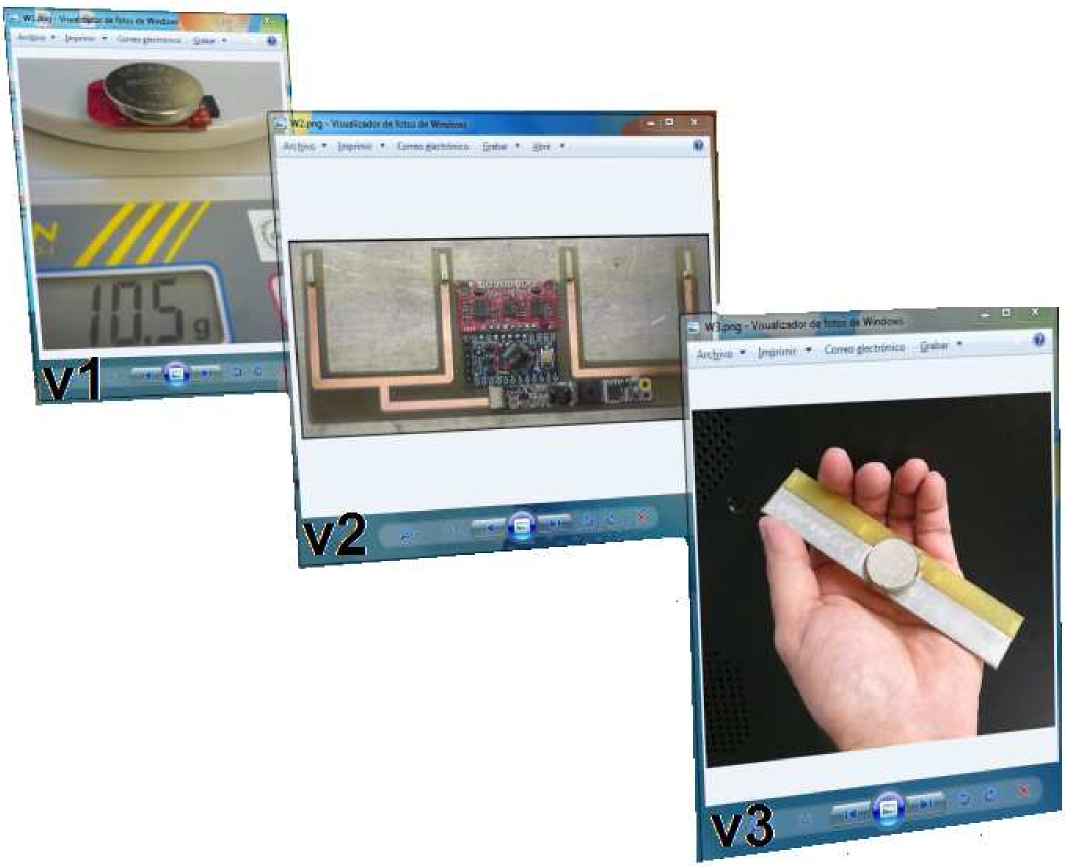


Figure 3 WikiSat versions

2.4 Wiki-Launcher

The Wiki-Launcher is a scalable design able to put a femto-satellite in orbit. When we said scalable we mean that depending on the mission parameters, this launcher is constructed. If the payload is large or the orbit is high, the total size will be greater but the complexity remains constant.

2.4.1 Performances

Number of stages: Two
Propellant: Solid propellant APCP based
Stage thrust limit: 350 N
First stage burnout time: 81 s
Second stage burnout time: 3 s
Time to orbit: less than 7 minutes

2.4.2 Subsystems

Following, a reduced number of subsystems are required to accomplish the objective of this mini-launcher which is to inject, in a simple and autonomous way, a femto-satellite. Some subsystems have double purpose trying to use synergies between subsystems. These subsystems are:

- Safe launch and flight subsystem
- Propulsion subsystem
- Thermal control subsystem
- Electrical power supply subsystem
- Attitude determination and control subsystem
- Position determination and navigation subsystem
- Trajectory simulation subsystem
- Structure subsystem
- Jettison subsystem

CHAPTER 3. NOZZLE DESIGN

In a nozzle design it is involved not only the development of the nozzle itself. We have to take into account many parameters that surround this design such as the propellant used in the rocket, the structure characteristics and the ignition system.

3.1 Low cost propellant study

Rockets create thrust by reacting propellants within a combustion chamber. The result of the propellants reaction is an amount of hot gasses at a high pressure which are expanded and accelerated through a nozzle creating this forward force known as thrust [9].

Almost all rocked propellants consist of two or more chemical components which react together. These two common components are an oxidizer and a fuel.

Studying rocket propellants we can find two main categories of them:

- Liquid rocket propellants
- Solid rocket propellants

3.1.1 Liquid rocket propellants

Liquid propulsion rocket system is the most popular in rocket propulsion because of its high thrust levels and specific impulse with a low propellant tank volume. This is because of the reasonably high density of liquid propellants. All of this permits having tanks with a light weight and reaching the performances required for the rocket propulsion.

In liquid rocket engines fuel and oxidizer are stored in separately tanks and are pumped into the combustion chamber where they are mixed and burned producing a controlled and constant flow and thrust level.

Liquid Rocket engines can be reused for several flights and has the advantage of being throttled at real time. One disadvantage of this type of rocket is that the center of mass is severely affected. The reason for this is because the propellant is a very large proportion of the total mass of the vehicle and while the propellant is being burnt the center of mass is moving closely to the center of drag, being a cause for losing control of the rocket.

A typical liquid propellant rocket engine system includes the combustion chamber, nozzle, and propellant tanks, and all the components used to deliver propellant to the combustion chamber.

- Combustion chamber: is the one that provides enough length and area to allow the mixing of propellants in an efficiently way to achieve the chemical combustion.

- Nozzle: is connected directly to the combustion chamber and converts the enthalpy[13] of the hot gases to kinetic energy to produce the engine thrust.

- Injection: process where it is distributed the propellant to the combustion chamber. The injector has to ensure that the fuel and oxidizer enter the chamber in a fine spray, should deliver the propellants at high pressure and enable rapid mixing. All of this is done to obtain a correct performance of the rocket engine.

3.1.2 *Solid rocket propellants*

The second main category of chemical rocket engines is the one that uses solid rocket propellants. This type of propellant consists also in an oxidizer and a fuel.

Thermodynamically a solid propellant rocket motor is identical to a liquid propellant engine. The way how thrust is obtained, as a result of converting enthalpy to kinetic energy, is also the same. That means the design of the combustion chamber and the nozzle have the same restrictions in both systems and are identical in function.

Solid propulsion rockets differ from liquid ones in that the oxidizer and the fuel are mixed and bounded together in a solid compound, called grain, before the launching and not in real time. Propellants are cased into the combustion chamber to which is embedded the nozzle. Once the inner surface of the grain is ignited, the motor produces thrust continuously until the propellant is exhausted. That is why in solid propellant rockets the combustion chamber and the propellant tanks are the same.

The simplicity of the solid propellant rocket enables wide application. The exhaust velocity is not very high but the absence of propellant tanks and complicated valves and pipelines can produce a high mass ratio and low cost in some cases. The reliability is also very high due to its simplicity of the system.

There are different families of solid propellants detailed bellow [10].

- Black Powder propellants (BP)

The fuel used is charcoal and the oxidizer is potassium nitrate with sulfur as an additive. Black Powder is one of the oldest pyrotechnic compositions with application to rocketry. This propellant has poor properties such as specific impulse (80s) and fracture. Commonly used in amateur rocket models.

- Zinc-Sulfur propellants (ZS)

[13] Enthalpy: A thermodynamic quantity equal to the internal energy of a system plus the product of its volume and pressure. Enthalpy is the amount of energy in a system capable of doing mechanical work; http://wordnetweb.princeton.edu/perl/webwn?s=enthalpy

The fuel used is zinc and the oxidizer is sulfur. ZS application is basically for amateur rocketry due to its poor performance such as fast burn rate and low specific impulse.

- Candy propellants

As the most solid propellants candy propellants are an oxidizer and a sugar fuel that are melted together and shaped into a casing. This type of solid propellant generates a low-medium specific impulse so it is good for amateur rocketry but not for advanced experimental rockets.

- Double-Base propellants (DB)

Double-Based propellants are composed of two monopropellant fuel components. One of them has a high level of energy and the other one not. This is because the first monopropellant is able to achieve the specific impulse required and the second one is to control the instability of the first one. Usually it is used nitroglycerin and nitrocellulose gel and some additives to help the process of solidification. DB propellants are used in applications where a medium-high specific impulse is required and low smoke exhaust too. Some metals such as aluminum can be added to increase the performance.

- Composite propellants (CP)

Composite Propellants are composed of powdered oxidizer and fuel immobilized with a resin binder. Most common composite propellants use ammonium nitrate (AN) or ammonium perchlorate (AP) as oxidizer, which are inorganic salts, and magnesium or aluminum as fuel to increase the energy release and hence the combustion temperature. Composite propellants based in ammonium nitrate are less optimal in performance (Isp of about 210 s) than the ones based in ammonium perchlorate (Isp of about 265 s). Despite this, ammonium nitrate based is less toxic in exhaust than ammonium perchlorate based. The third component in solid propellant is what is called binder that holds the entire formulation in a structurally sound grain.

Table 3.1 Isp comparison

	Isp [s]	**Burn rate**	**Application**
BP	80	Very fast	Pyrotechnic
ZS	45	Extremely fast	Amateur
Candy	130	Fast	Amateur
DB	235	Medium	Medium-advanced rocketry
CP	265	Low	Professional rocketry

3.1.3 *Solid and liquid rocket propellants comparison*

On the one hand liquid-fuelled rocket engines produce a highest performance, but it is complicated and requires both high quality engineering in the combustion chamber and in the propellant delivery systems. The cost in general is very high, and for most applications the engine is used only once and then discarded.

On the other hand solid propellant rockets are much easier to store and handle than liquid propellant rockets. Depending on the compounds it can be reached high performances in specific impulse or thrust but not higher than liquid ones. Solid propellant rockets design is simpler than liquid ones because there is no injector or melting system to build. There is only the need of a combustion chamber where the compounds are mixed before the launching and thanks to the consistency of the propellant are used as a reinforce of the structure.
Because of their high performance, moderate ease of manufacturing, and a moderate cost, solid composite propellants and specifically APCP, finds widespread use in space, military, hobby and amateur rockets.

Table 3.2 Solid vs Liquid [9]

	Advantages	**Disadvantages**
SOLID	Higher propellant density	Generate a high demonstrate performance but lower than liquid
	Less components in the system	No option to control impulse delivery
	System more simple	Need of valves to control spacecraft attitude
	Once the propellant is cased there is no danger about volatility	No control of propellant once it is burned
	Combustion stability	No ground testing
	No control of propellant flow	No pre-launch checkout
	Easy propellant storage and easy to handle	Hard study to reach a good reliability
	Lower cost	Higher effect on the environment
	Easy manufacturing and in most the structure is reusable	Many have no clean exhaust
LIQUID		
	Generate the highest demonstrated performance	Low propellant density compared to solid
	Can control impulse delivery and energy management	Higher total number of components in the system
	Throttling control	More complicated
	Precise control of spacecraft attitude	Concerns about propellant volatility and leakage
	On/Off control	Combustion instability
	Easy to package	Difficult to control propellant flow
	Ground testing and pre-launch checkout	Less reliable for some applications
	High reliability of the system	Mixture ratio control
	Little effect on the environment	Special design in some cases and in most used only once
	Most have clean exhaust	Higher cost

In conclusion it can be said that to achieve a LEO orbit[14] for a 100 kg mass rocket solid propellants will be the best option because of all the reasons said above. Solid propellants are easier to storage and handle and are useful to serve as structure of the rocket.

The solid propellant used will be APCP because of its properties of high specific impulse with a low cost. More properties and characteristics of this propellant will be mentioned and explained bellow.

Candy is the other solid propellant good for its low cost but not so much for its specific impulse. It has been an option but it is better to reach a higher specific impulse even if the price rises a little bit too.

3.1.4 *APCP chemical properties*

Ammonium Perchlorate Composite Propellant (APCP), as said above, is a solid composite rocket propellant used in a wide range of fields such as explosives, flares, ammunition and specifically in rocketry[15].

The difference between APCP and other solid propellants is basically that originally the propellant is powder but once is treated and melted with the other components turns into a solid stage. This is very useful for the process of manufacturing.

APCP has a specific impulse about a range from 180 s to 265 s depending on the configuration and the metal fuel used. Ammonium Perchlorate Composite Propellant is called this way meaning that it is composed of a fuel and an oxidizer mixed and melted with a resin binder, combined into a homogeneous mixture.

Ammonium Perchlorate serves as oxidizer and forms the greater part of the propellant. The fuel used is a powdered metal, usually aluminum but also magnesium or zinc.

The compositions of APCP can change depending on which are the objectives to be reached and the application too. Mass proportions used normally in high performances are around 70% oxidizer, 15% fuel and 15% resin. For a lower performance but also low smoke exhaust it is used a proportion around 80% oxidizer, 18% resin and 2% fuel.

The propellant selected for the second stage is APCP (Ammonium Perchlorate Composite Propellant) with the following composition got from Moon 2.0 simulator:

[14] Implementation of a femto-satellite and a mini-launcher, Master thesis, Joshua Tristancho
[15] Google code project Moon2.0 (February, 2010) http://code.google.com/p/moon-20/

- 70% Ammonium Perchlorate
- 10% Resin (Styrene and Phthalic anhydride)
- 0.01% Catalyst (Methyl ethyl ketone peroxide)
- 20% Fuel (Aluminum powder pure crystalline)

Ammonium Perchlorate (AP) serves as oxidizer, Styrene mixed with Phthalic Anhydride as binders, Methyl Ethyl Ketone Peroxide (MEKP) serves as catalyst and Aluminum Powder serves as fuel in APCP manufacture. A catalyst is a substance that is present in a chemical reaction. It is in physical contact with the components and induces the reaction without affecting on it.

Ammonium perchlorate is an inorganic compound. It is the salt of perchloric acid and ammonium hydroxide. AP as other perchlorates is a potential powerful oxidizer and it is especially labile.
It is classified as an explosive for particle sizes less than 15 micrometers and confers little toxicity itself.

Table 3.3 Ammonium perchlorate properties

PROPERTIES	
Molecular Formula	NH_4ClO_4
Molar mass	117.49 g/mol
Appearance	white granular
Density	1.95 g/cm^3
Melting Point	>200 ℃
Structure	Orthorhombic (< 240 ºC); Cubic (> 240 ºC)

Styrene is a colorless liquid with a sweet smell and evaporates easily. It is an organic compound primarily used in the production of polyestirene plastics and resins. Styrene is also used as an intermediate in the synthesis of materials used for ion exchange resins and to produce copolymers.
Styrene is weakly toxic. Exposure to it in humans can cause eye irritation and effects on the central nervous system.

Table 3.4 Styrene properties

PROPERTIES	
Molecular Formula	C_8H_8
Molar mass	104.15 g/mol
Appearance	colorless oily liquid
Density	0.909 g/cm^3
Melting Point	-30 ℃
Boiling point	145 ℃

Phthalic Anhydride is a colorless solid organic compound used for the production of plasticizers for plastics. It has another major use in the production of polyester resins.

Table 3.5 Phthalic Anhydride properties

PROPERTIES	
Molecular Formula	$C_8H_4O_3$
Molar mass	148.1 g/mol
Appearance	white Flakes, solid
Density	1.53 g/cm^3
Melting Point	131 ℃
Boiling point	295 ℃

Methyl Ethyl Ketone Peroxide is organic peroxide. It is a high unstable explosive, colorless and oily liquid. MEKP is low sensitive to temperature and is more stable in storage. Exposure to it in humans can cause severe skin irritation and progressive corrosive damage.

Table 3.6 Methyl Ethyl Ketone Peroxide properties

PROPERTIES	
Molecular Formula	$C_4H_{10}O_4$
Molar mass	122.3 g/mol
Appearance	Colorless, high-viscosity liquid
Density	1.170 g/ cm^3
EXPLOSIVE DATA	
Shock sensitivity	High
Explosive velocity	Approx. 5,200 m/s

Powdered aluminum is a highly flammable powder created by grinding aluminum metal into fine grains of material. Powdered aluminum can burn violently in the presence of oxidizing substances and certain metal oxides. Hazard increases as fineness increases. It can react with water and with strong acids.

Table 3.7 Powdered Aluminum properties

PROPERTIES	
Symbol	Al
Element category	Metal
Standard atomic weight	26.982 g/mol
Phase	solid
Density	2.70 g/cm^3
Melting Point	660.32 ℃
Boiling point	2,327 ºC
Appearance	Fine, free-flowing, silvery powder.
Odor	Odorless
EXPLOSIVE DATA	
Shock sensitivity	High
Explosive velocity	Approx. 5,200 m/s

After seeing the properties of each component of the APCP mixture it can be seen the exhaust components once the propellant is burned on the Table 3.8. What it is important is to ensure that there are not many exhaust components in liquid state. Those components are the ones that can produce some problems either during the simulation or can cause serious environmental impact.

Table 3.8 Exhaust APCP components[16,17]

0.86205 H_2	0.66920 CO	0.51096 H_2O	0.48618 HCl
0.34579 Al_2O_3	0.29827 N_2	0.16393 H	0.06078 Cl
0.04975 HO	0.04803 CO_2	0.02369 AlCl	0.00885 AlOCl
$6.86\cdot10^{-3}$ $AlCl_2$	$5.40\cdot10^{-3}$ O	$4.46\cdot10^{-3}$ NO	$3.89\cdot10^{-3}$ $AlHO_2$
$2.30\cdot10^{-3}$ AlHO	$1.79\cdot10^{-3}$ AlO	$1.25\cdot10^{-3}$ O_2	$7.78\cdot10^{-4}$ Al
$6.74\cdot10^{-4}$ $AlCl_3$	$3.01\cdot10^{-4}$ Al_2O	$1.37\cdot10^{-4}$ Cl_2	$1.28\cdot10^{-4}$ AlH
$1.03\cdot10^{-4}$ CHO	$7.29\cdot10^{-5}$ OCl	$5.22\cdot10^{-5}$ HOCl	$5.21\cdot10^{-5}$ COCl
$5.11\cdot10^{-5}$ N	$4.64\cdot10^{-5}$ NH_3	$4.31\cdot10^{-5}$ Al_2O_2	$3.18\cdot10^{-5}$ NH_2
$2.55\cdot10^{-5}$ AlO_2	$2.28\cdot10^{-5}$ NH	$2.18\cdot10^{-5}$ CNH	$1.64\cdot10^{-5}$ HO_2
$9.22\cdot10^{-6}$ NHO	$6.80\cdot10^{-6}$ CH_2O	$4.30\cdot10^{-6}$ AlHO	$2.30\cdot10^{-6}$ CNHO

Al_2O_3, is the only one in liquid state and has 0.34579 moles per 100 g. This is an acceptable quantity and there is no need to take it into account.

3.1.5 APCP parameters

The design of a solid propellant rocket motor involves a detailed knowledge of the properties of the propellant. These physical, chemical and the grain properties affect in its burn rate (basic parameter).

---

[16] GUIPEP Graphical User Interface to PEP http://lekstutis.com/Artie/PEP/Index.html
[17] PROPEP program for determine the characteristics of different propellant formulations http://sunsite.unc.edu/pub/archives/rec.models.rockets/PROGRAMS

Also, note that it will carry out only a study of the second stage of WikiLauncher, assuming that the first phase would proceed similarly, simply adjusting the input parameters.
These are the dimensional and performance requirements that the rocket engine is going to achieve:

Table 3.9 Requirements for the rocket engine

Mass distribution	**Value**	**Units**	**Description**
MassProp	0.5485	kg	Propellant mass
Structure	0.0960	kg	Structure mass
Fairing	0.0300	kg	Nose cone mass
Payload	0.0200	kg	Satellite mass
Total mass			
mTotal	0.6945	kg	Total mass (18,6% stage2 dry mass)
Size			
Øchamber	0.0650	m	Combustion chamber diameter
Lchamber	0.0870	m	Chamber effective longitude
Vchamber	0.0003	m^3	Chamber effective volume
RhoProp	1,900.0	kg/m^3	Propellant density
Isp SL	around 220	s	Isp sea level
Isp V	around 260	s	Isp Vaccum

Once fixed the requirements we can proceed to do the simulation of the propellant. We have simulated the mixture seen before using a chemical simulator in the Moon2.0 and with an external one called GUIPEP/PROPEP.

In both cases the simulations has been done with HTPB as resin instead of Styrene and Phthalic anhydride. The results will not change so much because the components are chemically almost the same and the percentage too.

All the simulations have been done assuming the following hypothesis:

- Unidimensional flow for the continuity equations, for the energy and the impulse.
- No speed flow in the nozzle intake
- Completed combustion and adiabatic combustion
- Isentropic expansion in the nozzle
- Homogeneous mixture of reactive and products
- Ideal gas law applied.

With both simulators the results obtained are practically the same except in those that concern to the combustion chamber. In these cases we will choose the more restrictive that is the higher in both. The more pressure or temperature we can reach the more safety we will be able to achieve.

Table 3.10 Simulation results Moon 2.0 – GUIPEP/PROPEP

	MOON 2.0	GUIPEP/PROPEP	Units
Grain density	1.841	1.8414	kg/m³
Propellant temperature	298.14	298	K
Effective molecular weight	31.153	31.153	g/mol
Combustion chamber pressure	3.34	6.89	MPa
Chamber temperature	3,695	3,646	K
Nozzle exhaust pressure	0.101	0.101	MPa
Expansion ratio	5.4	9.59	

Additionally, GUIPEP/PROPEP offers information related to the resulting products:

Effective molecular weight

$$M_{eff} = \frac{m_{systyem}}{m_{products}} \tag{3.1}$$

M_{eff} **= 31.1526 kg/kmol**

Where:

System mass $m_{systyem}$ = 0.1 kg

Moles of gaseous products $m_{products}$ = 3.21 mol

Average molar heat capacity of the mixture: Is the average of the heat capacity per mole.

$$k = \frac{k_{nozzle} + k_{combustion}}{2} \tag{3.2}$$

k **= 1.1553 kg/kmol**

Where:

Molar heat capacity in the nozzle outlet k_{nozzle} = 1.1573 kg/kmol

Molar heat capacity in the combustion chamber $k_{combustion}$ = 1.1532 kg/kmol

- **Performances of the second stage engine**

Now we know the propellant properties produced during combustion, the chamber pressure and the chamber temperature. The next step is to calculate all the different parameters that will define the performance of the solid rocket engine.

Ideal specific impulse: it represents the change in momentum per unit amount of propellant used.

$$I_{SP} = \frac{1}{g_0}\sqrt{2\ T_0\left(\frac{R_u}{M_{eff}}\right)\left(\frac{k}{k\ -\ 1}\right)\left[1\ -\ \left(\frac{P_e}{P_0}\right)^{\left(\frac{k-1}{k}\right)}\right]} \tag{3.3}$$

I_{SP} **= 265.3414 s**

Where:

Standard gravity at sea level	g_0 = 9.81 m/s²
Combustion chamber temperature	T_0 = 3,695 K
Universal Gas constant of perfect gases	R_u = 8,314.04 J/kmol·K
Effective molecular weight	M_{eff} = 31.1526 kg/kmol
Average molar heat capacity of the mixture	k = 1.1553 kg/kmol
Exhaust nozzle pressure	P_e = 0.101 MPa
Combustion chamber pressure	P_0 = 10.1325 MPa

Thrust coefficient: determines the amplification of thrust due to gas expansion in the nozzle as compared to the thrust that would be exerted if the chamber pressure acted over the throat area only.

$$C_f = \sqrt{\frac{2k^2}{k\ -\ 1}\left(\frac{2}{k\ +\ 1}\right)\left[1\ -\ \left(\frac{P_e}{P_0}\right)^{\left(\frac{k-1}{k}\right)}\right]}\ +\ \frac{(P_e\ -\ P_a)\cdot A_e}{P_0\ \cdot\ A^*} \tag{3.4}$$

C_f **= 1.6768**

Where:

Average molar heat capacity of the mixture	k = 1.1553 kg/kmol
Exhaust nozzle pressure	P_e = 0.101 MPa
Combustion chamber pressure	P_0 = 10.1325 MPa
Atmospheric pressure assumed equal to P_e	P_a = 0.101 MPa
Exhaust nozzle area	A_e = 5.31cm^2
Nozzle throat area	A^* = 0.4028 cm^2

Exhaust characteristic speed: is a figure of thermochemical merit for a particular propellant and may be considered to be indicative of the combustion efficiency.

$$c^* = \sqrt{\frac{\left(\frac{R_u}{M_{eff}}\right)T_0}{k\left(\frac{2}{k\ +\ 1}\right)^{\left(\frac{k+1}{k-1}\right)}}} \tag{3.5}$$

c^* **= 1,552.3623 m/s**

Where:

Universal Gas constant of perfect gases	R_u = 8,314.04 J/kmol·K
Effective molecular weight	M_{eff} = 31.1526 kg/kmol
Combustion chamber pressure	T_0 = 3,695 K
Average molar heat capacity of the mixture	k = 1.1553 kg/kmol

Discharge coefficient: is the amount of mass flow per unit pressure on the entry of the convergent section of the nozzle and per unit of critical area.

$$CD = \frac{1}{c^*} \tag{3.6}$$

CD **= 0.00064 s/m**

Where:

Exhaust characteristic speed	c^* = 1,552.3623 m/s

Throat pressure: Pressure in the throat area of the convergent-divergent nozzle is calculated as follows.

$$P^* = \frac{P_0}{\left(1 + \frac{k-1}{2}\right)^{\frac{k}{k-1}}} \tag{3.7}$$

P^* **= 5.8092 MPa**

Where:

Combustion chamber pressure	P_0 = 10.1325 MPa
Average molar heat capacity of the mixture	k = 1.1553 kg/kmol

Throat temperature: Temperature in the throat area of the convergent-divergent nozzle is calculated as follows.

$$T^* = \frac{T_0}{\left(1 + \frac{k-1}{2}\right)} \tag{3.8}$$

T^* **= 3,428.8366 K**

Where:

Combustion chamber temperature	T_0 = 3,695 K
Average molar heat capacity of the mixture	k = 1.1553 kg/kmol

Exhaust Mach number: Exhaust Mach number in the exhaust nozzle outlet plane.

$$M_e = \sqrt{\frac{2}{k-1}\left[\left(\frac{P_0}{P_e}\right)^{\frac{k-1}{k}} - 1\right]} \tag{3.9}$$

M_e **= 3.3239 M**

Where:

Average molar heat capacity of the mixture k = 1.1553 kg/kmol

Combustion chamber pressure P_0 = 10.1325 MPa

Exhaust nozzle pressure P_e = 0.101 MPa

Exhaust nozzle temperature: Exhaust temperature in the exhaust nozzle outlet plane.

$$T_e = \frac{T_0}{1 + \left(\frac{k-1}{2}\right) M_e^2} \tag{3.10}$$

T_e **= 1,989.0868 K**

Where:

Combustion chamber temperature T_0 = 3,695 K

Exhaust Mach number M_e = 3.3239 M

Average molar heat capacity of the mixture k = 1.1553 kg/kmol

Optimum expansion ratio: is the ratio between the exhaust area and the throat area as a function of the Mach number.

$$\frac{A}{A^*} = \frac{1}{M_e}\left[\frac{1 + \left(\frac{k-1}{2}\right) M_e^2}{1 + \left(\frac{k-1}{2}\right)}\right]^{\left(\frac{k+1}{2(k-1)}\right)} \tag{3.11}$$

$\frac{A}{A^*}$ **= 13.1795**

Where:

Exhaust Mach number M_e = 3.3239 M

Average molar heat capacity of the mixture k = 1.1553 kg/kmol

Summary of the second stage engine shown on the following table:

Table 3.11 Performance parameters

Symbol	*Value*	*Unit*	*Description*
Atmospheric parameters			
Pa	0.1010	*MPa*	Atmospheric pressure assumed equal to Pe
Ta	298.00	*K*	Atmospheric temperature (25 ºC)
Combustion chamber parameters			
P0	10.1325	*MPa*	Combustion chamber pressure
T0	3,695.0000	*K*	Combustion chamber temperature (3,422 ºC)
A0	0.0033	*m²*	Combustion chamber area
cv	1.1532	*kg / K·mol*	Combustion chamber specific heat
cstar	1,552.3623	*m / s*	Exhaust characteristic speed
Cf	1.6768		Thrust coefficient
CD	0.0006	*s / m*	Discharge coefficient
Throat parameters			
Pstar	5.8092	*MPa*	Throat pressure
Tstar	3,428.8366	*K*	Throat temperature (3,155 ºC)
Astar	0.00004028	*m²*	Nozzle throat area
OER	13.1795		Optimum expansion ratio
Exhaust nozzle parameters			
Pe	0.101	*MPa*	Exhaust nozzle pressure
Te	1,989.0878	*K*	Exhaust temperature (1,795 ºC)
Ae	0.0005	*m²*	Exhaust nozzle area
Me	3.3239	*M*	Exhaust Mach number
cp	1.1573	*kg / K·mol*	Exhaust nozzle specific heat
Propellant characteristics and others			
Meff	31.1526	*kg / K·mol*	Effective molecular weight
mSystem	0.1000	*kg*	System mass
mProducts	3.2100	*mol*	Gaseous products mol
k	1.1553	*kg / K·mol*	Average specific heats in the mixture
knozzle	1.1573	*kg / K·mol*	Specific heat in the nozzle outlet
kcombustion	1.1532	*kg / K·mol*	Specific heat in the combustion chamber
γ	1.1532	*kg / K·mol*	Specific heats ratio
Performances second stage			
Isp	265.3414	*S*	Ideal specific impulse
g0	9.81	*m / s²*	Acceleration at Sea level
Rd	8,314.04	*J / K·mol*	Universal gas constant of perfect gases

- **Corrections for the ideal engine**

Combustion chamber efficiency: Combustion efficiency and wall heat loses reduces the theoretical pressure. However, solid propulsion has very high combustion efficiency as long as a good mixture is guaranteed in the propellant grain at the time that the grain size is very small (less than 100 μm).

$$\eta^* = \frac{\bar{c}^*}{c^*} \tag{3.12}$$

η^* **= 0.9736**

Where:
Average exhaust characteristic speed $\bar{c}^*$ = 1,511.4 m/s
Exhaust characteristic speed c^* = 1,552.3623 m/s

The average exhaust characteristic speed can be obtained from static tests and its value can oscillate around the 98% and 99% of the ideal one.

Nozzle efficiency due to the divergence: The flow through a real nozzle is not the same as in an ideal one. We have to take into account some effects due to friction, heat transfer (especially in the throat), non-ideal gas, incomplete combustion, non-axial flow, etc. For all these losses we use the following correction factor.

$$\lambda = \frac{1}{2}\left(1+\cos\alpha\right) \tag{3.13}$$

λ **= 0.9891**

Where:
Nozzle divergence semi-angle α = 12 º

Nozzle efficiency due to the discharge: This parameter expresses how a good design of the nozzle allows a better mass flow through the nozzle throat. The value of the correction factor is typically 0.91 for a propellant with a good nozzle design with smooth surfaces and minimal loss of heat.

$$\varsigma_d = \frac{\overline{\dot{m}}^*}{\dot{m}^*} \tag{3.14}$$

ς_d **= 0.91**

Nozzle efficiency due to the chamber pressure: The efficiency can be improved having smooth surfaces and a good throat design in the inlet section. This parameter is around 0.95.

ς_P **= 0.95**

Delivered specific impulse: Specific impulse calculated taking into account the nozzle corrections.

$$I_{SPdelivered} = I_{SP} \cdot \eta^* \cdot \varsigma_d \cdot \varsigma_P \cdot \lambda \tag{3.15}$$

$I_{SPdelivered}$ **= 220.8946 s**

Where:
Specific impulse I_{SP} = 265.3414 s
Combustion chamber efficiency η^* = 0.9736
Nozzle efficiency due to the discharge ς_d = 0.91
Nozzle efficiency due to the chamber pressure ς_P = 0.95
Nozzle efficiency due to the divergence λ = 0.9891

Table 3.12 Corrections for the ideal engine

Symbol	***Value***	***Unit***	***Description***
Combustion chamber corrections			
ηstar	0.9736		Combustion chamber efficiency
cstarav	1,511.4000	*m / s*	Average exhaust characteristic speed
Nozzle corrections			
λ	0.9891		Nozzle efficiency due to the divergence
α	12.00	*o*	Nozzle semi-angle of divergence
ζd	0.91		Nozzle efficiency due to the discharge
ζp	0.95		Nozzle efficiency due to the chamber pressure
Performances second stage with corrections			
_Ispdelivered	220.8946	*s*	Delivered specific impulse

3.2 Low cost structure study

When we talk about structure we are referring to the combustion chamber or casing. The case of a SRM serves both as a container for the propellant and as pressure chamber where the fuel is burned to provide thrust when expanded trough the nozzle.

The development of a case involves considering all the thermodynamic properties required, the mechanical forces and all the parameters related in the burning process. We also must consider the factors that will affect to the overall performance of the rocket and obviously the cost.

The case design must consider:

- External configuration, shape and size for a maximum performance of our rocket
- How the external conditions will affect to the mechanical, chemical and thermal properties of the case

The materials most commonly used for a case design can be from metals to reinforced plastics. Steels and alloys have been extensively used as structural materials, but also titanium alloys, aluminum alloys and reinforced plastics such as fiberglass reinforced composite material.

A good way to build a low cost structure would be using commercial tubes of carbon fiber that have the advantage of a low cost prize. Instead of this, the best way for smaller stages is to use steel cans which are very cheap, less demanding in design and also tolerate higher temperatures. The drawback is that the design is determined by the sizes offered by the market.

After managing different options we have decided to do the combustion chamber with soda cans made of steel.

Table 3.13 Steel properties

Material	**Alloy steel (heat treated)**	**Units**
Tensile strength	1,400 – 2,000	MPa
Modulus of elasticity	207	MPa
Density	7.84	g/cm^3
Strength to density ratio	205	
Melting Point	1,418	ºC

On the one hand cans are able to support large pressures. On the other hand steel is a good material for its mechanical and thermal properties. It may not seem the best but it achieves good results for our objective. The only inconvenient is that the design is limited to the can size and maximum volume, so the propellant quantity will be limited to these parameters.

We have to consider that the temperature generated by the propellant at the combustion chamber will be around 3,155 ºC. This is twice the melting point temperature of the steel. The solution to this ambiguity of thermal requirements we have added epoxy resin on the surface inside the can to create a kind of thermal protection. Epoxy resin will act as insulator giving the can thermal protection and structural stiffness.

Table 3.14 Chamber parameters

Size	**Value**	**Units**	**Description**
Øchamber	0.0650	m	Combustion chamber diameter
Lchamber	0.0870	m	Chamber effective longitude
Vchamber	0.0003	m^3	Chamber effective volume
Structure	0.0960	kg	Structure mass

3.3 Low cost nozzle study

3.3.1 Definition

As said above a nozzle is a mechanical device designed to control the characteristics of a fluid flow that converts the enthalpy of the hot gases produced after the propellant burning into kinetic energy and produces engine thrust. This conversion must be as efficient as possible in order to obtain the highest exhaust velocity along the desired direction.

3.3.2 Introduction to theory of nozzles

Nozzles can be single-nozzle or multi-nozzle. Single-nozzles are less efficient in terms of power referred to a specific size but using multi-nozzle ones in amateur rockets only make sense if it is reused to a maximum of four units [11].

Rockets typically use a fixed convergent section followed by a fixed divergent section for the design of the nozzle. This nozzle configuration is called a convergent-divergent nozzle.

Nozzle design [4] is directly related with the propellant and the combustion chamber characteristics. Normally it is defined by the combustion chamber size and the throat diameter. Cone shapes are usually used and this defines de exhaust nozzle diameter.

The design of the nozzle must trade off:

- Nozzle size, needed to get better performance, against nozzle weight penalty.
- Complexity of the shape for a good performance versus the cost of fabrication

For these reasons we have decided to develop the easiest and simplest nozzle design which is cone-shape nozzle.

The conical nozzle was used often in early rocket applications because of its simplicity and ease of construction. The cone gets its name from the fact that the walls diverge at a constant angle. A small angle produces greater thrust, because it maximizes the axial component of exit velocity and produces a high specific impulse. The penalty, however, is a longer and heavier nozzle that is more complex to build. At the other extreme, size and weight are minimized by a large nozzle wall angle. Unfortunately, large angles reduce performance at low altitude because the high ambient pressure causes overexpansion and flow separation. Typical angles for a good performance at the divergent section are those between 12º and 18º and angles under 60º for the convergent section.
Because of manufacture reasons we have chosen a 12º angle for the divergent section and 45º for the convergent one.

The theory of one-dimensional steady gas dynamics can be applied in our nozzle design because it involves the situation of a fluid flow in a short duct of small divergence and no curvature on the basis that the fluid behaves like a perfect gas and the gradients of pressure, temperature and density are negligible. The physical parameters of a perfect gas are constant and uniform on each cross-sectional area and the flow of perfect gas is an isentropic and adiabatic process, so we assume no energy loss or heat exchange.

Under these conditions, the motion of the gas in the nozzle can be considered as an isentropic process, and the equations of one-dimensional steady gas dynamics can be applied in the design of the supersonic nozzle for the second stage.

Bearing in mind these conditions a supersonic nozzle must be a converging-diverging nozzle which consists of three sections: convergent (subsonic zone), throat (critical zone) and divergent (supersonic zone).

In order to produce an exit with high momentum, good uniform and a tidy boundary, as well as with low turbulence and energy loss, the dimensions of each section in supersonic nozzle need to be designed correctly and calculated precisely on the basis of gas dynamics.

- **Convergent Section**

The function of the convergent section is to accelerate gas flow, but at the same time, to keep the flow uniform and parallel.

The characteristics of the convergent section are mainly determined by the converging ratio[18], calculated dividing the convergent inlet area with the throat area, which accelerates the gas flow and ensures the speed of flow to reach sonic speed.
There are numerous theories about the design of a converging curvature. However, most of them are quite complicated. In the case of our nozzle design there is no curvature so the shape will be a cone with an angle of 45º.

- **Throat Section**

The design of the throat section is relatively important because it is a transitional cross-sectional area which transfers the subsonic speed into the supersonic speed.

The throat size is chosen to choke the flow and set the mass flow rate through the system. The flow in the throat is sonic which means the Mach number is equal to one in the throat.

The cross-sectional area closer to the throat section cannot be varied quickly, so that a circular arc with quite a large radius is provided over the region of transition in gas dynamics.

In the case of the conical nozzle the radius of the circular arc is calculated as an approximation of: 1.5 times the nozzle throat radius.

- **Divergent Section**

The function of the divergent section is to further accelerate the flow, which latter has achieved sonic speed at the throat section, by means of expansion, until the exit reaches an expected Mach number that depends on the area ratio of the exit to the throat.

The expansion of a supersonic flow causes the static pressure and temperature to decrease from the throat to the exit, so the amount of the expansion also

[18] http://www.gasturb.de/GasTurb11Help/index.html?convergentdivergentnozzle.htm

determines the exit pressure and temperature. The exit temperature determines the exit speed of sound, which determines the exit velocity. The exit velocity, pressure, and mass flow through the nozzle determine the amount of thrust produced by the nozzle.

This section is the most important in a supersonic nozzle. The dimensions of the divergent section are shown below with the other parameters.

Once again, the curvature of the divergent section of our second stage nozzle will be null. The best angle in this section is estimated between 12º and 18º and by manufacture reasons we have decided to make it with a divergent angle of 12º. All these fixed parameters will modify or establish the bases for the calculation of the rest of them.

3.3.3 Characteristic Analysis on a Supersonic Nozzle

In a converging diverging nozzle gas flows[19] through the nozzle from a region of high pressure (combustion chamber pressure) to one of low pressure (exhaust nozzle pressure).

The gases that flow through the converging section are subsonic and increase their velocity while the area decreases. Then an increase in the area produces a decrease in the velocity. So, if the flow in the throat is subsonic, the flow downstream of the throat will decelerate and stay subsonic because of the divergence section.

At this point we say that the nozzle has become choked. You could delay this behavior by making the nozzle throat bigger but eventually the same thing would happen.

The reason for this behavior has to do with the way the flows behave at Mach 1. In a steady internal flow the Mach number can only reach 1 at a minimum in the cross-sectional area (throat).

So if the converging section is too large and does not choke the flow in the throat, the exit velocity will be very slow and will not produce much thrust.
On the other hand, if the converging section is small enough so that the flow chokes in the throat, if we increase the area the flow will go to supersonic (diverging section). For a supersonic flow an increase in the area produces an increase in the velocity.

It is exactly the opposite of what happens subsonically. For subsonic flows, the density remains constant, so an increase in the area produces a decrease in velocity to conserve mass. For supersonic flows, both the density and the velocity are changing as we change the area in order to conserve mass.

[19] http://www.fas.org/man/dod-101/sys/missile/docs/RocketBasics.htm

- **The flow pattern**

According to the gas velocity and how it behaves in the nozzle there are 7 patterns of flow that we can study:

- Subsonic flow (see Figure 4): the flow is completely subsonic along the whole nozzle. This is because the difference between pressures is not far enough and the nozzle converging section is too large. The flow accelerates out of the combustion chamber through converging section reaching his maximum speed at the throat. Despite this, the maximum speed still remains subsonic and then decelerates through the diverging section and exhaust into the ambient.

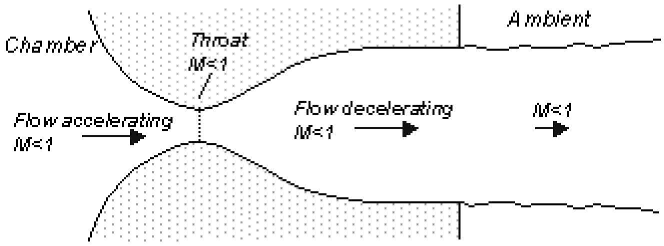


Figure 4 Subsonic flow pattern

- Flow just choked (see Figure 5): In this case the situation is similar than in the case before. The difference between pressures is higher but the flow remains subsonic all along the nozzle. The flow reaches M=1 at the throat but then choked and decelerates through the diverging section.

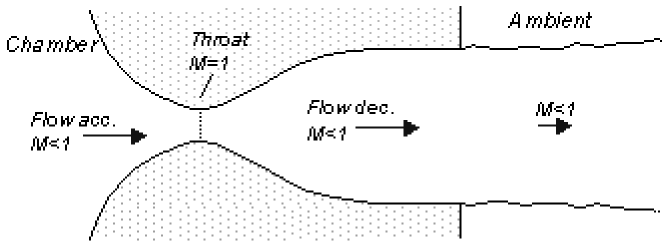


Figure 5 Flow choked pattern

- Shock in the nozzle (see Figure 6): Increasing a little bit the difference between pressures we can reach a state where the flow behaves subsonic before the throat and supersonic after, but before it reaches the end of the nozzle a shock wave is produced and then there is a flow deceleration along the diverging section and exhausts as a subsonic flow.

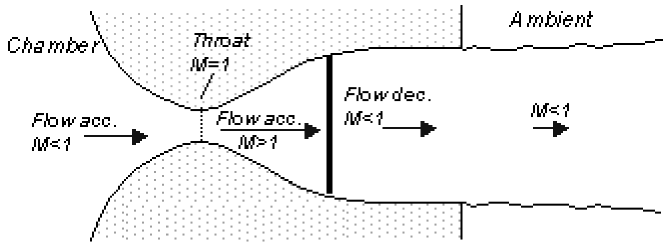


Figure 6 Shock in the nozzle flow pattern

- Shock at exit (see Figure 7): Just increasing the difference between pressures enough the supersonic region can be extended sitting the shock at the nozzle exit. After the shock the flow will be subsonic again so the thrust achieved will not be the optimal.

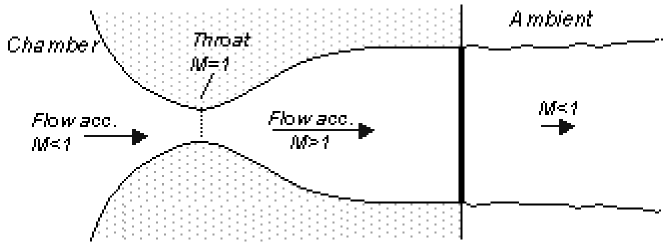


Figure 7 Shock at the exit flow pattern

- Over-expanded (see Figure 8): It is the same concept like the one with the shock at the exit but this time the ratio between pressures is enough to get supersonic velocity of the flow farer than the nozzle exit. The problem in this case is that the flow pressure at the end of the nozzle is lower than the one in the environment. The flow has been expanded by the nozzle too much and this causes a complex pattern of shocks and reflections which involve a mixture of subsonic and supersonic flow.

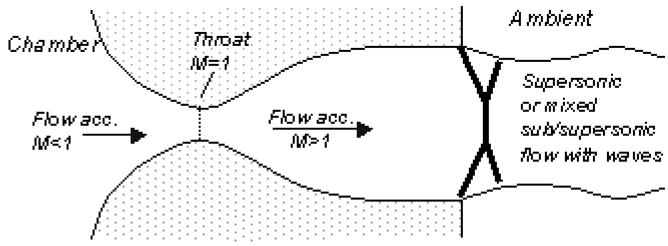


Figure 8 Over-expanded flow pattern

- Design condition (see Figure 9): Is the one that achieves the equilibrium and uniformity between pressures at the exit of the nozzle and the ambient pressure. The waves disappear and all the flow through the divergent section and farer is supersonic reaching a high momentum and a tidy boundary.

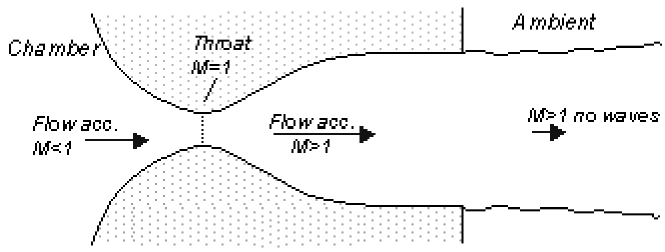


Figure 9 Design condition flow pattern

- Under-expanded (see Figure 10): Is another case of difference between pressures at the exit of the nozzle that causes a complex wave pattern. In this case the flow is supersonic all along the divergent sections and farer.

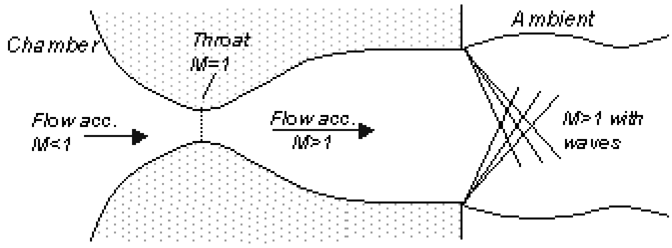


Figure 10 Under-expanded flow pattern

3.3.4 Nozzle design

We can say that the nozzle design is reduced to the calculation of the throat parameters, the total length (see Figure 11) and the ratio between the inlet, throat and outlet diameters.
So, from the initial data configuration and size we want to give to the nozzle and from the previous data parameters given by the propellant characteristics, we have calculated the size of the throat and the length as follows:

Nozzle throat area

$$A^* = Ae / OER \qquad (3.16)$$

A^* **=0.4028 cm²**

Where:
Nozzle throat diameter Ae= 0.72 cm
Optimum expansion ratio OER= 13.1795

Nozzle throat diameter

$$\varnothing_{nozthroat} = 2\sqrt{A^* / \pi} \qquad (3.17)$$

$\varnothing_{nozthroat}$ **= 7.2 mm**

Where:
Nozzle throat area A^*= 0.4028 cm²
Pi number π= 3.14159

Divergent ratio

$$\sqrt{\varepsilon} = R_{exhaust} / R_{throat} \qquad (3.18)$$

$\sqrt{\varepsilon}$ **= 3.6304**

Where:
Nozzle throat radius R_{throat} = 0.0036 m
Nozzle exhaust radius $R_{exhaust}$ = 0.013 m

Radius of curvature of the nozzle contraction: for conical nozzles it is applied this formula.

$$R_{curve} = 1.5 R_{throat} \tag{3.19}$$

R_{curve} **= 0.0054 m**

Where:

Nozzle throat radius R_{throat} = 0.0036 m

Nozzle divergent length

$$L_N = \frac{R_{throat}\left(\sqrt{\varepsilon}-1\right)+R_{curve}\left[\frac{1}{\cos\alpha}-1\right]}{\tan\alpha} \tag{3.20}$$

L_N **= 0.04479 m**

Where:

Nozzle throat radius R_{throat} = 0.0036 m

Divergent ratio $\sqrt{\varepsilon}$ = 3.6304

Radius of curvature of the nozzle contraction R_{curve} = 0.0054 m

Nozzle semi-angle of divergence α = 12 º

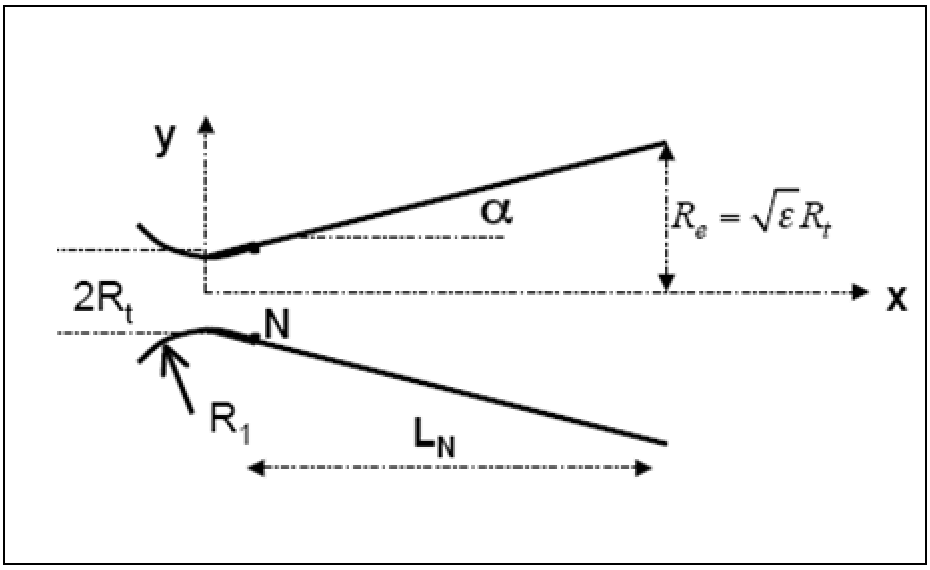


Figure 11 Nozzle length parameters

Table 3.15 Nozzle parameters

Symbol	***Value***	***Unit***	***Description***
Nozzle parameters			
Pe	0.1010	*MPa*	Exhaust nozzle pressure
Te	1,989.0878	*K*	Exhaust temperature
Ae	0.0005	*m²*	Exhaust nozzle area
Me	3.3239	*M*	Exhaust Mach number
cp	1.1573	*kg / K·mol*	Exhaust nozzle specific heat
_Øchamber	0.0650	*m*	Combustion chamber diameter
_Øexhnozzle	0.0260	*m*	Exhaust nozzle diameter
_Ønozthroat	0.0072	*m*	Nozzle throat diameter
_Lchamber	0.0870	*m*	Chamber effective longitude
_Vchamber	0.0003	*m³*	Chamber effective volume (0.003 m²)
_Rexhnozzle	0.0130	*m*	Exhaust nozzle radius
_Rthroat	0.0036	*m*	Nozzle throat radius
_Rcurvethroat	0.0054	*m*	Radius of curvature of the nozzle contraction
_sqrtepsilon	3.6304	*adimensional*	Divergent ratio
_Length Nozzle	0.0449	*m*	Nozzle divergent length
_Pstar	5.8092	*MPa*	Throat pressure
_Tstar	3,428.8366	*K*	Throat temperature
_Astar	0.0000403	*m²*	Nozzle throat area
_OER	13.1795		Optimum expansion ratio
_α	12.0000	*º*	Nozzle semi-angle of divergence
_B	45.0000	*º*	Nozzle semi-angle of convergence

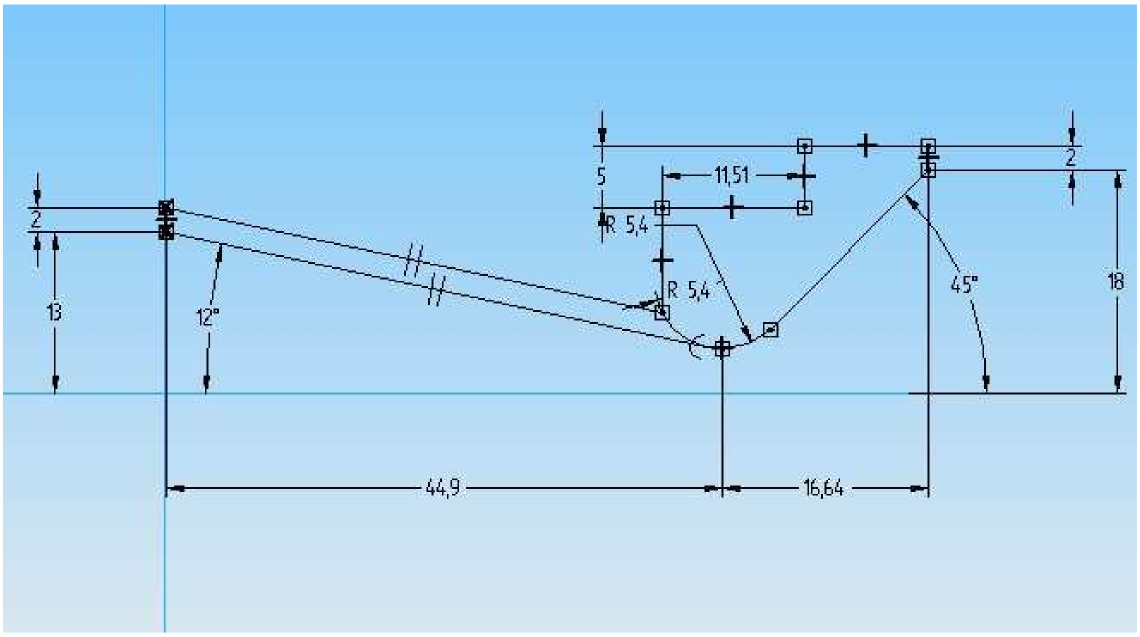


Figure 12 Revolution pattern measure for the interior shape of the nozzle

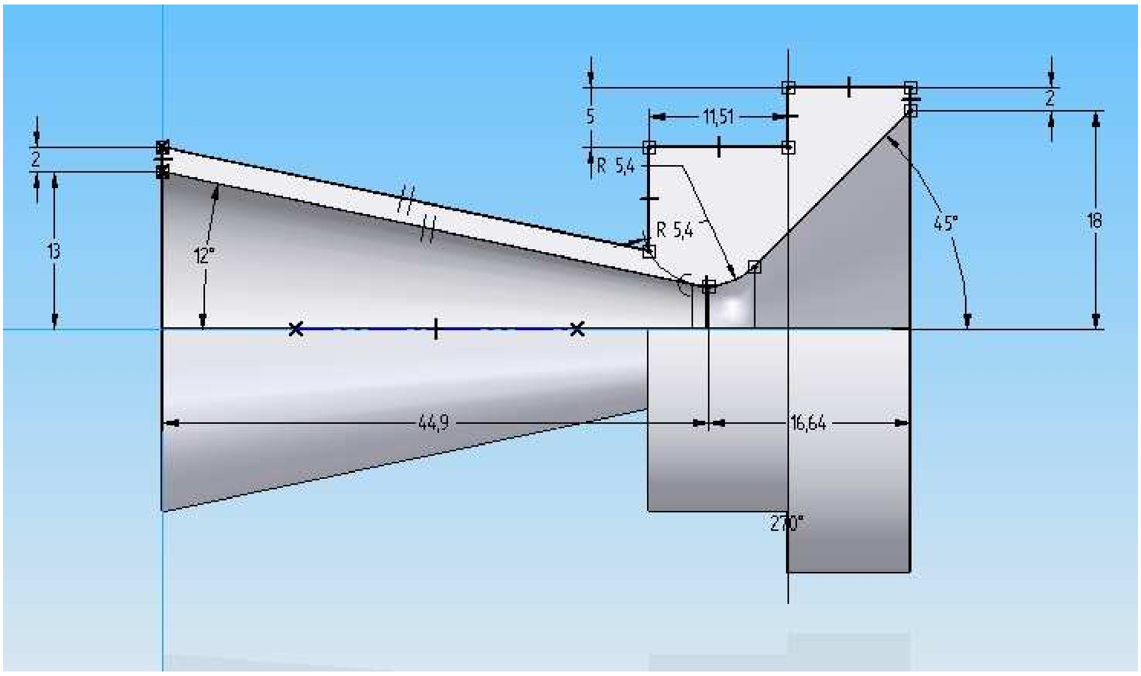


Figure 13 Measures of the nozzle

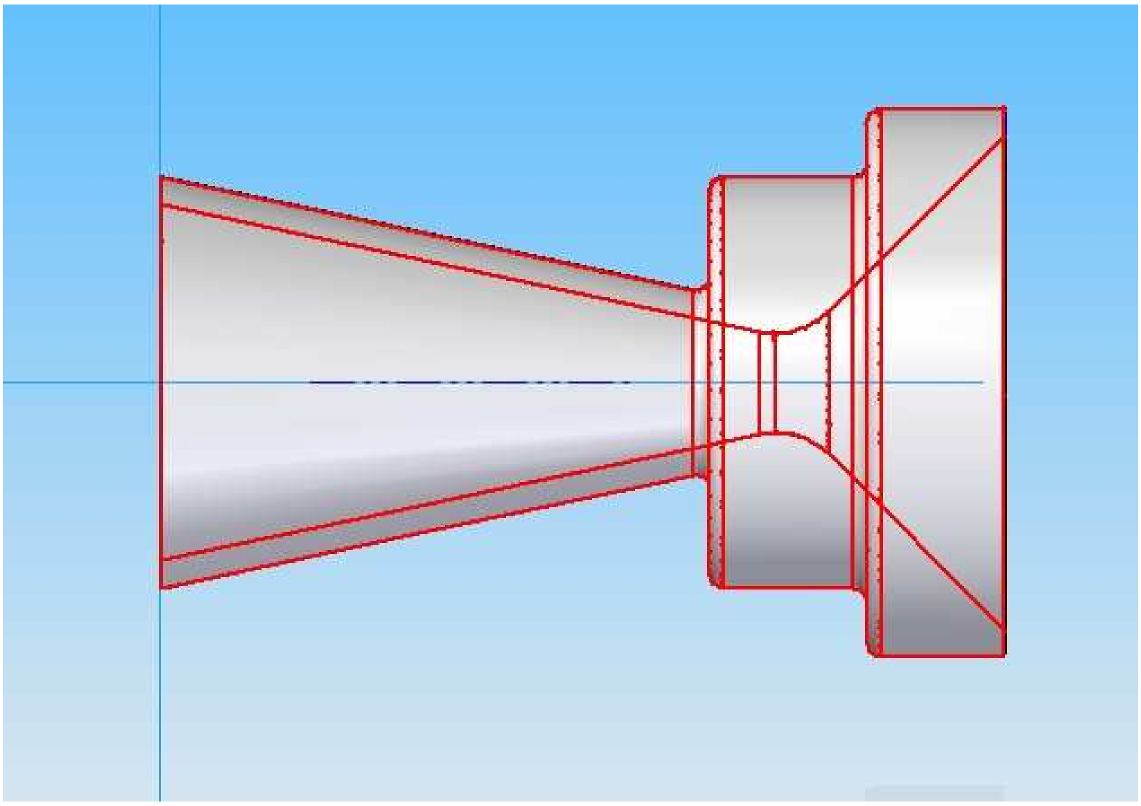


Figure 14 Interior and exterior view for the shape of the nozzle

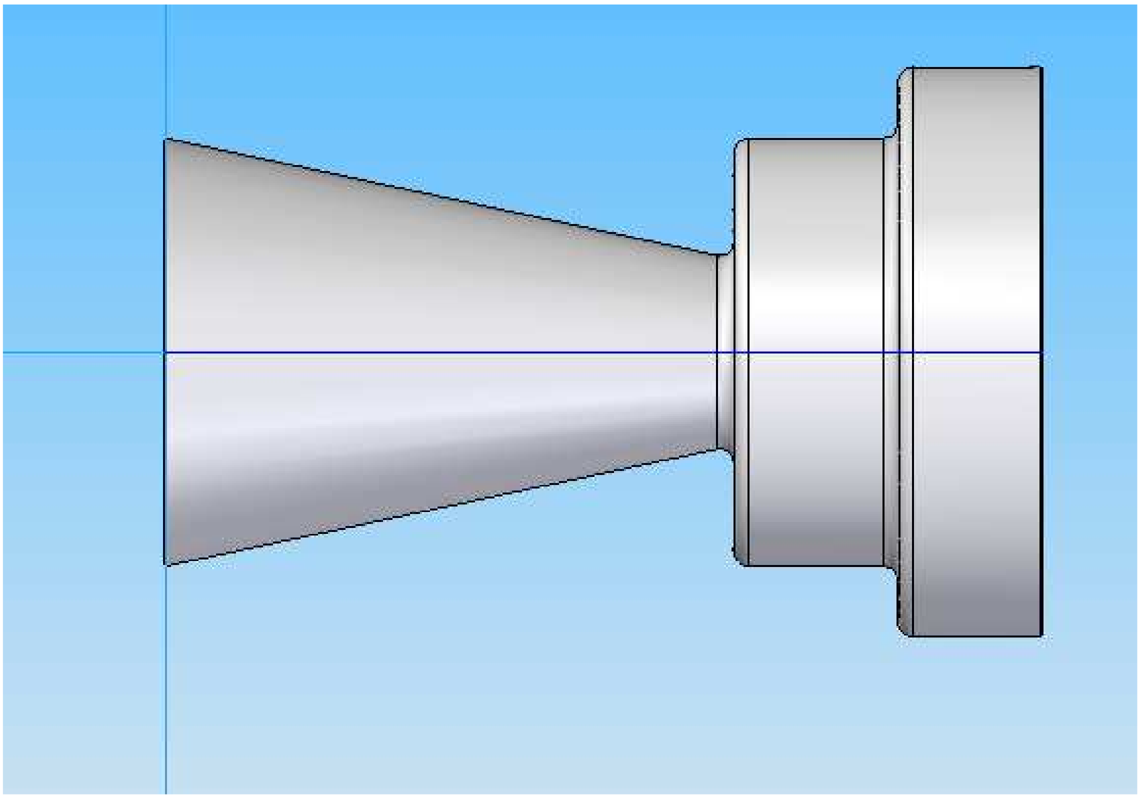


Figure 15 Lateral view of the nozzle

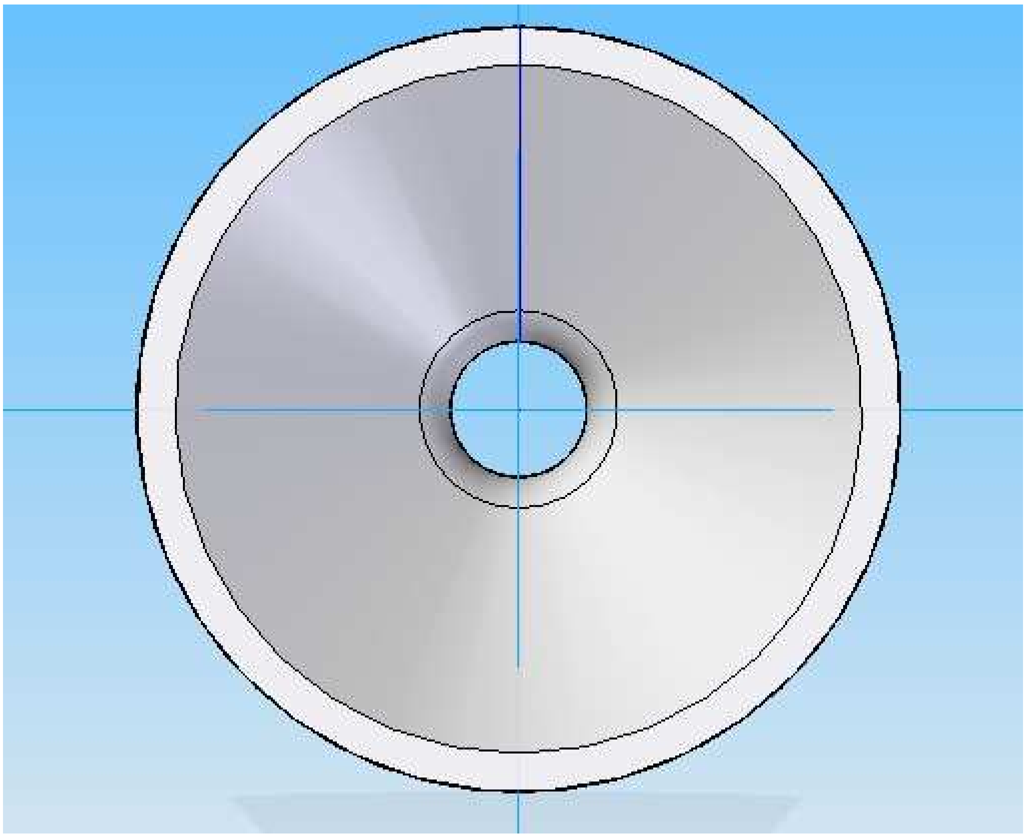


Figure 16 Front view of the nozzle

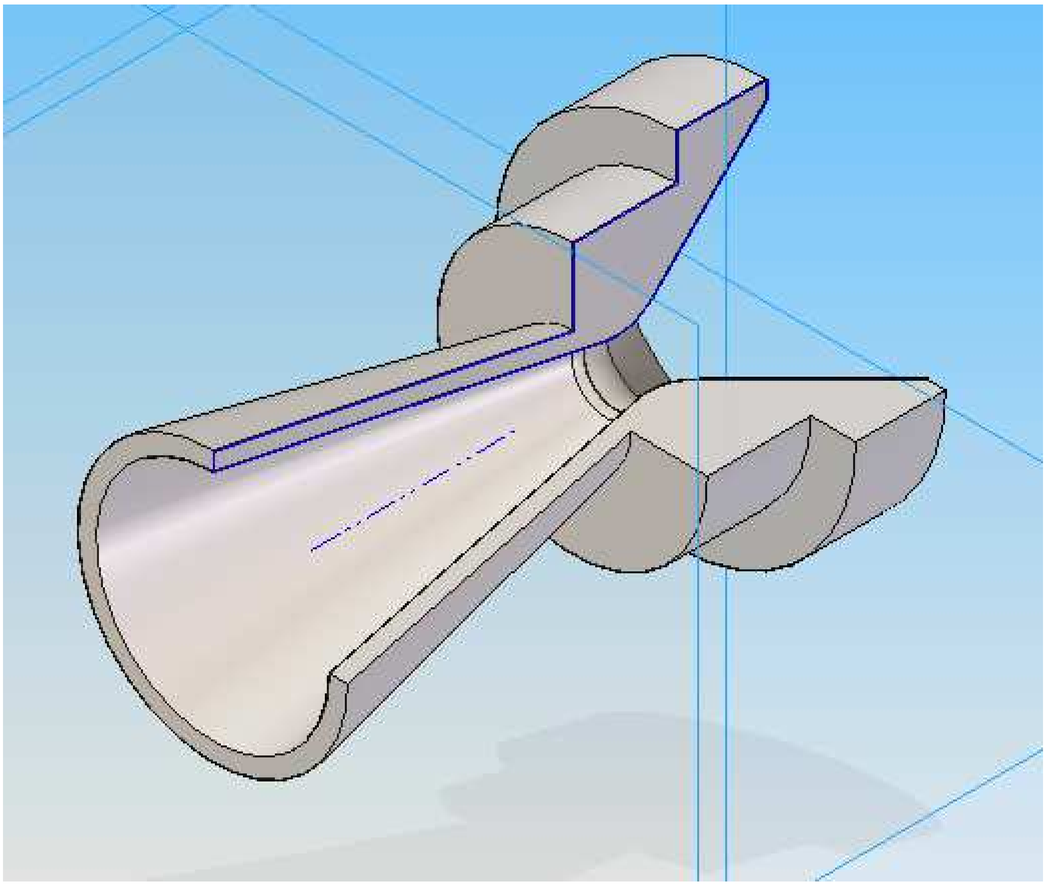
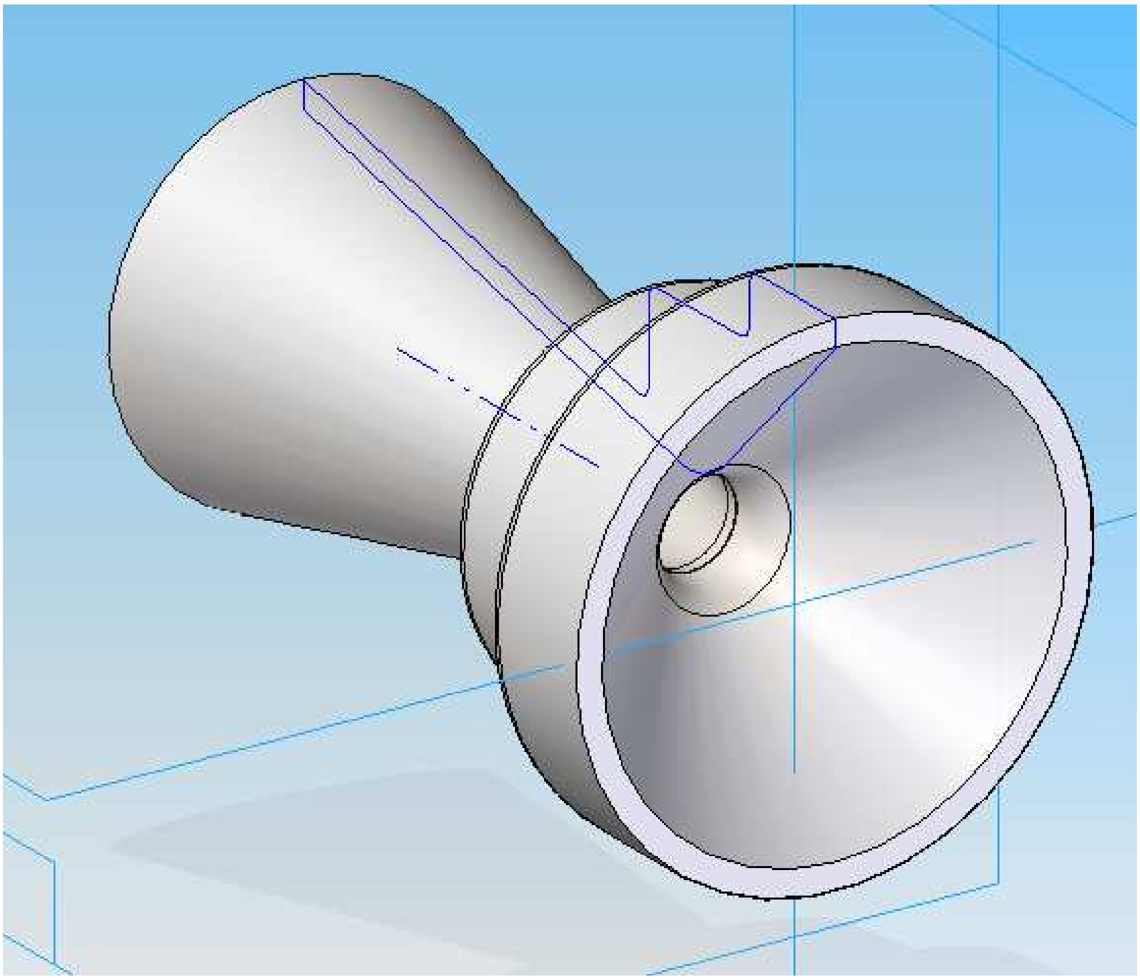


Figure 17 Perspective with ¼ section view of the nozzle

Table 3.16 Parameters of the nozzle and perspective front view

Second stage nozzle summary			
Cone shape / Converging – diverging / Aluminum made			
Symbol	***Value***	***Unit***	***Description***
_ Øinletnozzle	0.0360	*m*	Inlet nozzle diameter
_Øexhnozzle	0.0260	*m*	Exhaust nozzle diameter
_Øtozthroat	0.0072	*m*	Nozzle throat diameter
_Length Nozzle	0.0449	*m*	Nozzle divergent length
_Length conver	0.01664	*m*	Nozzle convergent length
_total length	0.06154	*m*	Total nozzle length
_α	12.00	*o*	Nozzle semi-angle of divergence
_β	45.00	*o*	Nozzle semi-angle of convergence



3.3.5 Nozzle simulation

We have done a simulation using a java applet[20] where introducing some of the nozzle parameters permits you to export graphs and data and shows you how the nozzle behaves. This is very useful because we will be able to see if our design meets the requirements that we want.

The purpose of this applet is to simulate the operation of a converging-diverging nozzle. This applet is intended to help to visualize the flow through a nozzle at a range of conditions.

The data introduced is:

- Ratio between the combustion chamber pressure and the exhaust nozzle pressure (0.00996)
- Average molar heat capacity of the mixture (1.1553 kg/K·mol)
- Optimum expansion ratio (13.1795)

The exported data is a huge table that gets values in small increments of the optimum expansion ratio, the Mach velocity and the ratio between outlet pressure and the combustion chamber pressure through the nozzle. It can also be seen the temperature distribution through the nozzle.

The following table is a reduced version focusing on the first value, the last (which refers to the nozzle exhaust) and the one in the middle that belongs to the throat.

Table 3.17 Exported data from the applet

INDEX	**OER**	**M**	**Pa/P0**
0	$1.338 \cdot 10^{1}$	$4.451 \cdot 10^{-2}$	$9.989 \cdot 10^{-1}$
27	$1.000 \cdot 10^{0}$	$1.000 \cdot 10^{0}$	$5.733 \cdot 10^{-1}$
146	$1.328 \cdot 10^{1}$	$3.329 \cdot 10^{0}$	$9.875 \cdot 10^{-3}$

Interpretation of the results:
On the applet simulation there are three panels that plot the flow conditions. The panel on the top shows the nozzle shape (see Figure 18) and the temperature distribution. The middle panel displays the pressure as a function of distance down the nozzle and the lower panel displays the Mach number as a function of distance.

[20] http://www.engapplets.vt.edu/fluids/CDnozzle/cdinfo.html

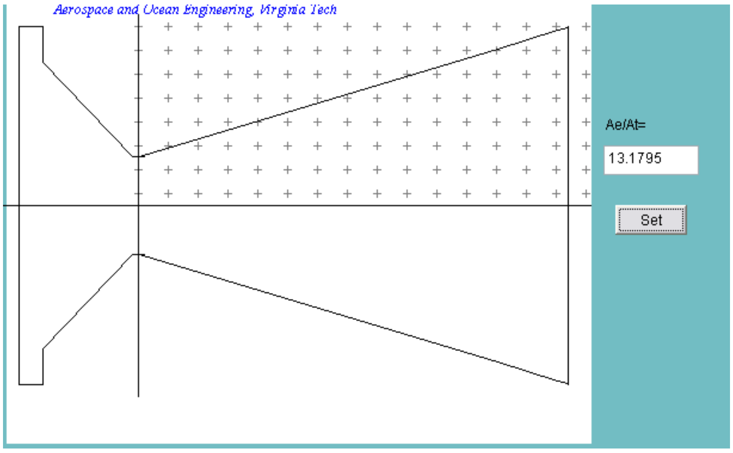


Figure 18 Nozzle shape simulation

When the flow accelerates (sub or supersonically) the pressure drops. On the simulation we can see how the flow is accelerating through the converging section till it reaches the throat. At this point the flow velocity is equal to M = 1 and after this moment the flow continues accelerating reaching supersonic velocity. The pressure continues dropping till the end of the nozzle where is reached the same pressure as in the ambient with a Mach velocity of M = 3.329. The temperature distribution is also seen at the top of the simulation.
To sum up, introducing some nozzle parameters, the simulation has shown us how the nozzle behaves (see Figure 19). With our design we have reached good results and the design condition seen before.

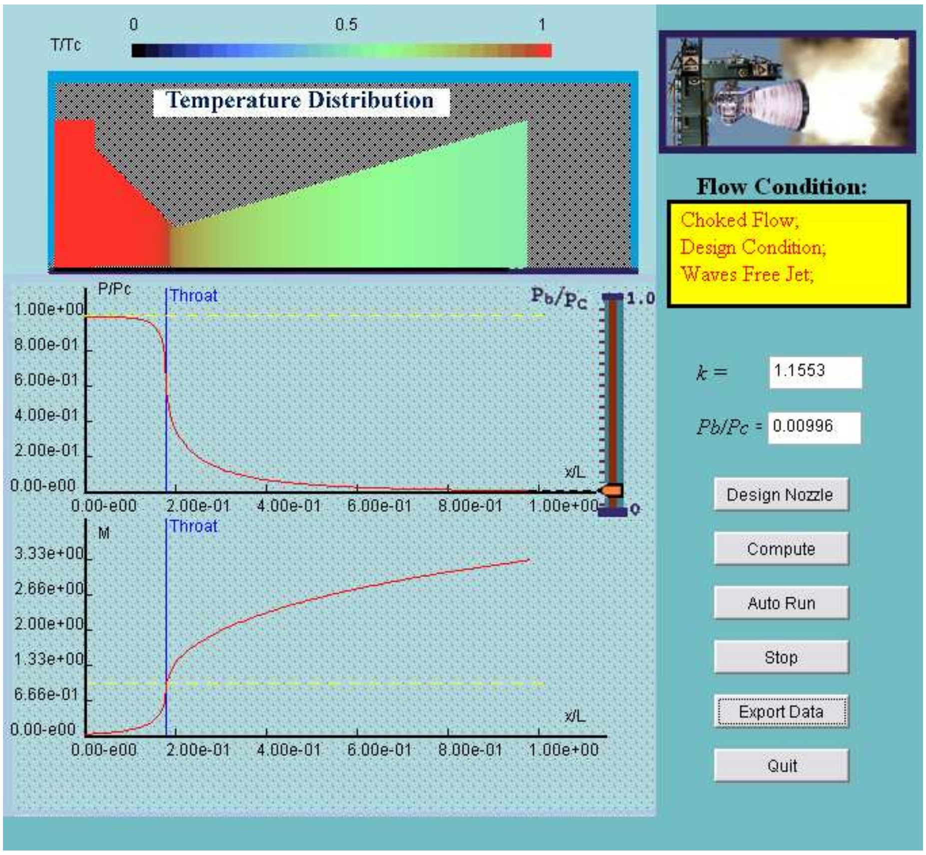


Figure 19 Simulation results

3.3.6 Nozzle Materials

The design of the second stage nozzle not only includes the shape and the size but also the materials.

The choice of materials for the nozzle is important for two reasons. On the one hand we have to bear in mind that the nozzle in most of solid rocket engines is one of the heaviest parts, so it has to be as light as possible. One the other hand is the component that will support most of the heating changes. That means that it has to be both light and temperature diffuser or resistant to high temperatures.

Until now, we have assumed for propulsion that no heat transfers from the system. This assumption has given good results because heat transfer consumes little energy compared to thrust. But in the designing of the nozzle is very significant.

The following table lists characteristics of typical refractory materials used for rocket applications including their response to high temperatures.

Table 3.18 Physical properties of typical refractory materials for rocket applications

Material	**Typical composition**	**Melting point (K)**	**Coefficient of thermal expansion ($10^{-6}K^{-1}$)**	**Thermal conductivity (W/m^2-K/m)**	**Specific heat (kcal/kg·K)**
Alumina	99.2%Al_2O_3; 0.6% SiO_2	2,322	4.0 at 283 to 1,280 K	2.45 at 0 to 1,273 K	0.28-0.30 at 290 to 2,000 K
Carbon	C	Sublimates 3,812	1.6 to 2.9 at 288 to 378 K	1.7 to 20 at 1,375 K	0.29 at 15 to 1,375 K
Graphite	C	2,300	1.0 to 2.6 at 288 to 378 K	35 to 206 at 1,373 K	0.29 at 15 to 1,283 K
Silicon Carbide	SiC	Decomposes at 2,483	3.7 at 288 to 1,773 K	16.3	0.23

Up to 500 ºC, the most used materials are aluminum alloys and fiberglass-resin composites, both of which have high-strength-to-weight ratios, are light in weight, easily fabricated, have good corrosion resistance, and are reasonable in cost. But when the temperature increases these materials main seem insufficient for a rocket nozzle.

There are different methods to protect rocket nozzles from absorbing excessive heat but the most used in amateur rocketry and one of the cheapest is to insulate the nozzle with materials with low conductivity, so heat flow to the walls decreases. One example are nozzles insulated with carbon/carbon composite material inserts that are able to maintain a high degree of integrity when there is an extremely rapid temperature increase in a high corrosive atmosphere.

In the nozzle throat, subjected to the most severe environment, are used erosion resistant materials. The most common materials are graphite and pyrolytic graphite. The stronger and more heat resistant pyrolytic graphite has high strength, high thermal conductivity, and low expansion. If graphite or pyrolytic graphite cannot be used because of nozzle size, the nozzle throat insert may be a refractory metal or alloy.

In the case of our low cost nozzle it have been studied various materials useful for the nozzle but in an easiest way we have decided to try to manufacture it with aluminum. In a first approach the aluminum properties cannot seem the best for our purpose but we will see it after the validation tests.

Table 3.19 Aluminum properties

PROPERTIES	
Symbol	Al
Phase	Solid
Element category	Other metal
Appearance	colorless oily liquid
Density	2,375 g/cm^3
Melting Point	660.32 ℃
Boiling point	2,519 ℃
Heat of fusion	10.71 kJ/mol
Heat of vaporization	294.0 kJ/mol
Specific heat capacity	(25 ℃) 24,200 J/mol·K
Crystal structure	Face centered cubic
Thermal conductivity	(17 ºC) 237 W/m·K
Young's modulus	70 GPa
Shear modulus	26 GPa

3.4 Low cost igniter study

This section is concerned with the mechanism or the process for initiating the combustion of a propellant grain.

Solid propellant ignition consists of a series of events:
- Receipt of a signal, usually electric
- Heat generation
- Transfer of the heat from the igniter to the grain surface
- Spread the flame over the surface area
- Elevate the chamber pressure

The device used to start this process is the igniter. Motor ignition must usually be complete in a fraction of a second.

3.4.1 Igniter definition

A device to initiate propellant burning, usually consisting of a small pyrotechnic device which produces sufficient heat and flame to cause the surface of the propellant grain to ignite[21].

3.4.2 Amateur and professional igniters

Practically all igniters employed in amateurs and professional SMRs are pyrogen or pyrotechnic variety. Pyrotechnic devices have extensive and reliable

[21] http://www.fas.org/man/dod-101/sys/missile/docs/RocketBasics.htm

heritage for space use, in a variety of different applications. In pyrogen igniters the igniter itself acts as a small SRM to provide gases and high pressures for igniting the entire inner surface of the grain simultaneously.

Pyrotechnic means a device which consists of a cartridge containing combustible powders in contact with an electrical resistance wire. Initiation of burning develops extremely hot flames, which spread to the solid propellant grain or pyrotechnic charge of the main igniter.

Pyrogen means a device in which combustion of the igniter material takes place within a closed chamber and the combustion products are exhausted into the rocket motor.

3.4.3 *Igniter design*

The igniter desing is a critical phase, especially in the second stage phase, because is a critical point. If the igniter fails the stage is not burned and the objective will not be achieved.

Based in this requirement we have tried to develop the igniter in two different ways and choose which is the best.

- Nichrome wire

Nichrome[22] is a brand for a wire of nickel-chromium resistance, a non-magnetic alloy of nickel and chromium. The most used nichrome alloy is about 80% nickel and 20% chromium, but there are many others to satisfy various applications. It is silver-gray in color, is resistant to corrosion, and has a high melting point of about 1,400 ℃. Because of its relatively high stre ngth and resistance to oxidation at high temperatures, is widely used in heating elements, such as in hair dryers, electric ovens and toasters.

Nichrome is used in explosives and fireworks industry as a reliable ignition system, such as electric matches and igniters model rocket. Other areas of application include motorcycle mufflers, and in a microbiological laboratory equipment.

[22] http://www.worldlingo.com/ma/enwiki/fr/Nichrome

Table 3.20 Nichrome properties

Material characteristics	Value	Units
Elasticity module	$2.2 \cdot 10^{11}$	Pa
Specific gravity	8.4	None
Density	8,400	kg/m^3
Melting point	1,400	℃
Electrical resistance at 20ºC	$1.08 \cdot 10^{-6}$	Ω·m
Specific heat	450	J/kg℃
Thermal conductivity	11.3	W/m·℃
Thermal expansion	$14 \cdot 10^{-6}$	m/m/℃

The design of a nichrome igniter is just a nichrome wire, also called bridge wire, attached across two lead wires at one end. That end is dipped in a chemical mixture that will burn easily, such as powder. Both lead wires will be connected to the 3V battery which the satellite works with. When a voltage is applied to the wire, the current flows and the wire heats up. It becomes so hot that burns the powder and starts the propellant ignition.

All the calculations are reduced to the battery we will apply, the diameter of the nichrome wire and the temperature that is about to reach. To do this, instead of calculating with Ohms law and other electric formulas we have applied a calculator[23] found in Jackob's online web (see Figure 20).

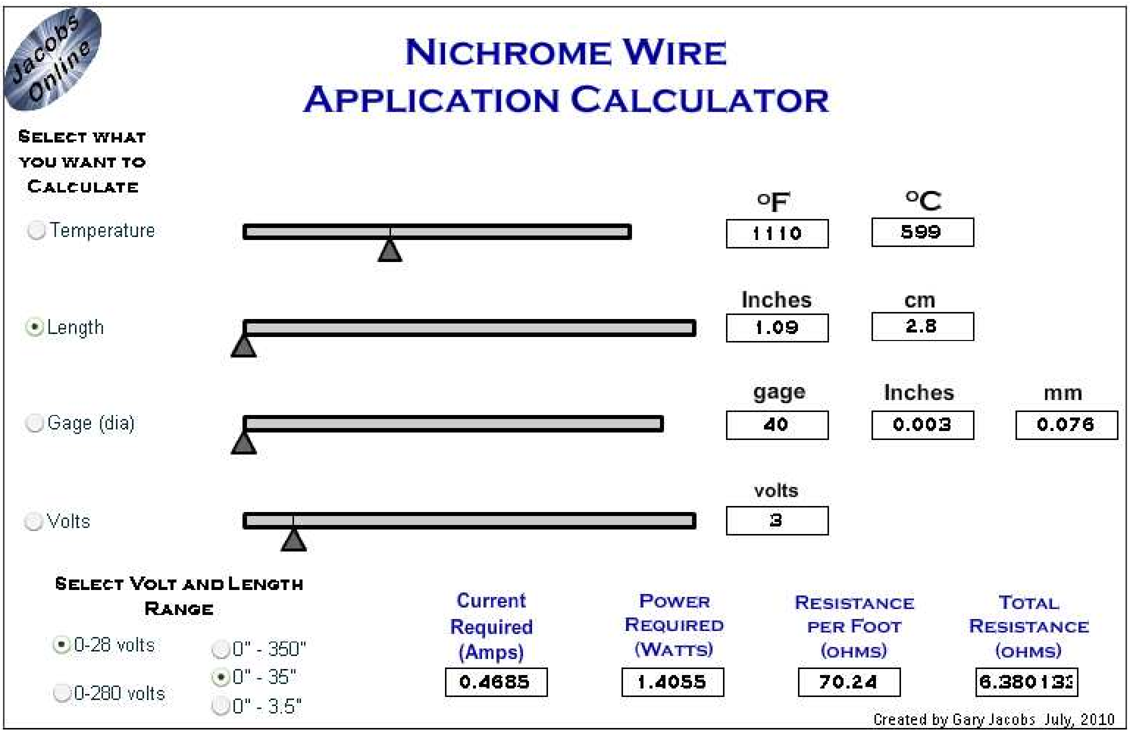


Figure 20 Nichrome wire application calculator

[23] http://www.jacobs-online.biz/nichrome/NichromeCalc.html

The nichrome wire used is about 0.076 mm of diameter. The temperature is the one needed to burn the powder and the volts the ones that the battery supplies. So the length is the parameter to be calculated by this application. The result is a wire of 2.8 cm long, difficult to manipulate but reasonably in terms of fabrication (see Figure 21).

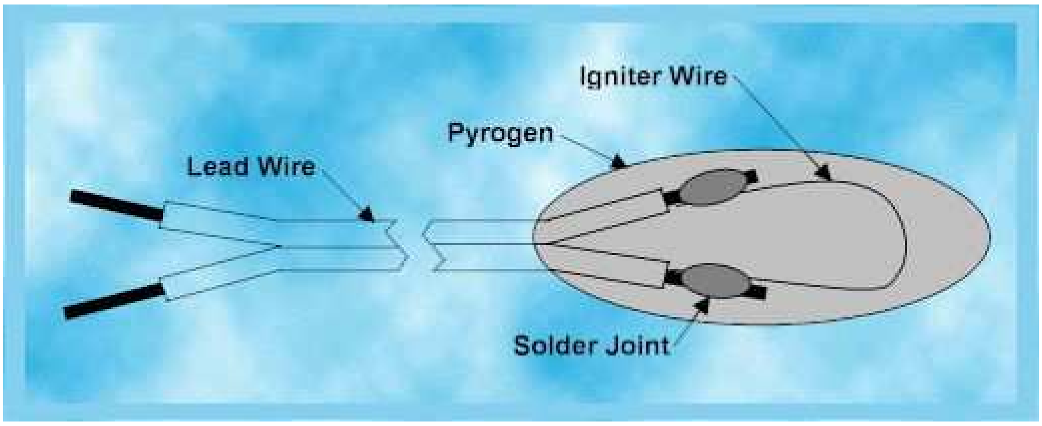


Figure 21 Igniter design

- Mini light bulbs

The second option is using low current bulbs. The design is an igniter that requires very low electrical power, reliable and especially useful in cold weather operation such as the second stage of our rocket.

These bulbs require 1.2 V and 20 mA to fire (see Table 3.21). When a voltage is applied to the bulb the filament heats up. So to fabricate the igniter we have to fill up a mini light bulb with powder, cover it with resin and then when the bulb is connected to a 3 V battery the filament will become so hot that will ignite the powder and all the bulb itself will explode and burn starting the propellant grain ignition.

Table 3.21 Bulb igniter materials

Material used	**Value**	**Units**
Mini light bulb	1.2	V
Powder		
Epoxy resin		

CHAPTER 4. NOZZLE CONSTRUCTION

In this chapter we will see all the development of the second stage construction. This involves the propellant, the structure itself, the igniter and of course the nozzle.

4.1 APCP synthesis

The APCP synthesis is a mixture of components, explained in section 3.1.4, that activates a reaction to form a kind of rubber. For safety reasons, the mixture will be done under inert atmosphere like nitrogen because Ammonium Perchlorate can react with the humidity in the air.

- First step is to activate the epoxy like Styrene and Phthalic anhydride adding 0.2% of a cathalyzer like Methyl ethyl ketone peroxide. From now and on, we only have few minutes to complete the mixture and casting process.
- Second step is to mix the Aluminum powder with the activated epoxy to avoid water ingestion from the air.
- Third step is to mix this with Ammonium Perchlorate that represents the 70% in mass.
- Fourth step is to cast the propellant inside the Stage2 structure trough the nozzle by gravity. Special care shoul be taken to avoid bubbles during this process. For this reason, vacuum is made inside the combustion chamber. The central region is generated using a shaft.

The reaction is exothermic and it reaches up to 70 ºC. After few minutes, the mixture is in gel state and there is no danger with humidity from then and on. The curing process remains for 3 days.

4.2 Structure assembly

The process for structure manufacturing is based on a can and epoxy. The epoxy layer is introduced inside a can, see Figure 22, when the catalyst is added but before the gel phase; the epoxy stays attached to the can by centrifuging the cam. The can is used to hold the pressure while it is really light. This can is made of steel and it is able to resist high temperatures.

The epoxy is used as an ablative material. In the vacuum there is no conductive path in order to cool down the hot temperature produced in the combustion chamber. Ablation is a property of a material that is able to consume heat by burn itself. The thickness of the layer inside the can is enough to resist during the burning time and it is about 1 to 3 mm. In real tests we saw how, after the burning time, this material remained burning few minutes which mean that still is able to absorb more heat. When we putted not enough material, the steel has changed its color and also in occasions a hole and explosion occurred like the fourth burn (See Section 5.3.2.).

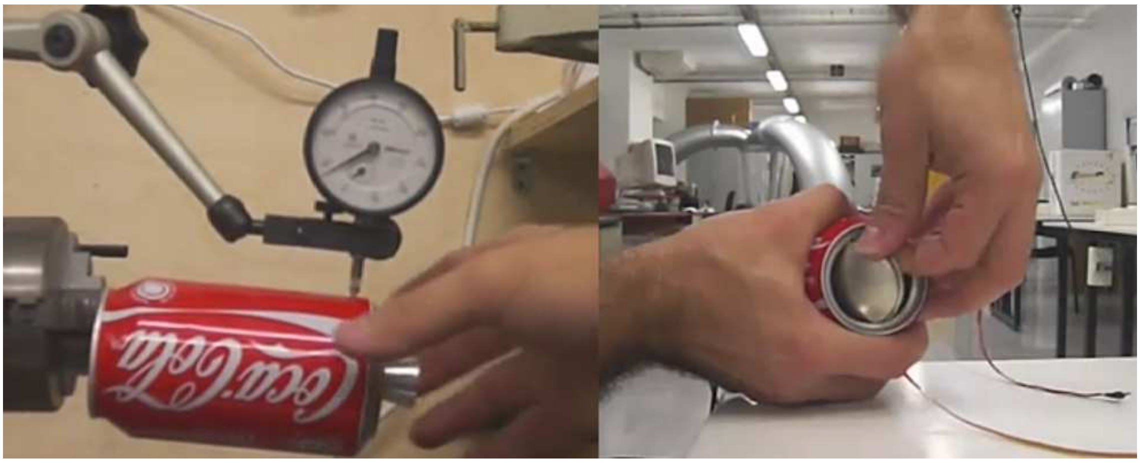


Figure 22 Centrifugation of epoxy and bulkhead to close the chamber

After the ablative material is cured, the can is filled by gravity means with propellant. To close the combustion chamber, see Figure 22, we use a bulkhead and sealed with extra epoxy.

4.3 Nozzle fabrication

We have used simple or amateurs manufacture techniques based on our cone shape nozzle design. This design is not the optimal such as with other shapes but for our purpose and bearing in mind the size of the nozzle is an easy way to do it. The throat manufacture could be made with graphite inserts but in a first approach we will do the entire nozzle with aluminum and then, after the validation tests, we will see if it is necessary.

The basic tool used for the nozzle manufacture is a lathe[24]. A lathe is a machine tool which rotates the work piece on its axis to perform various operations such as cutting, sanding, knurling, drilling, or deformation with tools that are applied to the work piece to create an object which has symmetry about an axis of rotation.

Before explaining the process, we have to say that the nozzle fabrication design won't be equal as the nozzle study design. One of the reasons is that one of the tools used for the diverging section has a minimum of 9 mm diameter. That means that the throat won't be as little as it is required because of that reason. On the second hand the aluminum cylinder used is about 30 mm diameter so the converging maximum diameter will be 30 mm long, not 40 mm as in the previous design. This parameter affects on the converging section longitude. It will be shorter, 3 mm approximately (see Figure 23 and Figure 24).

[24] http://en.wikipedia.org/wiki/Lathe

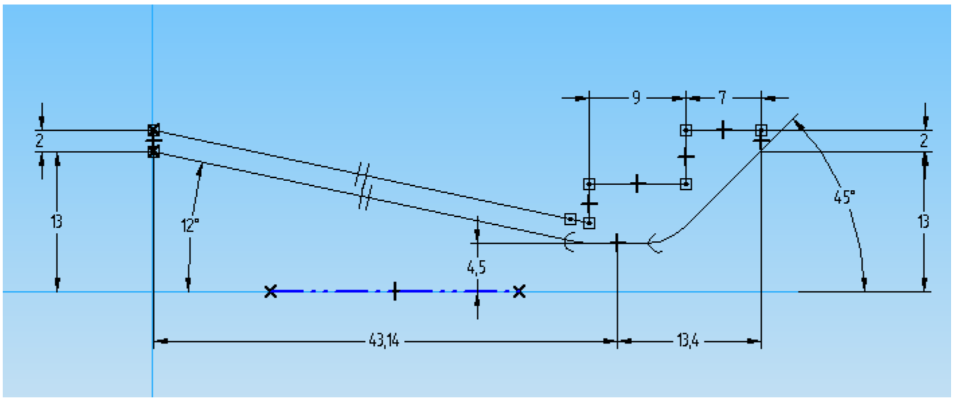


Figure 23 Nozzle size

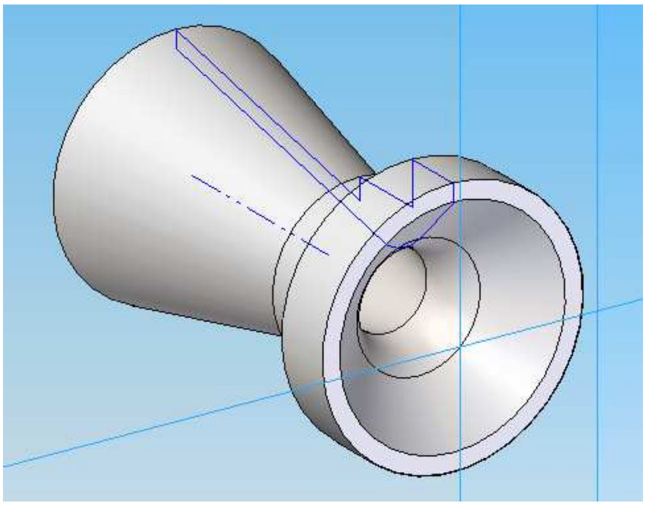


Figure 24 Nozzle

4.3.1 Nozzle fabrication procedure

1- Take a cylinder piece of aluminum of 95 mm of longitude and 30 mm diameter

2- Cut the cylinder obtaining the desired longitude (60 mm approximately plus 10 mm to hold it with the lathe) (see Figure 25)

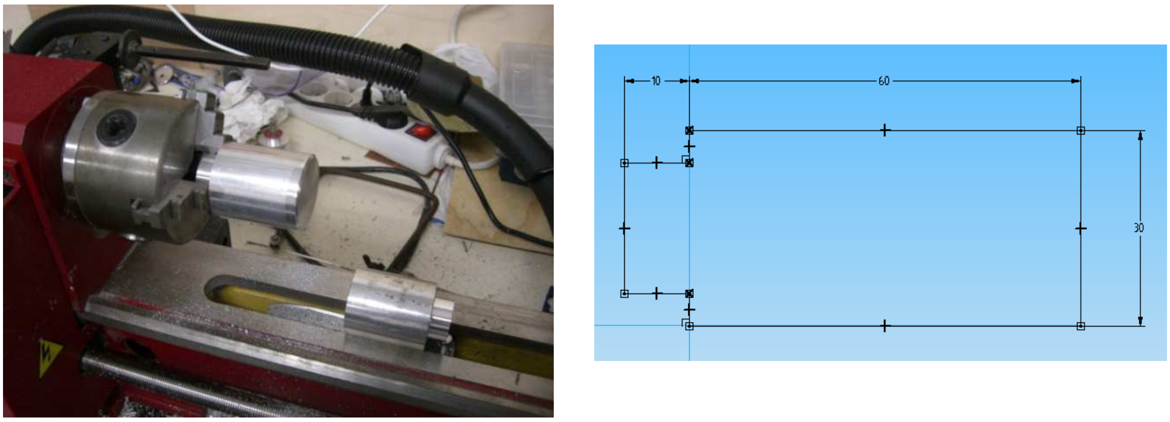


Figure 25 Cutting the aluminum cylinder

3- Use the radial saw to make the section flat and smooth the face cut. Now the aluminum cylinder is 59 mm long approximately

4- Take the lathe pointer and mark the center of the section and a 26mm diameter ring

5- Drill the center with various bits with a maximum diameter of 9 mm (see Figure 26)

Figure 26 Drilling the aluminum cylinder

6- Face the converging section of the nozzle

7- Begin to drill the converging section with an angle of 45º with a 45º tool (see Figure 27)

Figure 27 Nozzle convergent section manufacture

8- Change the tool for one better for internal surfaces. Finish the converging section till it reaches the throat

9- Surface polishing and sanding with scissors and sandpaper (see Figure 28)

Figure 28 Polishing of the convergent section of the nozzle

10-The throat is at 13mm approximately with a 9 mm diameter

11-Reduce the outer edge diameter to 20 mm from 7 mm to 16 mm length from the cylinder end

12-Reduce the outer diverging edge with a 12º semi-angle till it reaches a 41 mm long (see Figure 29)

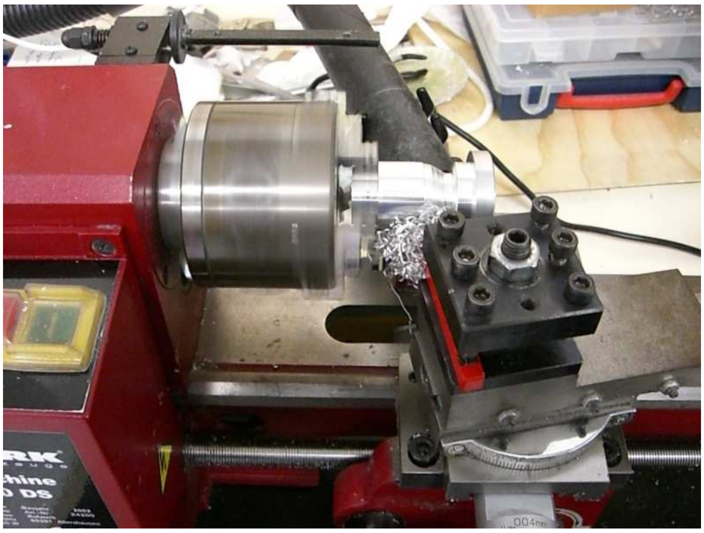


Figure 29 Outside of the divergent section manufacture

13-Cut the 10 mm used for holding the nozzle with a 2 mm margin. Now the nozzle is 57 mm long (see Figure 30)

Figure 30 Cutting the nozzle

14-For the diverging section drill the outlet with a 12º semi-angle conical tool

15-The conical part obtained is 44 mm long. The edge at the beginning of the nozzle is about 2 mm wide (see Figure 31)

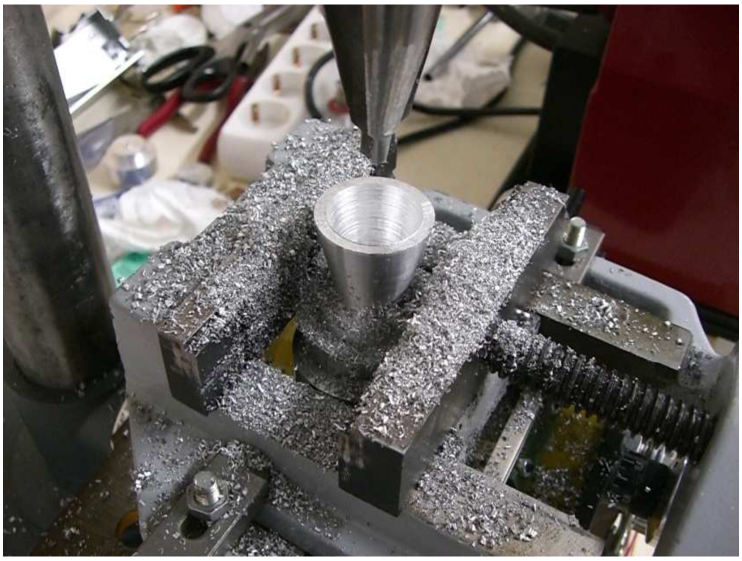


Figure 31 Drilling the divergent section

16-The nozzle is 57 mm long, very close to the design (see Figure 32)

Figure 32 Nozzle first two prototypes

At last, just say that for some future nozzle designs there is the option of adding graphite inserts in the nozzle throat. This modification will both make the nozzle optimal for thermal reasons and will set the design to the previous requirements because we can make the nozzle throat thinner with 7 mm diameter.

4.4 Igniter construction

We have seen in igniter design section two different ways to fabricate the igniter. On the one hand there is the nichrome wire option and on the other hand the one with light bulbs.

4.4.1 Nichrome igniter fabrication

We have followed the steps of fabrication from jackob's rocketry web side[25] and are explained bellow as follows:

1- Spread the two lead wires apart and wrap some wraps of nichrome wire around the bare end of the shorter wire
2- Solder both wires with tin or solder flux
3- Wrap nichrome wire around the insulated part of the larger lead wire leaving some distance between each wrap
4- Finish wrapping nichrome wire around the bare end of the larger wire and solder it too.
5- Now we have both lead wires connected by the nichrome wire (see Figure 33)
6- Once we have the soldering done we have to dip the wire into de pyrogen mixture (resin with powder)
7- Keep dipping until the wire is covered all along with the mixture

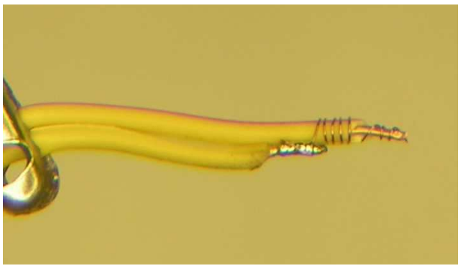


Figure 33 Nichrome wire heating test

Another variation for the previous tests consists on soldering the nichrome wire to two lead wires and keeping it in contact with powder to see if it burns (see Figure 34). We will see if it works.

[25] http://www.jacobsrocketry.com/aer/homemade_wire-wound_igniters.htm

Figure 34 Nichrome wire test variation

4.4.2 Mini light bulb igniter fabrication

The light bulb manufacture is easier and cleaner.

1- Check bulb continuity with the tester
2- Make a small hole on the top of the bulb (see Figure 35)

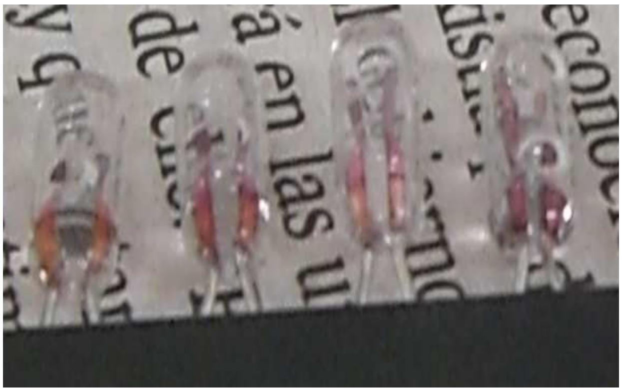


Figure 35 Light bulbs with holes

3- Carefully fill the bulb up with powder and the cover it with epoxy resin to keep the powder in (see Figure 36)

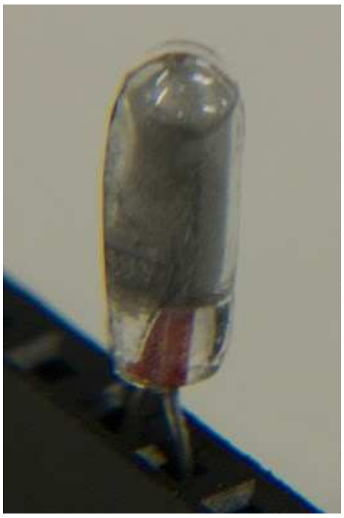


Figure 36 Light bulb full of powder and covered with resin

4- Recheck the continuity with the tester to see if the bulb filament it is okay.
5- Connect both bulb ends to two different wires to make the battery connection more simple (see Figure 37)

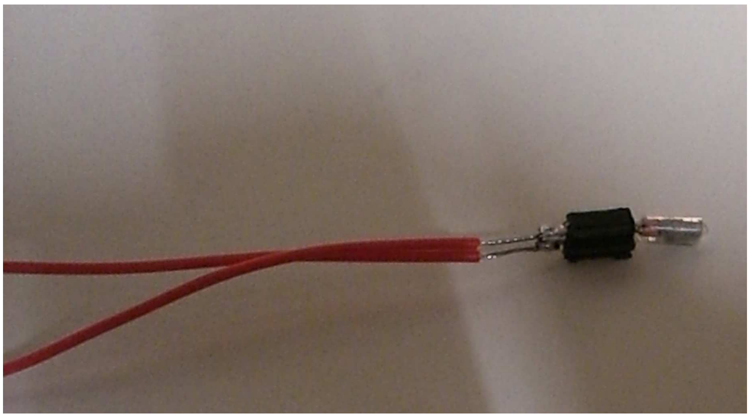


Figure 37 Igniter bulb

This design will be the one we will use because of its reliability, easy way of fabrication and results achieved.

CHAPTER 5. VALIDATION TESTS

5.1 Safety measures

Before explaining all the tests done to validate all the second stage design we are about to detail some safety measures followed during both the fabrication process and the validation tests.

5.1.1 *Prevention of harm to others*

To avoid accidents with harm to other people not involved on the tests we will proceed to establish some requirements:

- Try to signal the area to prevent others not to coming in
- Ensure that the test area is safe
- Create a radius of action and ensure that anyone outside it will enter the test area while some tests are developing

5.1.2 *Personal Protective Equipment*

Using Personal Protective Equipment, the worker is protected against external aggression that generates the performance of their work.

Collective protections are designed to reduce the corresponding risk while individual protections only act as complementary.
Personal protective equipment protects:

- Head
- Face and eyes
- Ear
- Abdomen
- Upper and lower limbs
- Skin
- Body in general

Some personal protective tools are:

- Helmet: mainly protects against head injuries cause by objects or any kind of projection or impact (see Figure 38)

Figure 38 Helmet

- Goggles: protect the eyes against particles impact, dust and dangerous radiation (see Figure 39)

Figure 39 Goggles

- Screen protection: plastic screens that protect when involving any work like polishing, drilling or manipulating any kind of tool that generates particles that can damage the face (see Figure 40)

Figure 40 Screen protection

- Mask: the objective is to filter the contaminants of the environment (see Figure 41)

Figure 41 Mask

- Hearing protectors: ideal to protect against high noise levels (see Figure 42)

Figure 42 Hearing protectors

- Gloves: protect against cuts, abrasions, bangs and radiation (see Figure 43)

Figure 43 Gloves

5.2 Check list

It is the procedure done each time we do a test. It is very useful because following all the steps we create a habit of safety and this way we don't forget anything important.

Here is an example of checklist used when proving the igniters with the first rocket prototypes:

- Check if the igniter continuity
- Place the igniter inside the amateur rocket motor
- Paste the igniter with epoxy resin, not on the top just a little bit on the bottom
- Connect the igniter to the wire through the hole done in the PVC tube
- Carefully insert the rocket in the PVC tube, sliding it to the bottom checking every moment there is no wire inside the tube
- Now we have the igniter connected and the rocket inside the PVC tube
- Insert the 9 V battery connector
- Turn on the ground cameras
- Turn on the camera placed in the tube
- Connect the 9 V battery

5.3 Tests[26]

5.3.1 Igniter tests

When testing igniters we have done first some validation tests to see which was better for our rocket, nichrome igniter or the mini light bulb.

5.3.1.1 Nichrome igniter

- **Nichrome igniter test 1**

Test: Check if nichrome wire reaches enough temperature

Materials: Nichrome wire with several lengths, a 9 V battery and lead wire.

Procedure: Solder the nichrome wire to both lead wires and wrap it to the insulation part. Connect to the 9 V battery and see if it melts.

26 http://www.blogger.com/i.g?inviteID=8042331534884948298&blogID=5628330591042660474

Test results: We tried it several times with several wire lengths and just in a few cases the insulation part melted.

- **Nichrome igniter test 2**

Test: Check if nichrome wire reaches enough temperature to burn powder with just being in contact.

Materials: Nichrome wire with several lengths, a 9 V battery, lead wire, powder and a plastic cap.

Procedure: Solder the nichrome wire to both lead wires and maintain it in contact with some powder deposited in a plastic cap. Connect to a 9 V battery and see if it burns.

Test results: We tried it several times with several wire lengths and the powder did not burn.

5.3.1.2 Mini light bulb

- **Mini light bulb test 1**

Test: Check if the bulb explodes with some powder inside

Materials: 1.2 V Bulb, a 9 V battery, powder and lead wire.

Procedure: Insert some powder into the bulb. Cover the hole with resin and connect the bulb to a 9 V battery. The extra current melts the bulb filament and causes the powder burning and so the bulb explodes.

Test results: We tried it several times with different quantity of powder. The best result obtained is filling up the bulb with the maximum quantity of powder and covering it carefully with the less resin possible, just the necessary to close the hole. That way is the optimal for reaching a 100% of reliability. After these tests (see Figure 44) we decided to use bulbs because of its simplicity, reliability and cost. Next validation test is to check if the igniter burns the rocket propellant.

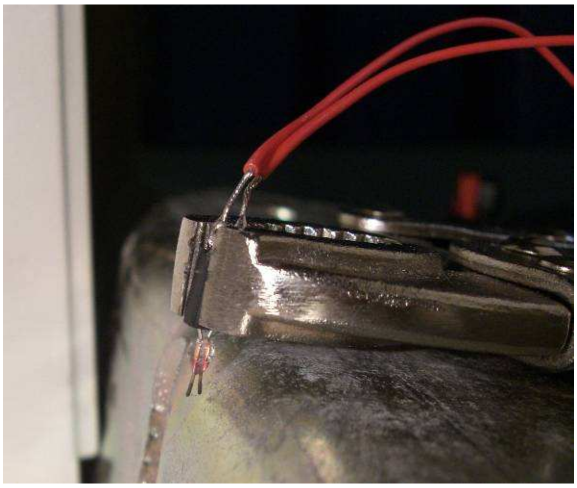


Figure 44 Igniter bulb test

- **Mini light bulb test 2**

Test: Check if the igniter works with amateur rocket motor

Materials: The rocket consists in a soda can and a semispherical crystal (see Figure 45), a weco D7-3 engine as propellant, an igniter, a PVC tube, an electrical 1,5 m wire, a 9 V battery and a tester.

Figure 45 Igniter test with amateur rocket engine

Test results: The igniter worked perfectly, the stage burn and lift of the tube. When approaching to the peak altitude the stage start making loops until it begins to fall in a straight line, this shows that the stage has a pretty good stability.

- **Mini light bulb test 3**

Test: Test if the igniter works with amateur rocket motor. This test is practically the same as Test 3 but in this case there are some chamber construction parameters changed.

Materials: The rocket consists in a soda can and a semispherical crystal, a weco D7-3 engine as propellant, an igniter, a PVC tube, an electrical 1,5 m wire, a 9 V battery and a tester.

Test results: The igniter worked perfectly, the stage burn and lift of the tube about 15 or 20 m.

5.3.2 Combustion chamber and nozzle burns

- **Stage 2 first burn**

Test: Testing the new igniter connection, an amateur engine support and a prototype nozzle (see Figure 46).

Materials: It is used a standard steel support screwed to the floor for safety reasons and the amateur second stage motor.

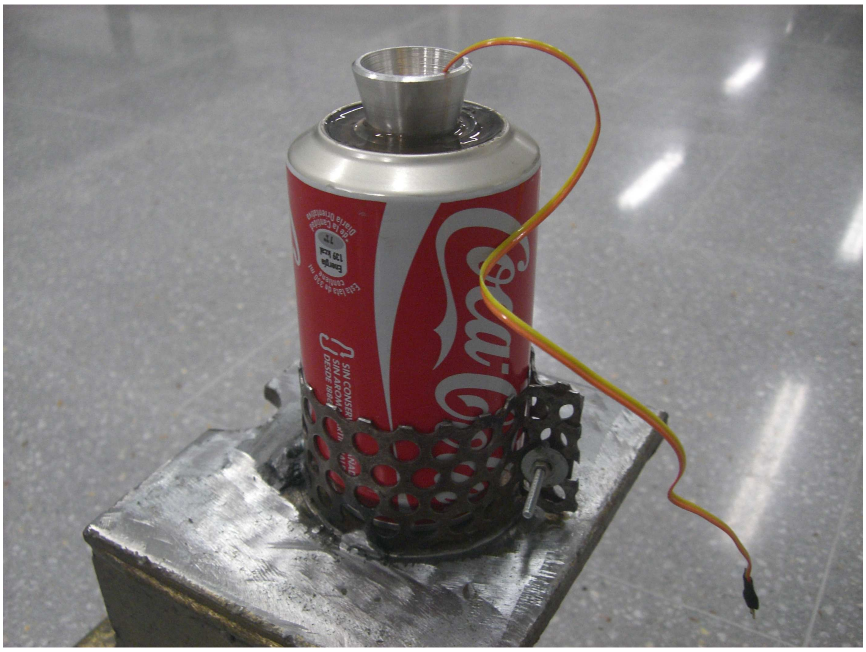


Figure 46 Stage 2 first burn prototype

Test result: The igniter works perfectly, the engine is on and on for about 3 seconds, the can has a deformation because of heating reasons (see Figure 47). The nozzle worked perfectly, with no deformation, no melting and a leaving a good burn flow.

Figure 47 Stage 2 first burn prototype burned

- **Stage 2 second burn**

Test: Take the can and the nozzle to the limit (see Figure 49). To do this we have add three more amateur engines to the can (see Figure 48). Then analyze the results.

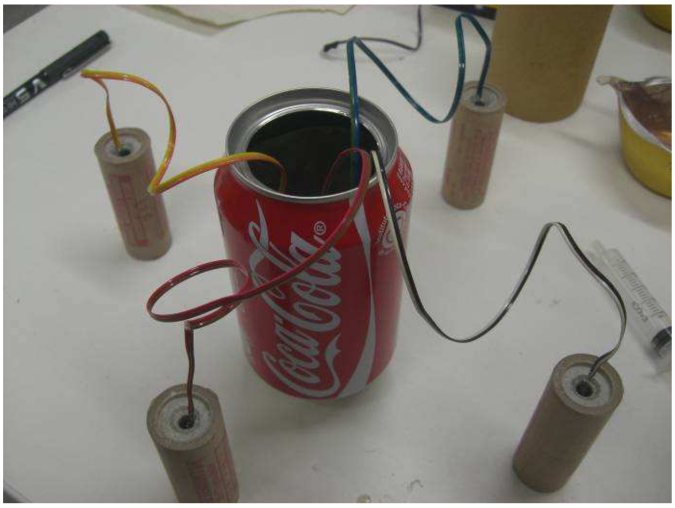


Figure 48 Stage 2 second burn with four engines

Materials: It is used a standard steel support screwed to the floor for safety reasons and the amateur second stage motor with four engines inside.

Figure 49 Stage 2 second burn prototype

Test result: The igniters work perfectly and their reliability is proved. The engine burns during 3 seconds. The can has a deformation and a clean cut is seen after the first second of combustion (see Figure 50). The nozzle works well. We have to improve the way of placing the engines inside the combustion chamber. Also modify the engines because have their own nozzle and that causes a bad flow and deterioration in the nozzle.

Figure 50 Stage 2 second burn with four engines

- **Stage 2 third burn**

Following the validation of the stage 2 we got to the third version of it. In this third prototype we decided to use 7 grains (one in the middle with its encapsulation and its nozzle and 6 around without neither encapsulation nor nozzle). With this configuration we reach the maximum capacity of the can. Only the central grain will star burning with the igniter and the other 6 are expected to burn thanks to the heat, pressure and fire coming from the central one. The can is coated with epoxy resin in order to support higher temperatures.

Test: Take the can and the nozzle to the limit. To do this we have 7 engines inside the can (see Figure 51). Then analyze the results.

Materials: It is used a standard steel support screwed to the floor for safety reasons and the amateur second stage motor with seven engines inside.

Test result: The igniter works and lights up the stage. A few moments since the ignition starts the stage explodes and the grains that were not held do not burn.

Figure 51 Stage 2 third burn prototype

- **Stage 2 fourth burn**

We keep testing the stage 2 and in this fourth test we put 7 grain, one wrapped with cardboard and the other six covered with resin. Once placed all the grains, the can is filled up with more resin to hold all grains and to create a solid structure.

Test: Take the can and the nozzle again to the limit in order to achieve a stable and reliable stage to reach space. To do this we have 7 engines inside the can. Then analyze the results.

Materials: It is used a standard steel support screwed to the floor for safety reasons and the amateur second stage motor with seven engines inside held together with resin.

Test result: The igniter works and lights up the stage. The stage burns during 6 seconds. A leak appears and ends in an explosion. The can breaks in a curious way, a nearly perfect cut in the lateral (see Figure 52). The resin gives more pressure and temperature resistance. The flow seems perfectly regular and the nozzle, once cleaned, looks like new without any deformation (see Figure 53). Aluminum works.

Figure 52 Stage 2 fourth burn prototype burned

Figure 53 Nozzle cleaned after the test

- **Stage 2 fifth burn**

With this test we reach the fifth prototype. This stage consists of 6 engine grains coated with glass fiber and resin to achieve a better heating protection of the can (see Figure 54).

Test: Take the can and the nozzle again to the limit in order to achieve a stable and reliable stage to reach space. To do this we have 6 engines inside the can. Then analyze the results.

Figure 54 Stage 2 fifth burn prototype

Materials: It is used a standard steel support screwed to the floor for safety reasons and the amateur second stage motor with seven engines inside held together with resin.

Test result: The combustion chamber remains intact for the first time to a more than one grain configuration. The nozzle releases because the resin that holds it burns (see Figure 55). The flow is not uniform because the grains do not burn exactly at the same time.

Figure 55 Stage 2 fifth burn prototype burned

CHAPTER 6. CONCLUSIONS

6.1 General conclusions

In the present paper the second stage design, construction and validation for a low cost mini-launcher has been analyzed.

There has been a previous study of materials and high performance propellants. Then the second stage nozzle design has been developed based on thermodynamic laws and on the best propellant configuration. Moon 2.0 has been a good tool to know all these parameters.

After doing the design we have manufactured the nozzle and the other components of the second stage with low cost materials and a lot of effort and persistence.

The tests have shown us that:

- The low cost nozzle manufacturing model seems to be useful and suitable for the mini-launcher project. The weak point would be its efficiency
- The aluminum nozzle works well for burning rates less than 3 seconds. If we want to increase this time, the best option is using graphite inserts in the nozzle throat. Graphite good thermal properties will allow the nozzle support higher temperatures and longer burning rates
- The mini bulb igniters have been proved to be very useful, reliable, very cheap and easy manufacturing
- Solid propellants like APCP are quite good for this mission but very difficult to obtain from the amateur point of view

We have shown that it is possible:

- to design, test, and build a second stage nozzle for a low cost mini-launcher with ordinary materials and resources
- to design, test, and build a reliable igniter for a the second stage with ordinary materials and resources
- to design, test, and build a second stage motor for a low cost mini-launcher with ordinary materials and resources

6.2 Environmental impact

The development of this project based on searching technologies and materials with a low cost, has bring us to use all kind of resources, especially most of them ordinary items and materials.

We have tried to optimize resources in order to reduce the impact to the environment. Waste the lowest electricity as possible and reusing materials in the development.

Referring to the launch the environmental impact caused by the propellant would be the minimum if we bear in mind that, as said in section 3.1.4, just one component resulting from the APCP burning can be damaging. This component is Al_2O_3, the only one in liquid state, and it is about 0.34579 moles per 100 grams. This is an acceptable quantity and there is no need to take it into account.

On the other hand there is other kind of stuff that generates waste. We are talking about all the waste generated during the fabrication process.
When manufacturing the second stage all the useless materials are residues. Here we are referring to the aluminum particles, powder, glass fibers, wires, bulbs, resin, soda cans and all this kind of components used for all the tests and manufacturing.

Because of the sizes we are working with, this project doesn’t mean a significant impact upon the environment. The small size of the second stage assures that it will be disintegrated during the re-entry in the atmosphere and the pollution generated is really low compared with a big launcher. That means that the use of small second stage motors reduces pollution and costs.

CHAPTER 7. BIBLIOGRAPHY

[1] Tristancho, J., *Implementation of a femto-satellite and a mini-launcher*, upcommons, 2010

[2] Humble, R.W., Henry, G.N., Larson, W.J., *Space propulsion analysis and design*, McGraw-Hill, New York, 1995

[3] Baker, A., Dutton, S., Kelly, D.W., *Composite materials for aircraft structures*, American Institute of Aeronautics and Astronautics, Reston, 2004

[4] Man, H.C., Duan, J., Yue, T.M., "Design and Characteristic Analysis of Supersonic Nozzles for High Gas Pressure Laser Cutting", *Elsevier*, Hong Kong, 1997

[5] Guobiao, C., Dingqiang, Z., "Coupling Simulation of Heat Transfer and Temperature of the Composite Walled Nozzle of Rocket", School of Astronautics, Beijing University of Aeronautics and Astronautics, China

[6] Soukhanovskii, V.A., Kugel, H.W., Kaita, R., Majeski, R., Roquemore, A. L., "Supersonic gas injector for fueling and diagnostic applications on the National Spherical Torus Experiment", Lawrence Livermore National Laboratory, Livermore, California and Princeton Plasma Physics Laboratory, Princeton, New Jersey, 2004

[7] Stine, G.H., *Handbook of model rocketry*, J. Wiley, N.J., 2004

[8] Sutton, G.P., *Rocket propulsion elements*, John Wiley & Sons, New York, 2001

[9] Hill, P.G., *Mechanics and thermodynamics of propulsion*, Addison-Wesley, Reading, Massachusetts, 1992

[10] Turner, M.J.L., *Rocket and spacecraft propulsion: principles, practice and new developments,* Springer-Verlag, New York, 2004

[11] Mattingly, J.D., *Elements of gas turbine propulsion,* McGraw-Hill, New York, 1996