MASTER THESIS

Implementation of a femto-satellite and a mini-launcher

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BY

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ABSTRACT

In this Master Thesis we begin with a short analysis of the current space market, with the aim of searching solutions that allow us to implement femto-satellites (that is, satellites with a mass less than 100 grams) and mini-launchers (in this case less than 100 kilograms).

New synergies will be explored in order to reduce drastically the cost of development, construction, operation and disposal of femto-satellites and mini-launchers for operations in LEO (Low Earth Orbits below 300 kilometers of altitude) and short duration, about one week. http://code.google.com/p/moon-20

Two application examples based on current technologies that are not usually space compliance will be shown. The first example pretends to fulfill the N-Prize requirements and it is called WikiSat. The second example is an Earth observation application called EPSCSat.

The N-Prize is a competition that consists in putting into orbit a satellite of less than 20 grams with less than 10,000 sterling pounds of launch cost. This tiny satellite shall be tracked for at least nine turns around the Earth at a height in excess of 100 kilometers above the ground to qualify for the N-Prize.

The femto-satellite design will be extended to the EPSCSat and it is presented as an additional application of this work. The EPSCSat is a satellite with a camera, an optic for Earth observation and a fast download link thanks to the use of a radio-link in the S-Band.

Keywords: Femto-satellite, Mini-Launcher, Space industry, N-Prize, Google Lunar X-Prize, WikiSat, EPSCSat, WikiLaucher, PicoRover

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INTRODUCTION

The space industry is one of the most expensive and demanding commercial sectors [6]. From its very beginning, the space industry has been strongly developed and supported by governmental organizations, as was the case for the aviation industry. For many reasons, the space industry remains controlled by federal organizations, following a quite different evolution in comparison to the aviation industry, contrary to expectations in the early 1960s. One of these reasons is the incompatibility of civilian interests and developments with the space strategic sector or the national security issues, since the space industry was improved by the military sector and has many double-use technologies. In the USA, the ITAR[1] regulations were enacted; these regulations severely restrict the export of both technological hardware and information to foreign countries, as well as limits the access of non-US individuals to such technologies. In the space industry, this results in the fact that US citizens have many restrictions to collaborate with foreign companies. Nonetheless, thanks to the Internet the flow of information is overcoming many of the ITAR regulations, despite the efforts to the contrary by the US government, as well as many other industrialized countries.

The components of the space industry require a qualification for the space. The space environment is very exigent and not all materials are suitable for this environment. The conditions of the space environment are very different from the earth environment ones. Radiation makes unusable materials like plastics. Electronic components can have an unpredictable behavior and they must be shielded. Safety protocols like code error correction have to be implemented. Thermal gradients are wide, the temperature changes fast and beyond the usual ranges inside the atmosphere. Due to the high vacuum condition, outgassing occurs in every material[11], turning weak every day that are exposed to vacuum. Insulators play an important role in this sense, but in addition thermal flow works in a different way: radiators and heaters are part of the main thermal subsystems because in many cases there is no atmosphere to dissipate the excess of heat.

The energy in the space is other issue to overcome. It is mainly provided by solar cells, a technology that is very well developed in terms of efficiency and mass. When this source is not available in the space, for example during the eclipse, electrical power is usually provided by batteries, which are another good example of very well-developed technology. In the deep space, energy is limited and new energy sources are improved by nuclear means. For example the IR radiation of any planet can provide energy using thermoelectric cells.

The satellites are usually classified by their mass as shown in Figure 1. Femto-satellites have less than 100 grams of mass. We define a mini-launcher as a vehicle with less than 100 kilograms able to transport to the space a small payload.

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[1]ITAR. International Traffic in Arms Regulations. http://en.wikipedia.org/wiki/ITAR
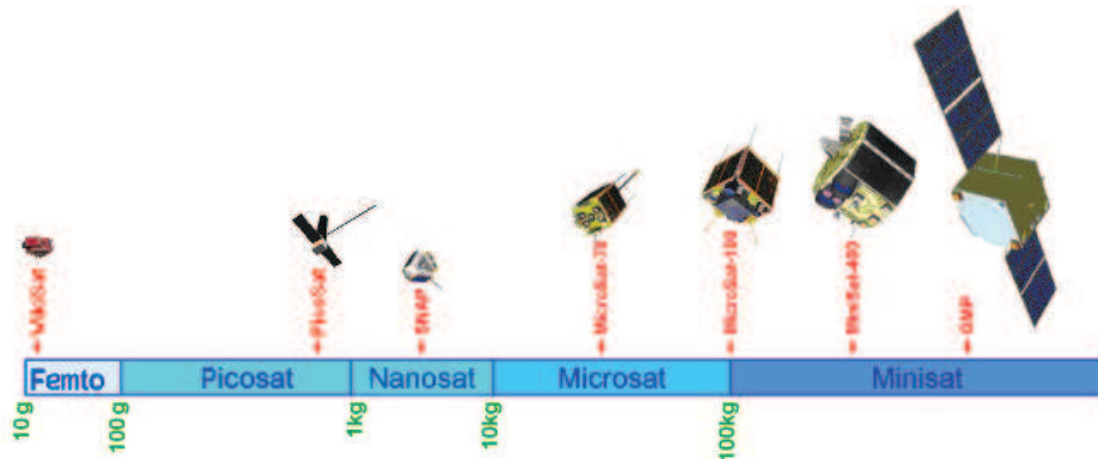
Figure 1: Scale of satellites as a function of mass

Current launchers are so expensive that the cost of the satellite is relatively low in relation to them. For this reason, it is very common that a launcher has a main client, except for a mission based on constellations or piggy-back opportunities. Even the current technologies (also called drivers) allow to build very small and light satellites for the most kinds of missions, but the main client do not want to assume the risk when a small satellite can interfere with the main mission. In addition, a layer of high reliable components, called space qualified components, increases the weight and only some providers can build these components. This situation makes that the space sector is handled only by very few companies in such a way that is very difficult to compete against these companies. The space sector remains in a kind of monopoly situation, where competitiveness is diminished.

In this work we propose to break this slow trend that the space sector has kept for half a century.

In chapter 1 we will present a novel selection of Commercial-off-the-shelf (COTS) parts that allows to have a complete satellite for less than 100 grams and at a very low cost still maintaining a good reliability. We will study the parameters that affect the launch cost of a small launcher when is used for a complex mission like putting a lunar rover explorer in the Moon. New technological drivers and methodological approaches will be presented as well as synergies for low cost missions.

In chapter 2 we will present two examples of satellites using this COTS components: the *WikiSat* and the *EPSCSat*, which are in the scale of femto-satellites, as shown in table 1.

In chapter 3 we will introduce a *mini-launcher* able to put in Low Earth Orbit small payloads with a cost below the barrier of one million dollars. A low cost solid propellant process will be presented to reduce the manufacturing cost. A few number of tools will be required to speed up the development time. A good example is a full cycle simulator from the propellant design, the launcher physical parameters, staging and trajectories, until the ground coverage; with hardware in the loop able to test real hardware.

Table 1: Nomenclature for small satellites

Group name	Wet mass	Example
Satellite classification		
Large satellite	greater than 1000 kg	-
Medium satellite	500 to 1000 kg	GMP
Mini satellite	100 to 500 kg	MiniSat400
Micro satellite	10 to 100 kg	MiniSat100
Nano satellite	1 to 10 kg	SNAP
Small satellites		
Pico satellite	0.1 to 1 kg	CubeSAT
Femto satellite	less than 100 g	WikiSat

CHAPTER 1. A NEW MARKET FOR FEMTO-SATELLITES

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

Until recently, space missions were the sole province of huge companies and government organizations. Aerospace corporations were hosted and feed mostly by the administration, and often were highly focused on military applications. Due to the enormous cost of launch (caused by the use of expendable rockets or, even worse, of the Space Shuttle), American and European agencies based space developments on highly reliable satellites, which were also very costly. Russia and India[1] based space developments in lower budgets, but at the cost of an increased rate of failures.

When the space industry was dedicated to civil applications like the space exploration, its inertia was strong, making it difficult to change the business model. This is one of the reasons for the dissimilitude between the progresses on aviation industry and space industry. To overcome this limitations, new solutions must be offered to space mission designers, that nowadays are limited to a very small choice of launchers in a market that is ruled by a few companies, and which is furthermore subject to legal and political issues that make it difficult to pick the right launcher for a mission.

*A*ccess to space should not only be granted in grounds of freedom, but also should be cheap, even if current space industry solutions make it seem expensive.

This chapter is an effort to find new technologies suitable for space. We want to focus this effort in short missions (like CubeSat missions) enough to validate the model, because afterwards it will be easy to change the business model and to emigrate to larger missions. In this chapter, new synergies are explored in order to dramatically reduce the cost of development, construction, operation and disposal of femto-satellites and mini-launchers for operations in LEO (Low Earth Orbits below 300 kilometers of altitude) and short duration, less than one month. In this regard, we will analyze parameters that are relevant for such missions. A list of mission drivers will be presented as well as a list of Commercial-Of-The-Shelf (COTS) components that will be used in the femto-satellite and in the mini-launcher

We have grown in Castelldefels (Barcelona) a group called *PicoRover* formed by teachers, students and collaborators with a clear, elevating goal[7]: "*Go to the Moon*". Promoted by a few universities like the *UPC* University in Catalonia or the *Antonio Nebrija* University in Madrid, this group is a part of the *Team FREDNET* contestant for the Google Lunar X-Prize. Members are self-motivated by the idea of being a part of this project while basic needs are provided by a secular job different from this.

The main characteristic of our group is the potential to develop very fast, complex products with a very low cost because we are in the state of the art. We adapt our development to the market needs because we can use the latest available new technologies. In this line, we are contestant for prizes like *N-Prize* or the *Google Lunar X-Prize*. We use the

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[1] http://www.pwc.com/in/en/press-releases/Aerospace-Industry-in-India-fastest-growing.jhtml

prize rules as a requirements for our missions which in our case are not commercially constrained. We are not committed to make money wining the prize, but we will try to make money developing new products inside local industries.

The N-Prize is a competition promoted by the British citizen *Paul H. Dear* that consists in putting into orbit a satellite of less than 20 grams with less than 10,000 sterling pounds of launch cost. This tiny satellite shall be tracked for at least nine turns around the Earth at a height in excess of 100 kilometers above the ground to qualify for the N-Prize. The prize will be available for entrants whose satellites complete their 9th orbit before 19:19:09 (GMT) on the 19th September 2011.

The Google Lunar X-Prize, promoted by the American company *Google*, consist of two prizes and a extra bonus: The First Prize contestant will win $20 million if their system reaches the lunar surface, travels 500 meters and transmits "*Mooncasts*" including HD video before December 31, 2012. If not won by then, that prize drops to $15 million until terminating on December 31, 2014. The Second Prize is $5 million during the whole period until that date. Bonuses to prizes (totaling another $5 million) are available if more than 5 km is traveled; if human artifacts are imaged; or if the system survives the lunar night.

1.1.. New solutions for femto-satellites and mini-launchers

In this section, a list of components is compiled as a result of a thorough study of the current consumer market. These components allow us to improve performances in weight, robustness and accuracy for some space applications. These components are not space compliant, at least until their validation. If so, instead of using very well-known and rather old technology, we propose to validate these new components and to improve the space market in regard to those old components. This new approach implies taking higher risks during the actual mission, but the reduction in cost and complexity of the satellites, and its faster development -as well as the faster development of space technology allowed by it- likely more than compensates for it. It seems natural that small companies are more well–suited for this change in business model. In the Annex B there is a list of local Catalonian companies related with the space industry that could benefit of this new paradigm.

Some examples of these components are given in the next few paragraphs:

Batteries. Low weight, high specific power LiPo *(Lithium–ion Polymer)* rechargeable batteries. As an example, the TP250-1SJPL2 battery[2] has a high power density of $1.8\,watts$ per gram (total weight 10 grams) and high energy density ($0.25\,Ah$) in a volume of 40 x 20 x $5\,mm$. The SCiB[3] battery is a high power density of more than 1 Watt per gram (total weight 150 grams) and high energy density ($4.2\,Ah$) in a volume of 62 x 95 x $13\,mm$. This battery allows a recharge with a current as large as 50 amperes, in 5 minutes. It operates well in extreme temperatures, with sufficient discharge at temperatures as low as $-30\,^{\circ}C$.

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[2]LiPo http://www.rctoys.com/rc-products/TP-250-1SPLJ2.html
[3]SCiB http://www.toshiba.co.jp/about/press/2007_12/pr1101.htm

Solar cells. High efficiency Ultra Joint solar cells[4] reach up to 28% of efficiency and weight only 8.4 grams per square decimeter. UTJ is one of the best power sources for satellites. This component is restricted by ITAR and can not be exported from the USA.

IMUs. Low weight high accuracy IMU[5] (Inertial Measurement Unit) giving up to $10\,G$ (on 3 axes) accelerations sensing and 3 gyros up to 400 degrees per second of angular rate sensing. This kind of devices weights less than 7 grams in an embedded board with a volume of 32 x 31 x 16 millimeters. Maximum current is only 27 milliamperes at 5 volts and also it has a build-in temperature sensor. This component is restricted by ITAR and can not be exported from the USA.

SMD. Also called surface mount devices (polyesther embedded systems) present a high level of integration, very low weight and consumptions. These are cheap and easily available.

Solid state memory. High memory capacity with passive storage up to 10 years. The USB 2.0 superStick[6] has up to 4 Gigabytes in a volume of 34 x 12 x 2 millimeters. This memory only weights 1.62 grams and is resistant to ESD (Electrostatic Discharges) up to 4 kilovolts.

3D printers. Fast prototyping techniques by 3D direct printing structure. Also useful for casting of aluminum and other low melting point metals.

Insulators. High efficiency insulator materials like Aerogel[7] and Polyurethane FOAM[8] are commercially available. This kind of materials is very light, having a density of 20 grams per cubic meter! Light FOAM is very easy to apply and gives a good thermal and corrosion protection to the main components. Also wax is used in order to give a thermal protection to embedded components.

Motors. Some sensor movements or deployments are required in the space. There is a full family of micro-motors for micro-robotics and space applications. For example, the micro-stepper-motors like the 6415 SONCEBOZ[9] which gives 3.2 Millinewtons-meters of torque in a volume of 30 x 28 x 14 millimeters and weights 13 grams. Other example of high integration brushed motor is the A-max[10] which reaches up to $6000\,rpm$ with 1 Millinewton-meter of torque and 0.5 watts. This micro-motor weights 12 grams and has a volume of 12 millimeters of diameter and 21 millimeters of height. In the 16 millimeters brushless version the EC-max[11] reaches up to $5000\,rpm$ with 3.5 Millinewtons-meters of torque and 5 watts. Very light gear heats are available for high accuracy movements.

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[4]UTJ http://www.spectrolab.com/DataSheets/TNJCell/utj3.pdf

[5]ONAVI 23503-400-0100-A http://www.o-navi.com/Gyrocube3A_4.pdf

[6]KINGMAX http://www.kingmax.com/material/download/3/superstick.pdf

[7]Aerogel http://www.matweb.com/search/DataSheet.aspx?MatGUID=c864d25c235648d6b11711fd324b64d4

[8]FOAM http://www.matweb.com/search/DataSheet.aspx?MatGUID=91d44cae736e4b36bcba94720654eeae

[9]SONCEBOZ http://www.sonceboz.com/medias/produits/fiches-techniques/6415.pdf

[10]Amax D12 200938-08-102-e https://shop.maxonmotor.com/

[11]ECmax D16 283825-08-173-e https://shop.maxonmotor.com/

1.2..   Parameters that affect the launch cost

The purpose of using existing technologies not qualified for space is to reduce the development cost, and to reduce the payload mass, since it means reducing the cost of the launch. If at the same time we were able to increase the success rate of launches, this would result in a decrease of the assurance cost. We study parameters that affect the launch cost of a small launcher when is used for a complex mission like to put a lunar rover explorer in the Moon.

In the following section, we have selected seven *presets* for missions to the Moon which are series of simulations. In annex C we have detailed each *preset*. We focus this study to know the effect of the following parameters: Payload mass, Kind of propellant, Engine technology, Staging, Launch trajectory, Launch site latitude and altitude.

1.2.1..   Series of simulations for the launch cost study

Figure 1.1 shows five trajectories corresponding to five of seven *presets* for a mission to the Moon and a lunar rover[14] of 500 grams as a payload developed in the *Moon2.0* simulator[17]. We have used this mission because it allows to test different aspects of the mission in similar conditions[8].
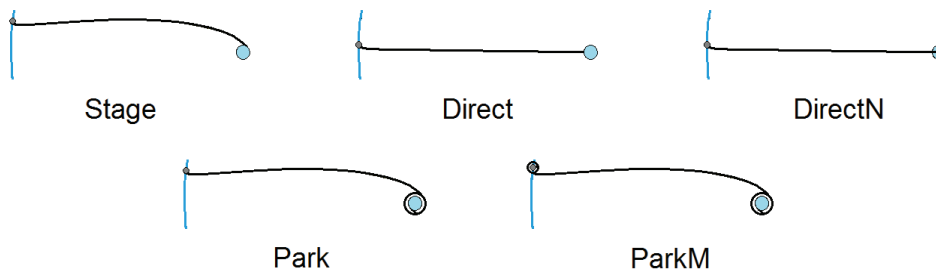
Figure 1.1: Different trajectory simulations for a mission to the Moon (*Presets*)

Presets Stage 1 to 3 We have performed three series of simulations varying the number of stages in the rocket. A single stage requires a large launcher. Two stages configuration is the optimum if solid propellant is used for the first stage and for small payloads in terms of cost. Three stages configuration is efficient if liquid propellant is used for all the stages but in this case, the cost is high, due to the engines cost.

Preset Direct Is a series of simulations without any Earth or Moon parking orbit. The launcher configuration is in two stages. The launch site is at a latitude of 28 degrees North for an Easting launch. This is one of the best solutions for a mission to the Moon.

Preset DirectN Additionally, this simulation uses the North pole as a launch site. There is an extra cost due to the fact that the Earth rotation is not used, and it has allowed us to confirm that the simulation software works properly.

Preset Park Is a series of simulations with LEO parking orbit and at least one rotation around the Earth. There is an extra cost due to the need to circularize the orbit but is not as large as expected.

Preset ParkM Is similar to the *Park* preset, but a Moon parking orbit is used as well. The extra cost is very small compared to the landing cost.

1.2.2.. Analysis of launch parameters

A series of simulations where done using a similar launcher and payload, using new technologies[1] in order to study the effect of typical launch parameters:

- Payload mass

- Kind of propellant

- Engine technology

- Staging

- Launch trajectory

- Launch latitude

- Launch altitude

One of the main causes for the high cost of a launcher is its development cost. We can reduce this cost if the subsystem designs are reused, or if we use some of the satellite subsystems as part of the avionics of the rocket. For example, the IMU of the satellite can perform exactly the same task for the rocket, thus reducing its mass and complexity. In addition, the use of new technologies that are available in the market allows us to reduce the payload mass too. As the total mass of the launcher is a function of the payload mass, a reduction in size means a reduction in the launch cost as well. The failure rate of rocket launchers have a clear impact on the cost of launch assurances; if the launch had a very low price, it would be possible to take higher risks –even to repeat a launch many times–, allowing a cycle of *trial and error*[12] that would conduce to a more reliable system.

Another source of cost is related to staging, and thus to complexity. Results in Table 1.1 for one, two and three stages cases show that the optimum configuration, using a low cost solid propellant, is two stages in a ratio of 90% of the propellant mass for the first stage and about 10% for the second stage if the second stage is based on liquid propellant and reuses the engine of the lunar lander. We are taking into account only the cost of the required propellant and the cost its engine; a larger liquid propellant engine is much more expensive than a larger solid propellant engine. As for the combination of LF2/LH2, it must be said that fluorine is the most electronegative atom in nature, but the combination H2F2 produces hydrofluoric acid, that is very contaminant and should be avoided as a rocket exhaust. Even, holding fluorine –also very corrosive– would not be easy.

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[12]Trial and error methodology: http://en.wikipedia.org/wiki/Trial_and_error

The cost of the type of launch could be significant. The latitude of the launch site is significant in the quantity of propellant if we launch in Easting direction as we can take profit of the rotation of the Earth. Also a reduction of propellant is experimented if a minimum launch orbit[15] is used instead of a straight trajectory. No Earth parking orbit reduces the cost of launch and because these are unmanned missions, no return points are required. Proposed accurate methods for self tracking also avoid the need of this parking orbit, simplifying the trajectory.

Depending on the type of mission some parameters may be more relevant than others but for the proposed mission to the Moon we have the following distribution for each parameter sumarized in the table 1.1. See annex C:Moon20simulator for further details.

Table 1.1: Comparison between launch parameters
From Canary Islands to the Moon except for Polar and Lat28.5 missions.
Payload (Pico Lunar Rover): 500 grams
Lunar lander based on liquid propellant: 9.5 kg
Mission time for all cases: 2.5 days
Launch altitude at 30 km for Balloon and Cannon missions.

Related to launcher construction			
Payload mass	50 kg	5 kg	0.5 kg
Mission cost	US$91M	US$9M	US$1M
Launcher total mass	203,600 kg	20,400kg	2,030 kg
Kind of propellant	LF2LH2/LLi	N2O4/MMH	APCP
Propellant cost	US$129	US$7,294	US$68,574
Launcher total mass	8.18 kg	838 kg	2,030 kg
Engine technology	LF2LH2/LLi	Liquid	Solid
Engine cost	US$100M	US$30M	US$1M
Launcher total mass	8.18 kg	838 kg	1,380 kg
Staging	1Stage	2Stages	3Stages
Launcher cost	US$3M	US$1M	US$2M
Launcher total mass	30,000 kg	1,380 kg	2,038 kg
Related to the launch trajectory			
Launch strategy	Direct trajectory	Parking trajectory	Optimum trajectory
Mission cost	US$56,787	US$48,718	US$46,841
Launch latitude	Polar	Lat28.5	Lat28.2
Trajectory cost	US$59,058	US$47,933	US$46,841
Launch altitude	0 meters (Rocket)	30 km Balloon	30 km Cannon
Propellant cost	US$ 46,841	US$ 26,583	US$ 2,105
Launcher total mass	2,038 kg	804 kg	131 kg

1.2.3.. Discussion of launch parameters

Some improvements can be obtained through the simulation series respect to the preferred launch configuration in the following cases:

Payload mass A cost of US$1.82M for each kilogram of payload.

Kind of propellant 530 times fuel saving using LF2LH2/LLi and 9 times using N2O4/MMH respect to the APCP.

Engine technology Up to 30 times higher cost for the liquid propellant and 100 times for the LF2LH2/LLi respect to the solid propellant.

Staging 22 times more massive launcher for a single stage and 1.5 times more massive launcher for 3 stages respect to the optimum 2 stages launcher.

Low cost launch trajectory Cases up to 1.21 times more costly for direct trajectory respect to the low cost launch trajectory and very similar for a 160 km LEO parking orbit.

Launch latitude Cases up to 1.3 times more costly for a polar **launch** trajectory respect to the Canary islands latitude of 28.2 degrees.

Launch altitude Up to 1.76 times of fuel saving from a 30 km balloon launcher.

1.3.. New technological drivers

Technological drivers are principles or laws which make feasible to accomplish a system need with open access resources. Some examples and *Wikipedia* references are included. These references were revised by the author to ensure its quality, completeness and usefulness inside the scope of this master thesis.

High power density batteries. It is possible to store a high amount of electrical power in a small volume and able to support high forces[13]. An example of this is a Coin battery.

High integration level. It is possible to integrate a large number of transistors and components in a solid state with a very low electrical consumption[14]. An example of this is the growing marked of SMD embedded components. Nevertheless, higher integration means higher sensitivity to radiation effects and shielding improvements are required.

Photovoltaic cell. It is possible to extract the energy from the sun light and convert it in a efficient way into electrical power with a very low weight. Example: UTJ[15] solar cell with up to 28% of efficiency.

Photodiode sensor. Concentrating the light, it is possible to have a high quality image[16] in a very integrated solid state component. An example of this is a high definition HD camera.

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[13]Li-Ion batt. http://en.wikipedia.org/wiki/Lithium-ion_battery
[14]Embedded http://en.wikipedia.org/wiki/Embedded_system
[15]UTJ http://en.wikipedia.org/wiki/Multijunction_solar_cell
[16]Photodiode http://en.wikipedia.org/wiki/Photodiode

Accelerometer. It is possible to sense the motion of an object in a very integrated solid state and very light device[17]. A good example of this is a solid accelerometer inside an IMU, also known as Inertial Measurement Unit.

Gyroscope. It is possible to sense the turn rate of an object in a very integrated solid state and very light device[18]. A good example of this is a solid angular rate sensor inside an Inertial Measurement Unit.

Synthetic Aperture Radar. It is possible to improve the range of a link concentrating and changing its direction without any mobile part. An example of this is a nano-SAR[19] based on the synthetic aperture of the radiation lobe.

1.4.. New methodological approaches and tools

The ten key factors that make this new approach feasible are:

1 The integration of both, the launcher and the satellite in the design cycle of a space mission.

2 The use of new technologies (like COTS[20] components) and its validation for the space qualification; this allows us to choose between the full range of the market as a function of the mission's needs.

3 The design of complex engineering projects based on the community and not in the industry.

4 A community of people motivated by the enjoyment of a common and elevated challenge which is in benefit of the whole community.

5 The use of local resources for a competition and a sense of team.

6 The integration of all the design's phases in the same tool. A simulator that covers from the propellant design, the launcher physical parameters, staging and trajectories, the ground coverage, etc. with hardware in the loop. The last allows us to test real hardware.

7 The use of local testing facilities that allow us to validate and qualify for the space new components.

8 The use of solid propellants, with a specific impulse greater than 200 seconds, give us access to the space with a low cost and low complexity.

9 The use of a propellant characterizer like the Constant Pressure Combustion Chamber (see the Annex D.7 for details about the CPCC) allows us to characterize new solid propellants using small specimens extracted directly from the manufacturer process.

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[17] Accelerometer http://en.wikipedia.org/wiki/Accelerometer
[18] Gyro http://en.wikipedia.org/wiki/Gyroscope
[19] SAR http://en.wikipedia.org/wiki/Synthetic_aperture_radar
[20] COTS Commercial-off-the-shelf

10 The possibility of building our own boards with SMD components reduces the cost of electronic devices in 50%; it gives us some independence from providers and avoids the extra cost of taxes when we import these components.

The boom of COTS (Commercial-off-the-shelf) presents a number of products for a consumer market that are suitable for high demanding applications. Many of these products were military applications. Next, a list of these products is presented that will be used in the femto-satellite and mini-launcher design. Many of them are commercially available in *Sparkfun.com* in the form of COTS, and they provide datasheets. A more specific product or a sum of them can be easily built using our own PCB board designs. This technology changes very fast, and so, during the development of these space programs (less than one year) many updates where done because new products appeared. Figure 1.2 show four good examples of the latest technology: the *GPS-08936*, the IMU *SEN-09431*, the *eZ430* Satellite-on-a-board and the thermo cell *G1-1.0-127-1.27*.
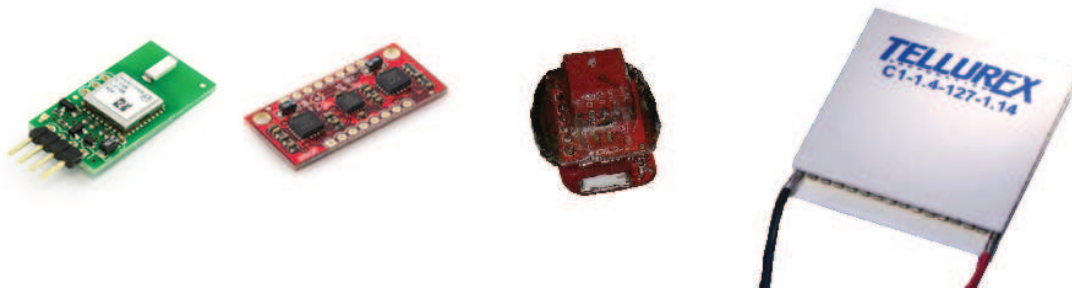
Figure 1.2: Preferred COTS. a) GPS b) IMU c) Satellite-on-a-board d) Thermoelectric cell

GPS Low cost high accuracy GPS. The preferred component for our space program is the GPS-08936. This is a tiny GPS module[21], with a accuracy better than 3 meters that uses the MN5010HS module in conjunction with a GPS chip antenna. Built in LNA[22] makes for reliable locks from satellites. Selectable baud rate and NMEA[23] and SiRF[24] outputs from a jumper on the back of the board.

IMU Low cost high accuracy 6DOF IMU. The preferred component for our space program is the SEN-09431. The 6DOF Razor[25] makes use of ST's LPR530AL (pitch and roll) and LY530ALH (yaw) gyros, as well as the popular ADXL335 triple-axis accelerometer, to give you six degrees of measurement on a single, flat board.

Satellite-on-a-board The preferred component for our space program is the eZ430-RF2500. The eZ430-RF2500T target board[26] is an out-of-the box wireless system that may be used with the USB debugging interface, as a stand-alone system with or without external sensors, or may be incorporated into an existing design. It has 21 configurable pins. It has a highly integrated, ultra-low-power MSP430 MCU with $16MHz$ performance. See http://www.ti.com/msp430 for additional details.

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[21]GPS http://www.sparkfun.com/commerce/product_info.php?products_id=8936
[22]LNA Low Noise Amplifier
[23]NMEA National Marine Electronics Association http://en.wikipedia.org/wiki/NMEA_0183
[24]SiRF http://en.wikipedia.org/wiki/SiRF
[25]IMU http://www.sparkfun.com/commerce/product_info.php?products_id=9431
[26]eZ430 http://focus.ti.com/lit/ug/slau227e/slau227e.pdf

New communications The use of free amateur S-Band for small operations like Wi-Fi or in our case, $2.4\,GHz$ for the eZ430 board, it saves up to 2 years of development time because it is not required to assign a dedicated frequency to this mission. To increase the link range, a high gain arrow antenna is used. Also for tracking, an easy, Internet based technique is used like the real-time APRS[27] based Tactical Tracking System. The preferred component for our space program is the SRB MX146 transmitter[28].

Cameras Low cost High integration cameras. The preferred component for our space program is the SEN-08668 HD camera[29] $1,300$ x $1,040$ pixels 11 x 11 x $6\,mm$.

Alternative power supply Usually, an UTJ solar cells is the best power to weight ratio for power supply but thermoelectric cells can be used as power supply and at a time to control thermal system. The preferred component for our space program is the thermal cell[30] G1-1.0-127-1.27.

1.5.. Synergies for low cost missions

Nowadays, when a company wants to launch a payload to the space, firstly it must design the payload and then choose an available launcher in the market. Following the *Space Payload Paradigm*, we present a different approach which consists in designing the launcher for a given payload or a kind of payloads. We have detected some advantages using this new paradigm (see the introduction of this Chapter *A new market for femto-satellites*) for small satellites and short life span missions. Synergies allow us to reduce the complexity and the weight of the system.

The first synergy that we propose is the use of the femto-satellite itself as a central processor unit for the mini-launcher. That way, no computer is required for the mini-launcher and the launcher is a set of stages with external autonomous control; hence it is an extension of the payload.

The second synergy is the integration of the thermal control system and the electrical power supply based on thermo-electric cells. When the sun overheats the satellite, batteries are charged for the eclipse phase and thermal control is produced. Also, when the engines burn the propellant, a heat source is available that can be used during the launch phase and it can be converted into electrical power.

Other synergy is the use of the structure as a thermal stabilization, this is to say, the satellite and the high gain antenna are embedded into resin. This simplifies the thermal model, reduces the time of development and the possibility of fault as well.

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[27] APRS Automatic Position Reporting System
[28] APRS http://en.wikipedia.org/wiki/Automatic_Packet_Reporting_System
[29] HD camera http://www.sparkfun.com/commerce/product_info.php?products_id=8668
[30] G1-1.0-127-1.27 datasheet http://www.tellurex.com/pdf/G1-1.0-127-1.27.pdf

1.6.. Applications for femto-satellites and mini-launchers

1.6.1.. Applications for femto-satellites by them selves

Femto-satellites by them selves give opportunities to make space research and commercial applications. Using these femto-satellites in the current market is only useful with swarm of thousands of these launched by a piggy-back opportunity.

Also on ground, the use of these femto-satellites by it-self, each one attached to a different person or vehicle, make a powerful tool with tactic capability. This configuration may save lives and mitigate the effects of disasters on the environment. In emergency situations, it is important to communicate from the field what has happened and where as quickly and accurately as possible so that appropriate action can be planned and executed. In these situations a femto-satellite network can improve an adaptive coverage to the emergency.

An example of this is a fast earthquake mapping by the First Responders (FR) when they have the first contact with the operation theater. This information is transmitted to the base camp and to the Head Quarters (HQ) in real time in order to allow to the Decision Makers (DM) to take a correct action plan based on augmented virtual reality AVR[31]. The sum of all points of view of each femto-satellites make a global map of the operation theater.

1.6.2.. Combination between femto-satellites and mini-launchers

When femto-satellites are combined with mini-launchers, the result is a new market of low-cost launch and operations, fast launch decision in few hours, the capability of adapting the satellite to a changing mission and short time of disposing around one week.

Earth observation applications are not the best market for this combination even using a swarm of femto-satellites could be a new way to have a multi-point sensing system, like to model the gravity field not only based on one sensor but a net of hundreds of sensors working together. Also this new approach can to make atmosphere studies from a multi-point of view never done since now.

Beyond of this, since each femto-satellite has a down-link and relay-link at the same time, the adaptation of the global coverage to the available ground stations raise new philosophies closer to neural networks where the capacity of the mission do not depends on a single processor or a single memory. This neural networks based on multi-agents brings the possibility to have Tera-FLOPs[32] of CPU capacity with parallel processing. Also it brings the capacity of to have Tera-bytes of memory like a FPGA[33] has, where the memory is located in each cell and not in a main storage memory.

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[31]AVR Augmented Virtual Reality
[32]FLOP FLoating point Operations Per Second
[33]FPGA Field Programmable Gate Array

CHAPTER 2. THE FEMTO-SATELLITE

The scope of this chapter is to define how a femto-satellite can be built based on new technologies and how it can be validated and qualified for the space.

2.1.. Femto-satellite requirements

The WikiSat[1] organization is the one in charge of the development and the implementation of the femto-satellite and the mini-launcher[4]. This organization has implemented in its designs the following directives and policies respect to the femto-satellite:

Simplicity directive We follow the KISS rule. KISS stands for Keep It Simple and Safe.

Absolute minimum directive Closer tolerances and a design only for its mission.

No redundancy policy Single fault tolerant system.

Preferred configuration Complete capabilities in the default configuration.

2.1.1.. System Requirements

S0: The WikiSat femto-satellite, as a system shall accomplish the need of winning the N-Prize[2], to track it for a minimum of nine orbits in a Low Earth Orbit higher than 100 kilometers with a mass between 9.99 and 19.99 grams before September 19th, 2011.

2.1.2.. High Level Requirements

HL00: The Power Supply subsystem shall provide electrical power for the computing of the orbit and the tracking.

HL01: The Communication subsystem shall transmit and receive information.

HL02: The Structure subsystem shall protect the femto-satellite components and be used as a thermal path for thermal loads.

HL03: The Attitude determination subsystem shall determine the attitude by inertial means and be helped by optic sensors.

HL04: The Position determination subsystem shall determine the position in the orbit by inertial means and be helped by optic sensors.

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[1]WikiSat http://code.google.com/p/moon-20/
[2]http://en.wikipedia.org/wiki/N-Prize

HL05: The Attitude control subsystem shall point the high gain antenna to the Earth in a passive way using the Earth's magnetic field.

HL06: The Tracking subsystem shall transmit its computed position to a ground station only when passing over the ground station's sky.

HL07: The Video recording subsystem shall record pictures and video if required.

2.1.3.. Low Level Requirements

LL000: The battery shall provide enough power for the whole mission at any time and in peak of power conditions for a limited period of time.

LL001: The electrical power subsystem shall be used in short times having an idle mode.

LL010: The ground communication link shall be disconnected when the ground electrical power source is not available.

LL011: The ground monitoring function shall be disconnected when the ground electrical power source is not available.

LL012: The downlink subsystem shall transmit the monitoring information from the femto-satellite to the ground station before the launch and using the low gain antenna.

LL013: The uplink subsystem shall receive the configuration information from the ground station to the femto-satellite before the launch and using the low gain antenna.

LL020: The structure shall have a high thermal inertia in order to be used as the thermal control.

LL021: The structure subsystem shall support physical loads up to $500\,G$.

LL022: The structure subsystem shall support a temperature range from -150 to $250\,°C$.

LL023: The structure subsystem shall have a surface with cooling properties in order to have a good heat flow in such a way that the resulting temperature remains inside the range between -40 to $60\,°C$.

LL030: The attitude determination subsystem shall be calculated from two different sources (Optical devices and gyros)

LL040: The position determination subsystem shall be calculated from two different sources (Optical devices and accelerometers)

LL041: The position determination subsystem shall guarantee an error less than one degree in latitude and one degree in longitude and the sum of both has to be an area less than $10,000$ square kilometers.

LL050: The attitude control subsystem shall point the high gain antenna towards the ground with an angular accuracy of 5 degrees.

LL051: The attitude control subsystem shall absorb any rotation energy produced by the radiation pressure wind in less than few minutes.

LL060: The tracking subsystem shall transmit the femto-satellite computed position at least once to every ground station in the list of available ground stations.

LL070: The Video recording subsystem shall take pictures with a resolution of at least $1,280$ horizontal by $1,024$ vertical.

LL071: The Video recording subsystem shall take videos with a frame rate of at least 15 frames per second.

LL072: The Video recording subsystem shall have the possibility of compressing pictures and videos using a JPEG compression.

2.1.4.. Additional requirements

Operational Requirements

AD000: The femto-satellite shall be able to receive the launch position in order to align the Inertial Measurement Unit before launch.

Safety Requirements

AD001: The femto-satellite battery shall be only used when the femto-satellite is deployed.

Performance Requirements

AD002: The tracking subsystem shall be able to illuminate an area of 200 kilometers in diameter.

Physical and Installation Requirements

AD003: The femto-satellite size shall be lower than 0.2 meters in any direction.

Maintainability Requirements

AD004: The femto-satellite shall be able to keep ready to launch at least for two years without any maintenance action.

Interference Requirements

AD005: The femto-satellite shall be electromagnetically compatible with the mini-launcher.

The WikiSat organization has three space programs.

WikiSat, a 20 **grams femto satellite** http://code.google.com/p/moon-20/wiki/WikiSat

Wiki-Launcher, a mini launcher less than 100 **kilograms** http://code.google.com/p/moon-20/wiki/WikiLauncher

PicoRover to the Moon, a minimal mission to win the GLXP http://code.google.com/p/moon-20/wiki/PicoRover

Also, the WikiSat organization has prepared 5 manuals for the WikiSat, 4 manuals for the Wiki-Launcher and 5 manuals for the PicoRover mission:

WikiSat ConOps Document has information related to the utility; it contains the user manual. This document is used by clients. http://code.google.com/p/moon-20/wiki/WikiSat_ConOps

WikiSat System Requirements Document has the list of requirements that the system shall meet. This document is used by engineers to develop the detailed subsystem document. http://code.google.com/p/moon-20/wiki/WikiSat_System_Requirements

WikiSat System Design Document has the definition of components for each subsystem. This document is used for engineers to build the system. http://code.google.com/p/moon-20/wiki/WikiSat_System_Design

WikiSat Program Management Plan has the planning to build and operate the system. This document is used by engineers to design the mission and for the operator. http://code.google.com/p/moon-20/wiki/WikiSat_Program_Management_Plan

WikiSat Engineering Management Plan has information about the role of each engineer of the organization. This document is used by the manager organization. http://code.google.com/p/moon-20/wiki/WikiSat_Engineering_Management_Plan

Wiki-Launcher manuals:

WikiLauncher ConOps Document has information related to the utility; it contains the user manual. This document is used by clients. http://code.google.com/p/moon-20/wiki/WikiLauncher_ConOps

WikiLauncher System Requirements Document has the list of requirements that the system shall meet. This document is used by engineers to develop the detailed subsystem document. http://code.google.com/p/moon-20/wiki/WikiLauncher_System_Requirements

WikiLauncher System Design Document has the definition of components for each subsystem. This document is used for engineers to build the system. http://code.google.com/p/moon-20/wiki/WikiLauncher_System_Design

WikiLauncher Program Management Plan has the planning to build and operate the system. This document is used by engineers to design the mission and for the operator. http://code.google.com/p/moon-20/wiki/WikiSat_Engineering_Management_Plan

PicoRover Manuals:

PicoRover ConOps Document http://wiki.teamfrednet.org/index.php/Picorover_ConOps

PicoRover System Requirements Document http://wiki.teamfrednet.org/index.php/Picorover_Requirements

PicoRover System Design Document http://wiki.teamfrednet.org/index.php/Picorover_Design

PicoRover Program Management Plan http://wiki.teamfrednet.org/index.php/Picorover_Program_Manageme

PicoRover System Engineering Management Plan http://wiki.teamfrednet.org/index.php/Picorover_Enginee

2.2.. Femto-satellite system design

In the next sections, a series of budgets is presented for the WikiSat space program. Also a boom view is presented with a detailed list of subsystems, the preferred components and their specifications.
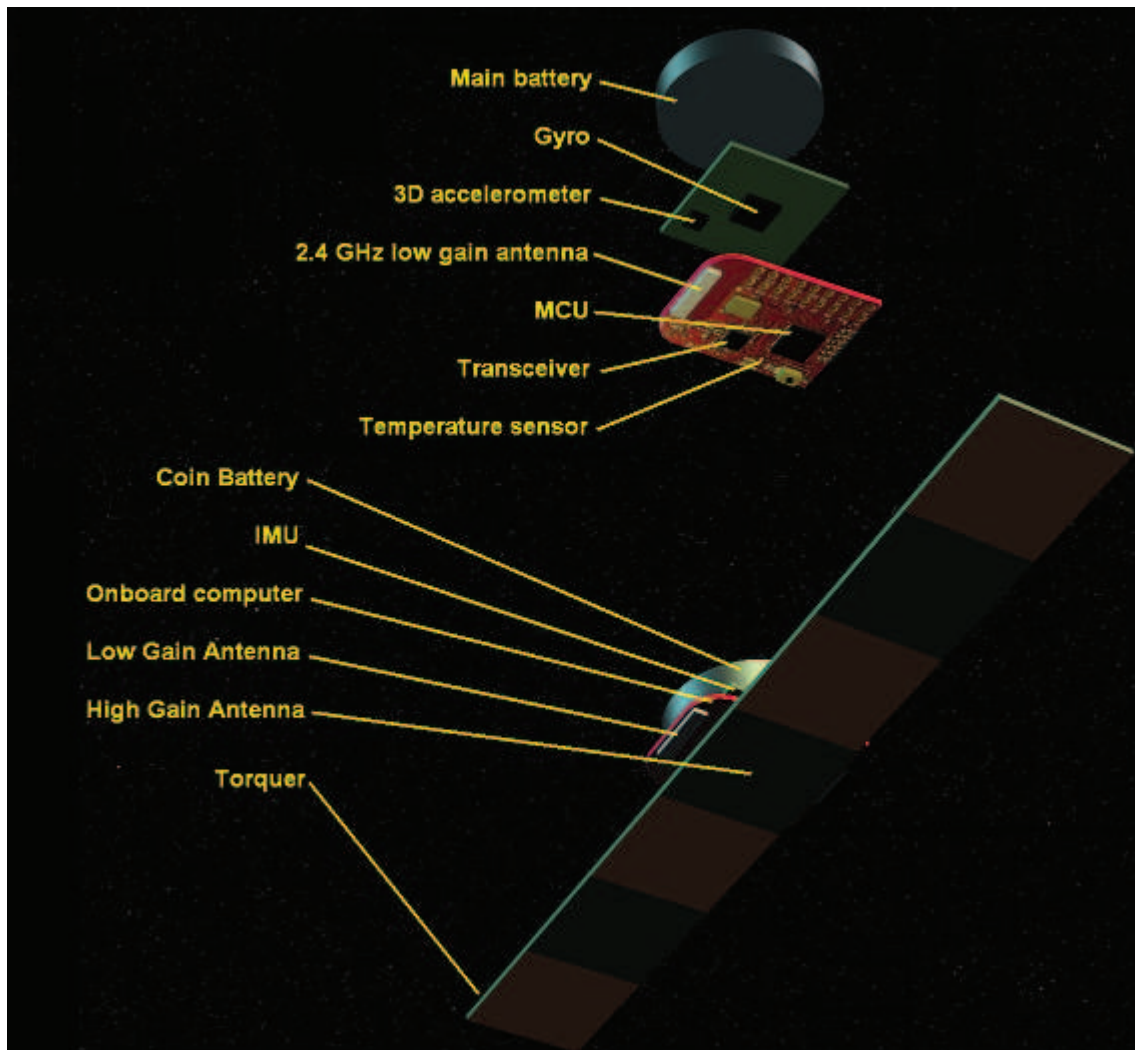
Figure 2.1: a) Link budget b) Mass budget c) Boom view

2.2.1.. Link budget

The link budget is based on the relation between the power available for communications, the orbital altitude, the gains of the emitting and receiving antennae, and the data rate, among other factors. A good link implies a high ration of the energy per bit (E_b) to noise density (N_0), which can be calculated by

$$\frac{E_b}{N_0} = P + L_l + G_t + L_s + L_a + G_r + 228.6 - 10 \log T_s - 10 \log R$$

where P is the emitted power (in dBW), L stand for losses (L_l are line losses, L_s spatial losses, and L_a atmospheric losses), G_t and G_r are, respectively, the gains of the transmitting and receiving antennae, T_s the system's temperature, and R is the data rate. Table 2.1 reviews the results for the Wikisat, that show that the link is of good quality (the signal to noise ratio is of $2.2\,dB$).

Table 2.1: Worst case link budget for the Femto-satellite

Magnitude	Value	Unit	Equivalent	Unit
Distance	200	km		
Frequency	2.40	GHz		
Band Width	53.0	kHz		
EIRP	-19.0	dBW	11.0	dBm
Receiver gain	20.0	dB		
Temperature	323.0	K	50	$°C$
Transmit loss	3.0	dB		
Offset loss	3.0	dB		
Receive loss	3.0	dB		
λ	0.13	m		
Free space losses	146.07	dB		
Power Received	-154.07	dBW	-124.07	dBm
Power noise	-156.26	dBW	-126.26	dBm
Signal to noise ratio SNR	2.2	dB		

2.2.2.. Mass budget

The femto-satellite has a default configuration which is composed by a default list of components. Table 2.2 presents such a list with the femto-satellite default configuration.

Table 2.2: Femto-satellite default component list and mass budget

Subsystem	Mass	Max. power	Idle	Temp. range	Size
	grams	*mW*	*mW*	*°C*	*mm*
Power supply subsystem	6.6	$3V, 610\,mAh$		-30 to 60	$D24.5$ x 5
Communication subsystem	0.1	100	0.1	-40 to 85	see eZ430
Structure subsystem	2.0	$-$	$-$	-116 to 204	30 x 25 x 7
Attitude determination subsystem	1.4	5	0.1	-30 to 70	$D5$ x 2
Position determination subsystem	1.2	65	0.1	-30 to 60	30 x 16 x 3
Attitude control subsystem	0.4	40	30	-150 to 550	1 x 0.25 x 0.25
Tracking subsystem	7.0	500	250	-40 to 85	140 x 25 x 1
Video recording subsystem	1.0	150	150	-20 to 60	11 x 11 x 6
TOTAL	19.7	860	330	-20 **to** 60	140 **x** 30 **x** 7

2.2.3.. **Thermal budget**

The heat flow received to the femto-satellite is $\dot{Q}_{in} = \alpha \cdot I \cdot F \cdot A$ and the heat emitted by the femto-satellite is $\dot{Q}_{out} = \sigma \cdot T^4 \cdot \sum (\varepsilon \cdot A)$; where α is intensity, I is the absorbance[3], F is the geometric factor, A is the effective area, σ is the Boltzmann constant of $5.67 \cdot 10^{-8}\,W/(m^2 \cdot K^4)$, T is the effective temperature and ε is the emissivity[4].

The maximum heat flow able to irradiate the femto-satellite (Maximum cooling heat flow) at the maximum operative temperature of 60 $°C$ is $\dot{Q}_{max} = \sigma \cdot T^4 \cdot \sum (\varepsilon \cdot A)$

$$\dot{Q}_{max} = 5.67 \cdot 10^{-8} \cdot (273 + 60)^4 \cdot \sum (0.92 \cdot 0.0042 + 0.15 \cdot 0.0042) = 3.133\,W.$$

The total heat flow when the femto-satellite is radiated by the sun is $\dot{Q}_{total} = \dot{Q}_{sun} + \dot{Q}_{albedo} + \dot{Q}_{IR} + \dot{Q}_{disipated} = 0.895 + 0.878 + 0.707 + 0.500 = 2.980\,W.$

$$\dot{Q}_{sun} = \alpha \cdot I \cdot F \cdot A = 0.15 \cdot 1,420 \cdot 1.0 \cdot 0.0042 = 0.895\,W$$

$$\dot{Q}_{albedo} = \alpha \cdot I \cdot F \cdot A = 0.92 \cdot 454.4 \cdot 0.5 \cdot 0.0042 = 0.878\,W$$

$$\dot{Q}_{IR} = \alpha \cdot I \cdot F \cdot A = 0.92 \cdot 244.0 \cdot 0.75 \cdot 0.0042 = 0.707\,W$$

$$\dot{Q}_{disipated} = 0.5\,W$$

When the femto-satellite is exposed to the sun, the albedo and the infra red, the resultant temperature for the worst case is

$$T = \sqrt[4]{\frac{\dot{Q}_{total}}{\sigma \cdot \sum (\varepsilon \cdot A)}} = \sqrt[4]{\frac{2.980}{5.67 \cdot 10^{-8} \cdot (0.92 \cdot 0.0042 + 0.15 \cdot 0.0042)}} = 329\,K = 56\,°C,$$

which is below the maximum operating temperature. In these conditions, when the satellite is in idle, the resultant temperature is $324\,K = 51\,°C$.

When the femto-satellite is in the eclipse, only it is exposed to the infra red, the resultant

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[3]Absorbance http://en.wikipedia.org/wiki/Absorbance
[4]Emissivity http://en.wikipedia.org/wiki/Emissivity

temperature for the worst case is

$$T = \sqrt[4]{\frac{\dot{Q}_{total}}{\sigma \cdot \sum(\varepsilon \cdot A)}} = \sqrt[4]{\frac{1.207}{5.67 \cdot 10^{-8} \cdot (0.92 \cdot 0.0042 + 0.15 \cdot 0.0042)}} = 262\,K = -11\,^{\circ}C,$$

which is inside the operating temperature range. In these conditions, when the satellite is in idle, the resultant temperature is $253\,K = -20\,^{\circ}C$.

The conclusion for this budget is that the femto-satellite does not need a thermal control subsystem because it is self regulated; it has a good balance between the incoming heat flow and the outlet heat flow. If temperatures in the worst case when the femto-satellite is exposed to the sun were too high then new materials will be used as a cooler subsystem. If temperatures in the worst case when the femto-satellite is in the eclipse were too low then heaters will be required.

Table 2.3: Thermal budget overview

	Maximum cooling heat flow					
	$\dot{Q}_{sun}$	$\dot{Q}_{albedo}$	$\dot{Q}_{IR}$	$\dot{Q}_{disipated}$	$\dot{Q}_{TOTAL}$	Temp. $K\,(^\circ C)$
Cooling	-	-	-	-	3.133	333 (60 $^\circ C$)
	Sun case					
	$\dot{Q}_{sun}$	$\dot{Q}_{albedo}$	$\dot{Q}_{IR}$	$\dot{Q}_{disipated}$	$\dot{Q}_{TOTAL}$	Temp. $K\,(^\circ C)$
Idle case	0.895	0.878	0.707	0.330	2.810	324 (51 $^\circ C$)
TX case	0.895	0.878	0.707	0.500	2.980	329 (56 $^\circ C$)
	Eclipse case					
	$\dot{Q}_{sun}$	$\dot{Q}_{albedo}$	$\dot{Q}_{IR}$	$\dot{Q}_{disipated}$	$\dot{Q}_{TOTAL}$	Temp. $K\,(^\circ C)$
Idle case	0.000	0.000	0.707	0.330	1.073	253 (-20 $^\circ C$)
TX case	0.000	0.000	0.707	0.500	1.207	262 (-11 $^\circ C$)

2.2.4.. Radiation budget

The exposure time, based on the *Moon2.0* simulations is 3 days but we are using one week.

The orbit is $200\,km$.

The highest femto-satellite area is $0.0042\,m^2$.

The femto-satellite drag coefficient is 0.5 at low speed and 2.5 at high speed.

The shield material is fiberglass (Polyester and glass fibers).

The emissivity of the fiberglass is about 0.92 and for the copper (Back plate high gain antenna) is about 0.15

The dissipated heat by the femto-satellite is between 50 to $500\,mW$.

The maximum cooling heat allowable by the shield and the high gain antenna is $3.133\,W$ at $60^\circ C$ in the mainframe.

2.2.5.. Coverage budget

Due to the low altitude of the femto-satellite, there are a few shadows without coverage. We are assuming that all the commercial space ground stations are available. There are five areas of shadow: Sahara desert, Indian ocean before Australia, Pacific ocean after Australia, a large area in the Pacific ocean and Atlantic ocean before Canary Islands. Similar situations occurs for the rest of the orbits.
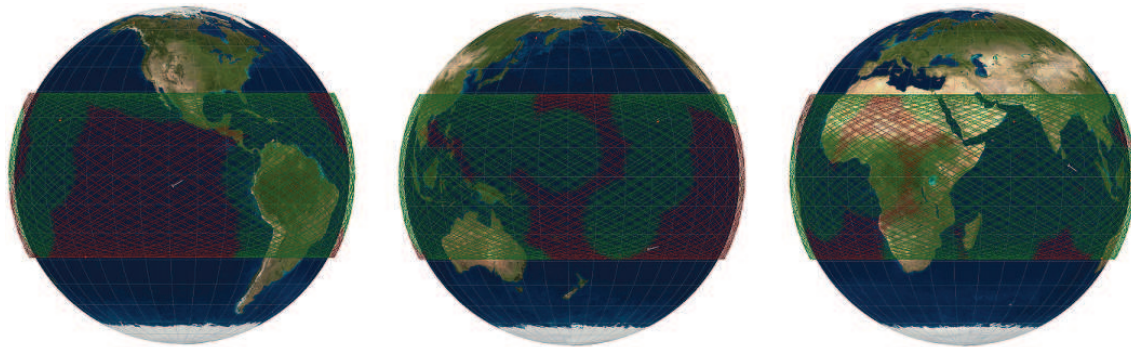
Figure 2.2: Detail of the coverage shadows due to the low altitude

To overcome this problem and to save battery, a selective broadcasting is performed by the femto-satellite when crosses over any scheduled ground station. A list of ground stations has to be uploaded before the launch.

2.3.. Femto-satellite detailed subsystem design

The WikiSat femto-satellite has as a default configuration the following subsystems:

- Power Supply subsystem

- Communication subsystem

- Structure subsystem

- Attitude determination subsystem

- Position determination subsystem

- Attitude control subsystem

- Tracking subsystem

2.3.1.. Power Supply subsystem

The Power Supply subsystem shall provide electrical power for the computing of the orbit and the tracking. The preferred component is a Coin battery[5] type CR2450. This battery is only used when the femto-satellite is deployed. The load of this battery is for the IMU and the MCU; it is only $50\,mA$ but it last all the mission. Only when the femto-satellite is over the ground station, it is used to transmit the information for the tracking system; this situation happens once each turn but it is $500\,mA$ for few milliseconds. Tracking is disabled when the total load is high in order to ensure the continuity of the IMU function and reduce as much as possible the accumulated error. This situation happens at the end of the femto-satellite life, during the eclipse condition and when the battery voltage is low. On ground or during the launch sequence, an external power supply will be available to feed the femto-satellite provided by the launch-pad or the mini-launcher.

Specifications:

- Weight: $6.6\,grams$

- Power: $3\,V$ and $610\,mAh$

- Size: $D24.5 \text{ x } 5\,mm$

- Temperature range: -30 to $60^{\circ}C$

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[5]CR2450 http://www.houseofbatteries.com/pdf/CSO-CR2450

2.3.2.. Communication subsystem

The Communication subsystem shall transmit and receive information using the low gain antenna (Fig. 2.3) in the $2.4\,GHz$ amateur band. Configuration information is sent from the ground station to the femto-satellite. Monitoring information is sent from the femto-satellite to the ground station. The preferred component is the wireless radio inside the eZ430 satellite-in-a-board[6] type CC2500.

Specifications:

- Idle power: $0.4\,mA$

- TX power: $100\,mA$

- Frequency: $2.4\,GHz$

- Specs: see eZ430 Satellite-in-a-board
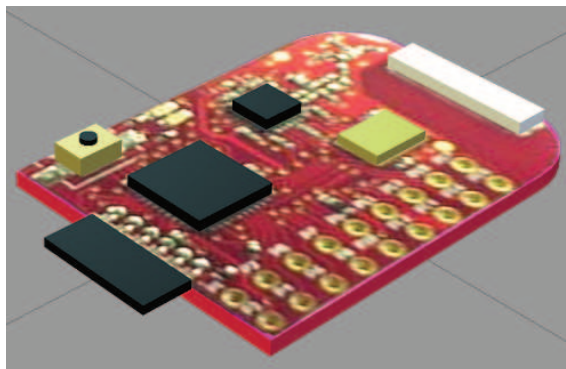
Figure 2.3: Detail of the low gain antenna in the eZ430 satellite-in-a-board

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2.3.3.. Structure subsystem

The Structure subsystem shall protect the femto-satellite components and be used as a thermal path for thermal loads. The preferred material is Fiberglass[7]. This material has a space qualification based on the NASA SP-3094 recommendations[8]. The fiberglass is composed by Polyester as the matrix[9], Glass fibers as the reinforcement[10] and Sthirene + Anhydric ftalic as the accelerator[11].

Specifications:

- Power: Not apply.

- Density: $1.7\,kg/dm^3$

- Modulus of Elasticity[12]: $0.0328\,GPa$

- Thermal conductivity[13]: $1.5\,W/(m{\cdot}K)$ at $300\,K$

- Emissivity[14]: 0.92

- Area: $0.0042\,m^2$

- Temperature range: -116 to $204^{\circ}C$

The high gain antenna (Fig. 2.4) is made of a thin copper layer[15] which also has a space qualification based on the NASA SP-3094 recommendations. Specifications:

- Density: $8.96\,kg/dm^3$

- Modulus of Elasticity: $110\,GPa$

- Thermal conductivity: $385\,W/(m{\cdot}K)$ at $300\,K$

- Emissivity: 0.15

- Area: $0.0042\,m^2$

- Temperature range: -150 to $1,083^{\circ}C$

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[7]Fiberglass http://www.matweb.com/search/DataSheet.aspx?MatGUID=d234c9575f63487cbd69a0df8bfcce34
[8]SP-3094 http://hdl.handle.net/2060/19750016776
[9]Polyester http://en.wikipedia.org/wiki/Polyester
[10]Fiberglass http://en.wikipedia.org/wiki/Fiberglass
[11]Accelerator http://en.wikipedia.org/wiki/Accelerator
[12]Modulus of Elasticity http://en.wikipedia.org/wiki/Elastic_modulus
[13]Thermal conductivity http://en.wikipedia.org/wiki/Thermal_conductivity
[14]Emissivity http://en.wikipedia.org/wiki/Emissivity
[15]Copper http://www.matweb.com/search/datasheet.aspx?MatGUID=9aebe83845c04c1db5126fada6f76f7e
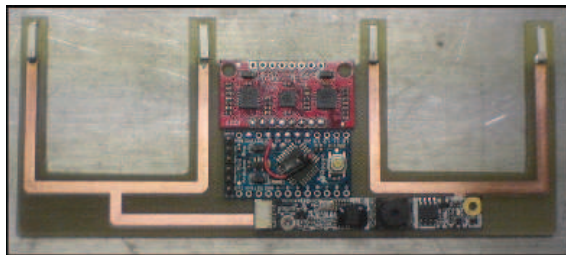
Figure 2.4: Detail of the synthetic aperture high gain antenna of the WikiSat prototype 2

2.3.4.. Attitude determination subsystem

The Attitude determination subsystem shall determine the attitude by inertial means and be helped by optic sensors. The preferred component is the light sensor[16] type SEN-09088. Four optic sensors determine the attitude and also it detects the eclipse.

Specifications:

- Weight: $1.4\,grams$

- Power: $5\,mA$

- Size: $D5$ x $2\,mm$ each

- Temperature range: -30 to $70^{\circ}C$

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[16]SEN-09088 http://www.sparkfun.com/datasheets/Sensors/Imaging/SEN-09088-datasheet.pdf

2.3.5.. Position determination subsystem

The Position determination subsystem shall determine the position inside the orbit by inertial means (Fig. 2.5) and be helped by optic sensors. The preferred components are a 3D accelerometer[17] type ADXL335, the 2D gyro[18] type LPR530AL and the vertical gyro[19] type LY530AL.

Specifications:

- Weight: $1.2\,grams$

- Power: $50\,mA$

- Ratio turn: $300\,degrees/second$

- Max. acceleration: -3 to $3\,G$

- Size: 7 x $7\,mm$ each

- Temperature range: -30 to $60°C$
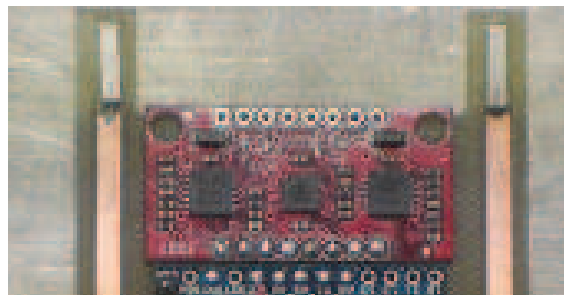
Figure 2.5: Detail of the 6DOF inertial measurement unit of the WikiSat prototype 2

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[17]ADXL335 http://www.sparkfun.com/datasheets/Components/SMD/adxl335.pdf

[18]LPR530AL http://www.sparkfun.com/datasheets/Sensors/IMU/lpr530al.pdf

[19]LY530AL http://www.sparkfun.com/datasheets/Sensors/IMU/LY530ALH.pdf

2.3.6.. Attitude control subsystem

The Attitude control subsystem shall point the high gain antenna to the Earth in a passive way using the Earth's field. The preferred component is the magnetorquer[20] type SM-0103. When the femto-satellite is deployed, it must be released without any spin. four magnets maintain the plane pointing to the ground following the Earth's magnetic field. A large coil is installed in order to absorb any spin energy due to the solar wind.

Specifications:

- Weight: $0.4\,grams$ each

- Power: $40\,mA$

- Maximum energy product: $39.8\,kJ/m^3$

- Size: 1 x 0.25 x $0.25\,mm$

- Temperature range: -150 to $550°C$

---

[20]SM-0103 http://www.eamagnetics.com/library/EAM-Standard-Alnico-Magnets.pdf

2.3.7.. Tracking subsystem

The Tracking subsystem shall transmit its computed position to a ground station only when it is passing over the ground station. The preferred component (Fig. 2.6) is a SMD Low Noise Amplifier[21] type SMA661AS. The frequency for the high gain antenna is different from the low gain antenna in order to avoid coupling. The tracking message contains the satellite ID, the position, the time and the battery voltage. International rules about amateur frequencies set that it is not allowed to encrypt the signal: because of this, the protocol used will remain hidden from the general public until the launch day.

Specifications:

- Idle power: $0.008\,mA$

- TX power: $500\,mA$

- Frequency: up to $70\,GHz$ (Optimum at $1.575\,GHz$)

- LNA: $18\,dB$

- Temperature range: -40 to $85°C$

Figure 2.6: Detail of the low noise amplifier of the WikiSat prototype 2

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[21] SMA661AS http://www.sparkfun.com/datasheets/Components/SMD/sma661as.pdf

2.3.8.. Video recording subsystem

The Video recording subsystem shall record pictures and video if required by the mission. The preferred component (Fig. 2.7) is the SEN-08668 $1.3\,Mpx$ HD camera[22], a solid state component. Pictures have a resolution of at least $1,280$ x $1,024$, videos have at least a frame rate of 15 frames per seconds and it has the possibility of JPEG compression. The optical format is $1/3.3$ inches.

Specifications:

- Idle power: $0.1\,mA$

- Max. power: $150\,mA$

- Resolution: $1,280$ x $1,024$

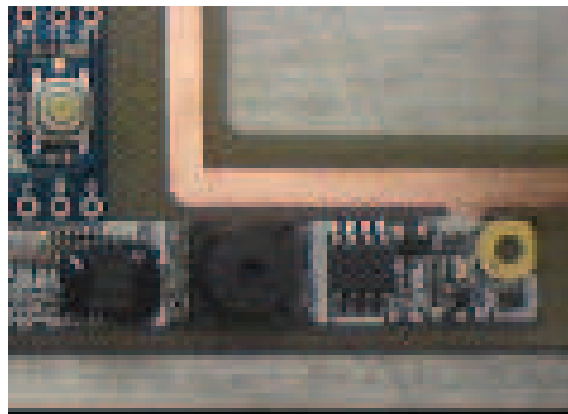- Rate: $15\,fps$

- Temperature range: -20 to $60°C$

Figure 2.7: Detail of the high definition camera of the WikiSat prototype 2

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[22]HD camera http://www.sparkfun.com/commerce/product_info.php?products_id=8668

2.4.. Femto-satellite space qualification

One possible way to validate new technologies is the use of piggy-back in any of the opportunities[9] that **several** large launcher offers[3]. The cost of one piggy-back was around $6,000$ to $8,000\,dollar/kg$ [9] in 1998. The cost of such opportunities is greater than the whole space program for the WikiSat because the kind of payloads are different as well as is its target market. So, we found that piggy-back do not fit our needs. We use high altitude balloons instead. Four types of tests are required to validate and qualify for the space each single component and the whole femto-satellite: Vacuum test, Near space test, Vibration test and Radiation test.

2.4.1.. Vacuum test

Vacuum tests were done for COTS components using a chamber at 10^{-5} atmospheres. We have tested embedded components inside this chamber and the satellite board as well. The time to reach a high vacuum condition is a few hours when pressure reaches to 10^{-3} atmospheres. Due to the capability of the satellite board to communicate by wireless radio, it is very easy to monitor any COTS component to validate and qualify for the space.

We have used the *UPC Applied Physics Department* high vacuum chamber facility to do these tests. The chamber size is a cylinder of 2 meters length and $160\,mm$ in diameter. See the video *http://www.youtube.com/watch?v=HvZUwmnT2tI* in YouTube.

2.4.2.. Near space test

Due to the use of domestic COTS, a validation campaign is mandatory. We are using near space balloon expeditions to have an initial approach of the components to be validated and certified for the space as shown in Figure 2.8. The frame was a weather balloon[23] type: HIM-800. The activity is regulated under the Federal Airworthiness Rule[24] FAR-101.

A prediction of the trajectory is demanded by the authorities in order to publish a NOTAM[25]. The web used is hosted by the *University of Wyoming* and it is called *balloonTraj*[26] having four hours of resolution, it take into account the local weather prediction and produces a KML[27] file that can be visualized in 3D format inside the application Google Earth[28].

We designed a repetitive procedure in agreement with the local airworthiness authority based on night launches, a free airways area and a mobile phone recovery system. We experimented many problems using this passive tracking method. Further campaigns will use a real time APRS[29] tracking system.

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[23]HIM-800 http://www.meteorologyshop.eu/Radiosonding_balloons/ENG_276_EUR_38_632__.html

[24]FAR-101 http://www.sscl.iastate.edu/far101

[25]NOTAM Notice to AirMan

[26]balloonTraj http://weather.uwyo.edu/polar/balloon_traj.html

[27]KML file format http://en.wikipedia.org/wiki/KML

[28]Google Earth http://earth.google.com/

[29]APRS Automatic Position Reporting System

Figure 2.8: Near space balloon expeditions

2.4.3.. Accelerations test

An accelerations device, like a centrifugal machine, is used in order to generate a number of g forces with the aim of seeing the effect in the femto-satellite. This device is used to calibrate the IMU[30] installed inside the femto-satellite.

Also a vibration device should be employed with a range in frequency between $10\,Hz$ and $500\,Hz$ to identify possible resonant frequencies of the satellite that could endanger its structural integrity.

2.4.4.. Electromagnetic and Radiation tests

An electrostatic discharge test (ESD) is required to ensure the capability of flight through a storm or in the static charges in the space produced by radiation.

An electromagnetic compatibility (EMC) test is required every time a new emitter component is installed and also it is required to check every component that show susceptibility to a magnetic or electric field[13]. Coupling mechanisms occur when new metallic components are installed inside the femto-satellite or near them. Due to short distances in the femto-satellite, near field has to be taken into account. The distance between the near field and the far field is a normalized value of *lambda divided by two times pi* where lambda is the wave length. Hence, the near field is larger for lower frequencies. The dominant energy for a near field is magnetic and it is electric for far field.

Figure 2.9 shows the validation test inside a vacuum chamber. An Infra Red source generate a controlled amount of heat. The thermocouple array can evaluate the incoming heat flow and the heat-shield reduce the effect of the radiation. A number of insulator layers can be set in order to protect the satellite board. Insulator particles can be used as a binder inside the resin that is used as a structure.

---

[30]IMU Inertial Measurement Unit
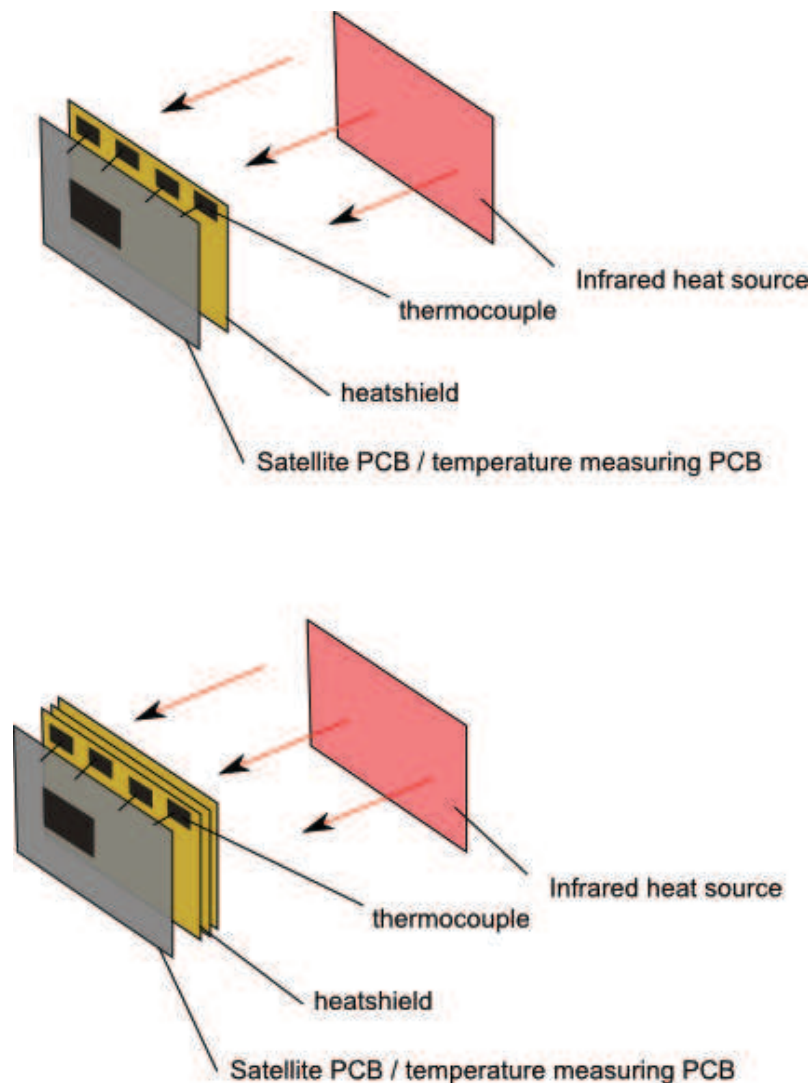
Figure 2.9: Infra Red radiation test. By Victor Kravchenko

We used a web based, free simulator called Spenvis[31] in order to check the effect of the orbit environment over the femto-satellite. Spenvis is the Space Environment Information System from ESA[32] see figure C.5 in the annexes.

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[31] SPENVIS http://www.spenvis.oma.be/
[32] ESA European Space Agency http://www.esa.int/

2.5.. Space programs for the femto-satellite

This section presents a few examples of possible civil applications[2] for the proposed femto-satellite. The main difference between applications is the payload or sensors but a common set of subsystems is given by default. The cost of to develop new missions is small in time and economical resources. The mission purpose gives a reason to the femto-satellite and determines the payload but not the mainframe because these are similar missions but different applications. This kind of missions are suitable for very dynamic and adaptive situations that change fast. An example of civil application with a very short adaptive time is an Earthquake.

2.5.1.. WikiSat space program

The *WikiSat space programs* is a mission based in the *N-Prize* to put a femto-satellite in Low Earth Orbit higher than 100 kilometers Low Earth Orbit (see Figure 3.4) with the following characteristics:

- A very short time of development (less than one year)

- A very low launch cost (less than 1,500 dollars)

- A very easy and automatic operation. No human action required in flight

- A very short time and operation cost (less than one week)
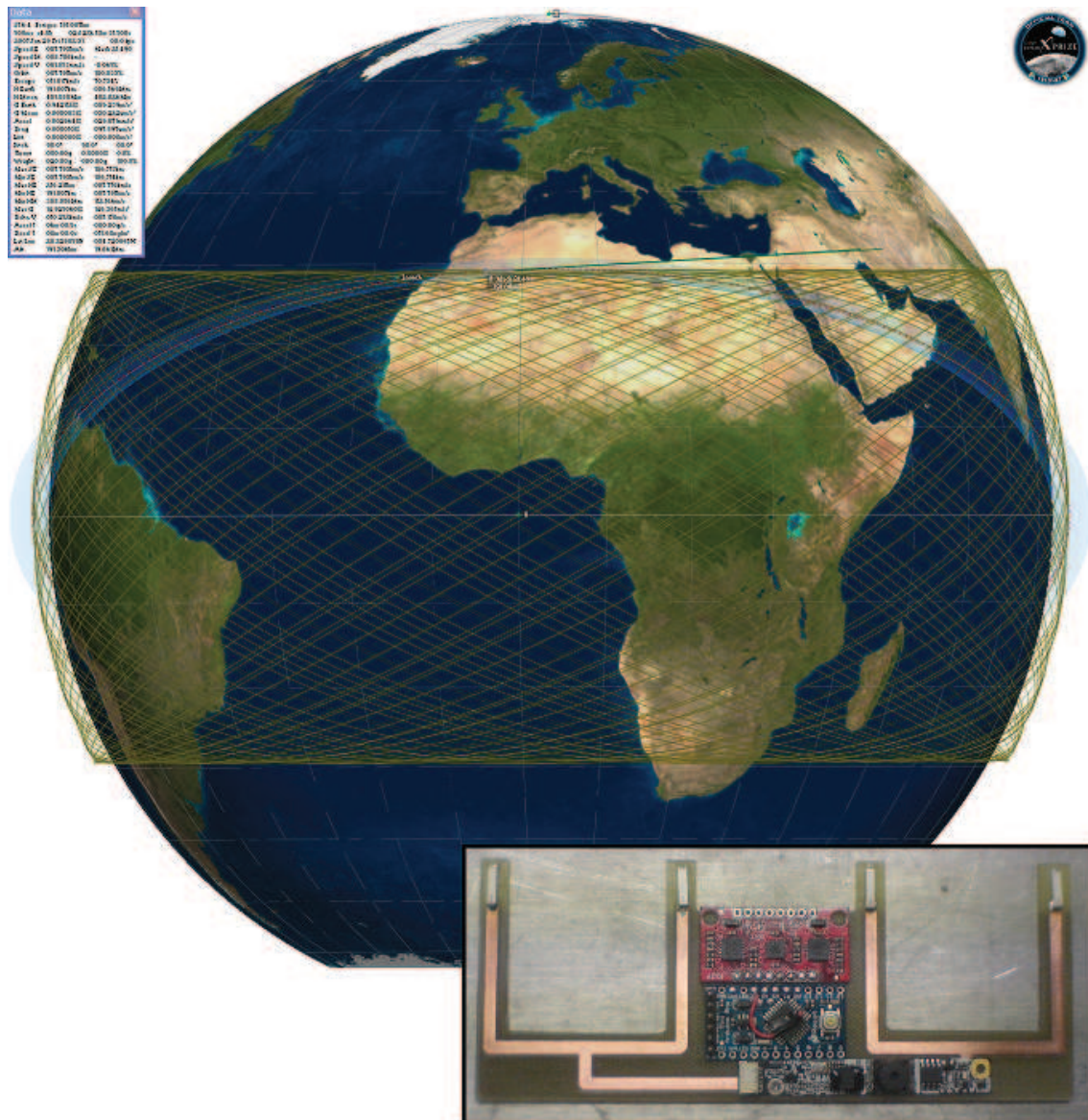
Figure 2.10: Wikisat trajectory and the technology demonstrator

2.5.2.. EPSCSat space program

The EPSCSat space program is based in the desire of the EPSC university to build and operate a femto-satellite[4]. It will be also used to validate the radio-link with the future EPSC ground station. This ground station will be a part of the GENSO[33] ground station network for small satellites in LEO Orbits at less than $300\,km$.

The EPSCSat mission is based on academic research. It will consist in putting a femto-satellite in Low Earth Orbit ($200\,km$ LEO) with a very short time of development (less than one year), a very low launch cost, high level of automatism a short operation time (less than one month) and a few communication requirements. For this reason, a HD camera and an optical system are to be added to the standard femto-satellite configuration.

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[33]GENSO http://www.genso.org/

CHAPTER 3. THE MINI-LAUNCHER

The scope of this chapter is to define how a mini-launcher can be built based on the new space paradigm and how it can be validated and qualified for the space. All this should be done keeping the costs as low as possible, at least four orders of magnitude below the cost of a commercial launcher.

3.1.. Mission design

The type of missions we would like to undertake are surveillance applications [5], fast response Earth observation (for example supporting activities on disaster management and emergency response), and dynamic space communication networks. These are mainly missions based on putting a femto-satellite in Low Earth Orbit, with a very low launch cost, quick response time for the actual launch, and short time of operation[16]. The launch operation and tracking[2] of a net of femto-satellites are also covered in this program; this would improve both time and spatial coverage. In the case of launching several femto-satellites, they would act as a swarm and not as a constellation, and hence their relative positions would not be controlled.

3.2.. Mini-launcher requirements

The WikiSat[1] organization is the one in charge of development and implementation of the femto-satellite and the mini-launcher. This organization has implemented in its designs the following directives and policies regarding to the mini-launcher:

Autonomous flight Low cost operation based in highly autonomous launch sequence.

Mobile launch-pad Flexible launch site based in a truck.

Simple ground support No complex ground support or fixed installations required.

Storage Long term storage capability without maintenance from the construction to the launch day.

Human action required A *Remove before flight* safety pin required to launch. Launch command based on the first stage ignition; no radio-remote launch policy.

Tracking Real time tracking of the first stage launcher that makes possible to recover it if a parachute is installed.

Environment No space debris policy. The disposal is part of the mission.

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[1]WikiSat http://code.google.com/p/moon-20/wiki/WikiSat_Engineering_Management_Plan

3.3.. Mini-launcher system design

The Wiki-launcher is composed by the *on-ground support system* and the *launcher system*. The launcher subsystems are currently simulated in the *Moon2.0* Google-code[2] web, freely available to the general public and runs under a BSD[3] free license.

The *on-ground support system* is composed by the following subsystems:

- Launch pad subsystem **LPS**

- First stage ignition subsystem **FSIS**

- Ground operator subsystem **GOS**

The *launcher system* is composed by the following subsystems:

- Safe launch and flight subsystem **SLFS**

- Propulsion subsystem **PS**

- Thermal control subsystem **TCS**

- Electrical power supply subsystem **EPSS**

- Attitude determination and control subsystem **ADCS**

- Position determination and navigation subsystem **PDNS**

- Trajectory simulation subsystem **TSS**

- Structure subsystem **SS**

- Jettison subsystem **JS**

---

[2]Moon2.0 project http://code.google.com/p/moon-20/
[3]BSD license http://creativecommons.org/licenses/BSD/

3.3.1.. Mini-launcher preliminary study

The design of the WikiSat mission involves in the process: the satellite, the launcher and the ground network. For this reason the WikiSat group has developed the so called *Moon2.0* simulator which includes all the basic parameters for a fast mission analysis not only for the mission or the satellite subsystems but also for the launcher. Because the difficulty to comply with the N-Prize requirements, it is mandatory to start the process design from the propellant, passing for the launcher performances, the engine, the nozzle, the staging, the trajectory optimization, the ground coverage, the thermal study, the satellite mission and the disposal for all the parts. The tool we generated is a key point to achieve this goal. All the equations of simulations were provided by the author but implemented in *Visual Basic* by Juan Martínez, one of the WikiSat team collaborators. Validations were done by means of numerical simulations using other simulators like *Agilent STK* and comparing with analytic solutions where possible.

A preliminary study shows that the best option for a low cost, low weight and best performance is to use a propellant with a specific impulse greater than 200 seconds. A complete study of the maximum altitude as a function of the launcher mass and the specific impulse, shows (Fig. 3.1) that is mainly not feasible to reach the space with low specific impulse, having greater efficiency for large launcher mass up to a limit. This study assumes for all the cases the use of two stage, solid propellant launcher which is one of the best configurations found by the authors in [15]. In addition, when we limit the launcher mass to 100 kilograms for a category of mini-launchers, the minimum specific impulse is 160 seconds. It is needed higher specific impulse in order to have a minimum DeltaV[4] of $7.760\,m/s$ to circularize the orbit. This study concludes that for a LEO orbit mission like the WikiSat, a minimum specific impulse at Sea level of 220 seconds is required to ensure at least nine turns.
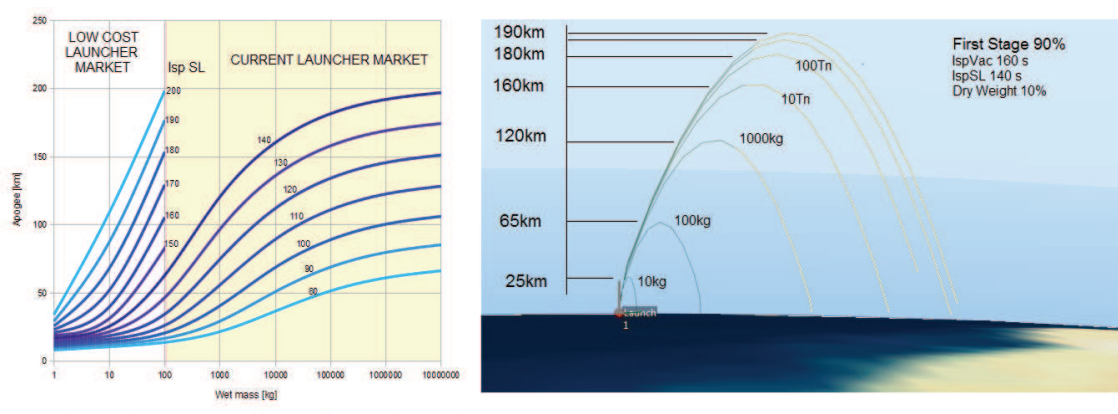
Figure 3.1: Maximum apogee as a function of the mass and the specific impulse

Many different study fields are required to design a solid propellant engine: Propellant properties, Chemical properties, Physical properties, Grain configuration related to the burn rate, etc. The following study is related to the first Wiki-launcher stage. An analogous study will be performed for the second stage in a future work.

Initial design parameters and constrains for the Wiki-launcher first stage:

---

[4]DeltaV. Change in velocity. http://en.wikipedia.org/wiki/Delta-v

- Max. diameter $0.107\,m$

- Max. propellant length $0.9\,m$

- Wet mass $10.8\,kg$

- Propellant mass $10.26\,kg$

- Solid propellant: APCP[5]

- Isp_V $260\,s$ in vacuum

- Isp_{SL} $220\,s$ at Sea Level

- $Thrust_V$ $208.6\,N$ in vacuum

- $Thrust_{SL}$ $176.5\,N$ at Sea Level

Notice that the maximum diameter has included the structure thickness which is about 2% of the total; it means that the structure, made in carbon fiber is one millimeter of thickness. Figure 3.2 presents a section of the Wiki-launcher where the second stage and an array of femto-satellites are detailed.

3.3.2.. Propellant characterization

We have selected the solid propellant APCP (Ammonium Perchlorate Composite Propellant) having the following composition:

- 70% Ammonium Perchlorate (AP)

- 10% HTPB/Curative (JOS)

- 20% Aluminum (Pure Cristaline) less than 100 microns in diameter

We have simulated the mixture in a chemical simulator to have the products during the combustion, temperature, pressure in the combustion chamber, physical and chemical properties of the resultant gases. We have used a chemical simulator inside the *Moon2.0* and an external one called *GUIPEP/PROPEP*. We assume the following hypothesis:

- Unidimensional flow for the continuity equations, energy and impulse.

- No speed flow in the nozzle intake

- Completed combustion and adiabatic combustion

- Isentropic[6] expansion in the nozzle

---

[5]APCP Ammonium Perchlorate Composite Propellant
[6]Isentropic Nozzle http://www.grc.nasa.gov/WWW/K-12/airplane/isentrop.html
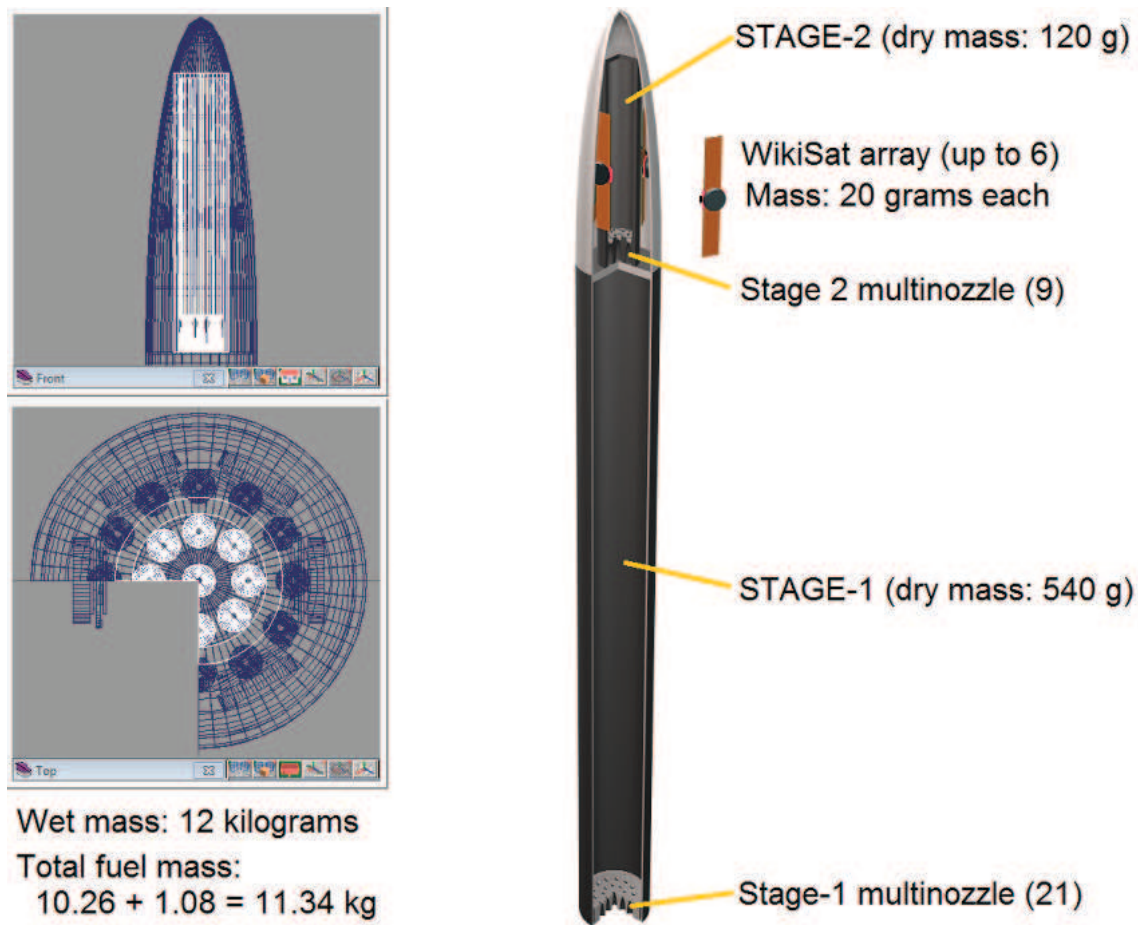
Figure 3.2: Initial Wiki-launcher mass budget for the propellant

- Homogeneous mixture of reactives and products

- Ideal gas law applied

The preliminary results shown for the *GUIDEP/PROPEP* and for the *Moon2.0* simulator:

- Grain density 1.8414 and $1.841\,kg/m^3$

- Propellant temperature 298 and $298.14\,K$

- Effective molecular weight M_{eff} 31.153 and $31.153\,g/mol$

- Combustion chamber pressure P_0 6.89 and $3.34\,MPa$

- Chamber temperature T_0 $3,646$ and $3,570.15\,K$

- Expansion ratio $\frac{A}{A^*}$ 9.59 and 5.4

- Nozzle exhaust pressure P_e 0.101 and $0.101\,MPa$

There is an important difference between the *GUIDEP/PROPEP* and the *Moon2.0* simulator related to the chamber pressure. Following calculations will take into account the higher

pressure because it means that higher safety factors will be taken in the maximum allow-able chamber pressure. The nozzle exhaust pressure imposes a more expanded nozzle. Grain density is ideal ant the temperature corresponds to the flame adiabatic temperature. Additionally, *GUIDEP/PROPEP* offers information related to the resulting products:

Effective molecular weight

$$M_{eff} = \frac{m_{System}}{1\,K \cdot m_{Products(g)}} = \frac{100\,g}{3.210\,mol} \tag{3.1}$$

Where:
$M_{eff} = 31.153\,g/(K \cdot mol)$ Effective molecular weight
$m_{System} = 100\,g$ System mass
$m_{Products(g)} = 3.210\,mol$ Gaseous products mol

Average specific heats in the mixture

Not all the combustion products are completely burned because the residence time in the combustion chamber is too short to achieve chemical equilibrium, and hence, properties through its travel are changing until the end in the nozzle outlet. The *GUIPEP* simulator allow us to calculate average value between the combustion chamber specific heat c_v where the volume is assumed constant and the nozzle outlet specific heat c_p where the pressure is assumed constant:

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

Where:
$k = 1.155\,kg/(K \cdot mol)$ Average specific heats in the mixture
$k_{nozzle} = 1.1573\,kg/(K \cdot mol)$ Specific heat in the nozzle outlet
$k_{combustion} = 1.1532\,kg/(K \cdot mol)$ Specific heat in the combustion chamber

And the Specific heats ratio[7] is:

$$\gamma = c_p/c_v \tag{3.3}$$

Where:
$\gamma = 1.1532\,kg/(K \cdot mol)$ Specific heats ratio
c_p Specific heat at constant pressure
c_v Specific heat at constant volume

---

[7]Specific heats http://www.grc.nasa.gov/WWW/K-12/airplane/specheat.html

3.3.3.. Performances of the first stage engine

Once the properties of the produced gas, the pressure and the combustion chamber temperature are known, performances of the engine can be calculated.

1. Ideal specific impulse: is the relative thrust by one unit of propellant mass for a second of combustion:

$$I_{SP} = \frac{1}{g_0} \sqrt{2T_0 \left(\frac{R'}{M_{eff}}\right) \left(\frac{k}{k-1}\right) \left[\left(1 - \frac{P_e}{P_0}\right)^{\frac{k}{k-1}}\right]} \tag{3.4}$$

Where:
$g_0 = 9.8\,m/s^2$ Acceleration at Sea level
$P_0 = 6.86\,MPa$ Combustion chamber pressure
$T_0 = 3,750\,K$ Combustion chamber temperature
$P_e = 0.101\,MPa$ Exhaust nozzle pressure
$M_{eff} = 31.153\,g/mol$ Effective molecular weight
$R' = 831,404\,J/(Mol \cdot K)$ Universal gas constant[8] of perfect gases
$k = 1.155\,kg/(K \cdot mol)$ Average specific heats in the mixture

This results in an ideal specific impulse of: $I_{SP} = 255.5813\,s$ Ideal specific impulse at Sea level

2. Thrust coefficient: Is the gain of thrust due to the gas expansion in the nozzle respect to the one in the throat.

$$C_f = \frac{(P_e - P_a)A_e}{P_0 A^*} + \sqrt{\frac{2k^2}{k-1}\left(\frac{k}{k-1}\right)^{\frac{k+1}{k-1}}\left[\left(1 - \frac{P_e}{P_0}\right)^{\frac{k}{k-1}}\right]} \tag{3.5}$$

Where:
$P_e = 0.101\,MPa$ Exhaust nozzle pressure
$P_a = 0.101\,MPa$ Atmospheric pressure assumed equal to P_e
$P_0 = 6.86\,MPa$ Combustion chamber pressure
$A_e = 0.1\,m^2$ Exhaust nozzle area
$A^* = 0.01\,m^2$ Nozzle throat area
$k = 1.155\,kg/(K \cdot mol)$ Average specific heats in the mixture

which gives a thrust coefficient of: $Cf = 1.6241$ Thrust coefficient

3. Exhaust characteristic speed: It is used a merit thermo-chemical figure for the thruster in order to evaluate the efficiency of the combustion.

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[8]Universal gas constant http://www.grc.nasa.gov/WWW/K-12/airplane/eqstat.html

$$c^* = \sqrt{\frac{T_0 \left(\frac{R'}{M_{eff}}\right)}{k \left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}}} \tag{3.6}$$

Where:

$T_0 = 3,750\,K$ Combustion chamber temperature
$R' = 831,404\,J/(Mol \cdot K)$ Universal constant of perfect gases
$M_{eff} = 31.153\,g/mol$ Effective molecular weight
$k = 1.155\,kg/(K \cdot mol)$ Average specific heats in the mixture

and this gives a speed: $c^* = 1,542.2\,m/s$ Exhaust characteristic speed

4. Discharge coefficient: Having the exit speed or exhaust characteristic speed, the discharge coefficient is calculate which gives us an idea about the combustion process inside the launcher engine. This coefficient is defined as the mass flow in the inlet of the convergent nozzle part per unit of pressure and per throat area. This coefficient is the inverse of the exhaust characteristic speed c^* and should be between $6 \cdot 10^{-4}$ and $7 \cdot 10^{-4}\,s/m$.

$$CD = \frac{1}{c^*} = 6.4843 \cdot 10^{-4}\,s/m \tag{3.7}$$

Where:
$CD = 6.4843 \cdot 10^{-4}\,s/m$ Discharge coefficient
$c^* = 1,542.2\,m/s$ Exhaust characteristic speed

5. Throat temperature and pressure conditions: Conditions in the throat area of the convergent-divergent nozzle is calculated as follows:

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

Where:
$T^* = 3,383.8\,K$ Throat temperature
$T_0 = 3,750\,K$ Combustion chamber temperature
$k = 1.155\,kg/(K \cdot mol)$ Average specific heats in the mixture

$$P^* = \frac{P_0}{\left(1 + \frac{k-1}{2}\right)^{\frac{k}{k-1}}} \tag{3.9}$$

Where:
$P^* = 3.9506\,MPa$ Throat pressure

$P_0 = 6.86\,MPa$ Combustion chamber pressure
$k = 1.155\,kg/(K \cdot mol)$ Average specific heats in the mixture

6. Exhaust conditions: Exhaust conditions are those in the exhaust nozzle in the outlet plane for example the temperature and the Mach number if the nozzle is inside the atmosphere and the exhaust temperature:

$$M_e = \sqrt{\frac{2}{k-1}\left[\left(\frac{P_e}{P_0}\right)^{\frac{k-1}{k}} - 1\right]} \qquad (3.10)$$

Where:
$M_e = 3.1365$ Exhaust Mach number
$k = 1.155\,kg/(K \cdot mol)$ Average specific heats in the mixture
$P_0 = 6.86\,MPa$ Combustion chamber pressure
$P_e = 0.101\,MPa$ Exhaust nozzle pressure

$$T_e = \frac{T_0}{1 + \frac{k-1}{2}M_e^2} \qquad (3.11)$$

Where:
$T_e = 2,068.8\,K$ Exhaust temperature
$T_0 = 3,750\,K$ Combustion chamber temperature
$k = 1.155\,kg/(K \cdot mol)$ Average specific heats in the mixture
$M_e = 3.1365$ Exhaust Mach number

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

$$\frac{A}{A^*} = \frac{1}{M}\left(\frac{1 + \frac{k-1}{2}M^2}{1 + \frac{k-1}{2}}\right)^{\frac{k+1}{2(k-1)}} \qquad (3.12)$$

Where:
$\frac{A}{A^*} = 9.751$ Optimum expansion ratio
$k = 1.155\,kg/(K \cdot mol)$ Average specific heats in the mixture
$M_e = 3.1365$ Exhaust Mach number

Finally, a summary of the first stage engine performances is presented in the Table 3.1.

Table 3.1: Engine performances for the first stage of the Wiki-Launcher

Symbol	Value	Unit	Description
Atmospheric parameters			
P_a	0.101	MPa	Atmospheric pressure assumed equal to P_e
T_a	298	K	Atmospheric temperature ($25°C$)
Combustion chamber parameters			
P_0	6.86	MPa	Combustion chamber pressure
T_0	3,750	K	Combustion chamber temperature ($3,477°C$)
A_0	0.102	m	Combustion chamber area
c_v	1.1532	$kg/(K \cdot mol)$	Combustion chamber specific heat
c^*	1,542.2	m/s	Exhaust characteristic speed
Cf	1.6241		Thrust coefficient
CD	$6.4843 \cdot 10^{-4}$	s/m	Discharge coefficient
Throat parameters			
P^*	3.9506	MPa	Throat pressure
T^*	3,383.8	K	Throat temperature ($3,110°C$)
A^*	0.01	m^2	Nozzle throat area
$\frac{A}{A^*}$	9.751		Optimum expansion ratio
Exhaust nozzle parameters			
P_e	0.101	MPa	Exhaust nozzle pressure
T_e	2,068.8	K	Exhaust temperature ($1,795°C$)
A_e	0.1	m^2	Exhaust nozzle area
M_e	3.1365	M	Exhaust Mach number
c_p	1.1573	$kg/(K \cdot mol)$	Exhaust nozzle specific heat

3.3.4.. Corrections for the ideal engine

Some corrections are required to the ideal performances because the ideal study do not take into account small loses. Corrections are provided in terms of efficiency that reduce ideal values.

1. Combustion chamber corrections: Combustion efficiency and wall heat loses reduces the theoretical pressure. However, solid propulsion has a 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 $100 \mu m$). The combustion efficiency is calculated as follows:

$$\eta^* = \frac{\bar{c}^*}{c^*} \tag{3.13}$$

Where:
$\eta^* = 0.95$ Combustion chamber efficiency
$\bar{c}^* = 1,511.4 \, m/s$ Average exhaust characteristic speed
$c^* = 1,542.2 \, m/s$ Exhaust characteristic speed

This efficiency lose is observed as a reduction of the specific impulse I_{SP}. The average exhaust characteristic speed $\bar{c}^*$ is often between 98 and 99% respect to the value of exhaust characteristic speed c^*.

2. Nozzle corrections: The flow through a nozzle is different from an ideal nozzle because the friction, heat transfer effect (mainly in the throat), no ideal gas, incomplete combustion, no axial flow, non uniform flow, etc. These loses are modeled in terms of the nozzle semi-angle of divergence (α) where for an angle of $12°$ has an efficiency of 0.99 and for an angle of $20°$ has an efficiency of 0.97:

$$\lambda = \frac{1}{2}(1 + cos\,\alpha) \tag{3.14}$$

Where:
$\lambda = 0.99$ Nozzle efficiency due to the divergence
$\alpha = 12°$ Nozzle semi-angle of divergence

Also the nozzle efficiency is modeled as a function of the discharge factor which gives an idea how well the flow goes through the throat and is the ratio between the average mass flow and the ideal mass flow:

$$\varsigma_d = \frac{\bar{\dot{m}}^*}{\dot{m}^*} \tag{3.15}$$

Where:
$\varsigma_d = 0.91$ Nozzle efficiency due to the discharge
$\bar{\dot{m}}^*$ Throat average mass flow
$\dot{m}^*$ Throat ideal mass flow

Also the efficiency can be improve having smooth surfaces and a good throat design in the inlet profile. This is the so called pressure chamber efficiency (ς_p) and is over 0.95.

$\varsigma_p = 0.95$ Nozzle efficiency due to the chamber pressure

Finally, a summary of the efficiency factors is presented in the Table 3.1 that reduces the ideal specific impulse ($Isp_{delivered}$) from an ideal engine.

$$Isp_{delivered} = \eta^* \cdot \lambda \cdot \varsigma_d \cdot \varsigma_p \cdot I_{SP} \tag{3.16}$$

Where:
$Isp_{delivered}$ Delivered specific impulse
η^* Combustion chamber efficiency
λ Nozzle efficiency due to the divergence
ς_d Nozzle efficiency due to the discharge
ς_p Nozzle efficiency due to the chamber pressure

$I_{SP} = 255.5813\,s$ Ideal specific impulse at Sea level

Other method to obtain the delivered specific impulse is using the following equation:

$$Isp'_{delivered} = \frac{c^* \cdot Cf}{g_0} \tag{3.17}$$

Where:
$Isp'_{delivered}$ Simplified delivered specific impulse
$c^* = 1,542.2\,m/s$ Exhaust characteristic speed
$Cf = 1.6241$ Thrust coefficient
$g_0 = 9.81\,m/s^2$ Normalized acceleration at Sea level

The result is closer to the previous method with 99.78% of accuracy.

Table 3.2: Corrections for the ideal engine due to efficiency factors

Symbol	Value	Description
η^*	0.95	Combustion chamber efficiency
λ	0.99	Nozzle efficiency due to the divergence
ς_d	0.91	Nozzle efficiency due to the discharge
ς_p	0.95	Nozzle efficiency due to the chamber pressure
$Isp_{delivered}$	$214.37\,s$	Delivered specific impulse

3.3.5.. Materials and thermal considerations

Following, and knowing the conditions and constrains for the combustion chamber, material selection to build the chamber is studied. The solid propellant can be used as a structure but in case of a fracture or a bubble, a hot point could happens. In this case is when a external structure and a thermal protection is required to ensure the stability of the solid propellant engine for few minutes until the burn out. We present two options summarized in table 3.3 with thermal expansion[9] and emissivity[10] values:

First option is to use aluminum as a structure with asbestos[11] and phenol-formaldehyde as a woven thermal protection inside.

Second option is to use Carbon-fiber[12] or CCC[13] as a structure with Pyrolytic graphite[14] gasified deposited inside as a thermal protection.

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[9]Thermal expansion values http://www.wisetool.com/designation/te.htm

[10]Emissivity values http://www.infrared-thermography.com/material-1.htm

[11]Asbestos http://www.matweb.com/search/DataSheet.aspx?MatGUID=1bfa107092794f5eaf1aef9b82801528

[12]Carbon-fiber http://www.matweb.com/search/DataSheet.aspx?MatGUID=ca447b2e5d934e9b8ea1636e71a6d6e0

[13]Carbon-Carbon composites http://www.substech.com/dokuwiki/doku.php?id=carbon-carbon_composites

[14]Pyrolytic Graphite http://www.matweb.com/search/DataSheet.aspx?MatGUID=07d26ab4d0e349aea3a200f447e9d2e1

Table 3.3: Mechanical and thermal properties of materials for the launcher structure

Property	Unit	Al	Asbestos	Alumina	CCC	Pyrolytic graphite
Density	g/cm^3	2.80	1.76	1.60	1.81	2.10
Max. Temperature	$°C$	660	1,600	1,700	3,650	3,650
Yield strength	MPa	280	42	200	4,137	80
Young's modulus	GPa	68	70	375	242	20
Emissivity	$(0\,to\,1)$	0.05	0.80	0.68	0.95	0.97
Thermal conductivity	$W/(m \cdot K)$	210	40	35.4	101	190
Thermal expansion	$\mu m/(m \cdot °C)$	23	3.4	7.9	1.3	6.0
Thermal diffusivity	$\cdot 10^{-7} m^2/s$	8,420	1.76	120	17	36

It is well known the health problems produced by asbestos when it is carelessly handled by humans. In this sense we want to recommend to substitute this product by a non-asbestos material like a refractory sheet cylinder[15] type RS-101 based on a ceramic fiber reinforcement structural alumina Al_2O_3 and SiO_2. Working temperature is up to $1,260°C$ and density $1.6\,g/cm^3$ and the structure could be made in CCC. Following calculations are made assuming the first option but similar procedure is applied for the second option.

1. Combustion chamber thickness The structure of the Wiki-launcher first stage is a tube where the thickness wall e limit is imposed by the expected pressure and the yield strength of the material used.

$$e = \frac{f \cdot d \cdot p}{2 \cdot \sigma} \tag{3.18}$$

Where:
$e = 0.0026\,m$ Combustion chamber thickness
$f = 2.0$ Safety factor
$d = 0.107\,m$ External diameter
$p = 6.89\,MPa$ Maximum internal pressure
$\sigma = 280\,MPa$ Yield strength

2. Ablation speed The ablation speed v_a mainly depends on the ablation temperature of the insulator and the combustion chamber temperature. The heat flow per unit area is:

$$\alpha = 6e_h - 4 \cdot T_0^{0.3} \cdot m_s^{0.905} \tag{3.19}$$

Where:
$\alpha = 515.72\,J/(s \cdot m^2)$ Heat flow per unit area
$e_h = 1.4\,kJ/kg$ Specific heat
$T_0 = 3,750\,K$ Combustion chamber temperature
$m_s = 285.62\,kg/m^2$ Mass per area

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[15]Refractory sheet cylinder RS-101 http://www.zrci.com/rs101.htm

Hence, knowing the heat flow per unit area, the ablation speed is:

$$v_a = \frac{\alpha(T_0 - T_a)}{Q_a \cdot \rho}$$
(3.20)

Where:
$v_a = 0.00015\,m/s$ Ablation speed
$\alpha = 515.72\,J/(s \cdot m^2)$ Heat flow per unit area
$T_0 = 3,750\,K$ Combustion chamber temperature
$T_a = 1,600\,K$ Ablation temperature
$Q_a = 4,200\,kJ/kg$ Ablation heat
$\rho = 1.76\,kg/m^3$ Insulator density

3. Erosion thickness The erosion thickness e_e is the part of the insulator dedicated to the ablation in order to protect the Wiki-launcher fist stage and depends on the ablation speed and the burnout time.

$$e_e = v_a \cdot t_{burnout}$$
(3.21)

Where:
$e_e = 0.0180\,m$ Insulator thickness
$v_a = 0.00015\,m/s$ Ablation speed
$t_{burnout} = 120\,s$ First stage burnout time

4. Pyrolyzed thickness The Pyrolyzed thickness e_p is the part of the insulator that remains burned inside the insulator.

$$e_p = 2\sqrt{\frac{a \cdot t_{burnout}}{2}}$$
(3.22)

Where:
$e_p = 0.0065\,m$ Insulator thickness
$a = 1.75 \cdot 10^{-7}\,m^2/s$ Thermal diffusivity
$t_{burnout} = 120\,s$ First stage burnout time

5. Thermal insulator thickness The thermal insulator of the Wiki-launcher fist stage is a tube where the thickness wall e_t limit is imposed by the Erosion thickness and the Pyrolyzed thickness.

$$e_t = \beta(e_e \cdot e_p)$$
(3.23)

Where:
$e_t = 0.0294\,m$ Insulator thickness

$\beta = 1.2$ Safety factor
$e_e = 0.0180\,m$ Erosion thickness
$e_p = 0.0065\,m$ Pyrolyzed thickness

6. Burst pressure the first stage Assuming the use of an aluminum structure as stated in the first option, the burst pressure (P_{burst}) is calculated as follows:

$$P_{burst} = \frac{2 \cdot e \cdot \sigma}{d} \tag{3.24}$$

Where:
$P_{burst} = 13.56\,MPa$ Burst pressure
$e = 0.0026\,m$ Combustion chamber thickness
$\sigma = 280\,MPa$ Yield strength
$d = 0.107\,m$ External diameter

This pressure is higher than the maximum operative pressure in the combustion chamber of $P_0 = 6.86\,MPa$ with a safety factor of $f = 2.0$. A pressure test has to be done in this sense in order to validate the first stage structure.

3.3.6.. Thrust curve

The design of the grain was done with a spread sheet simulator called PFC-Burn[16] which provides a thrust curve based on points. This curve showed in Figure 3.3, can be converted to the format *ENG* that can be read by *Moon2.0* and can run the simulation with this new engine.

3.4.. Mini-launcher detailed subsystem design

3.4.1.. On-ground support system

The *on-ground support system* is composed by a list of subsystems:

LPS Launch pad subsystem. Launch pad based on a truck. The launching point is mobile and no large facilities are required to launch the Wiki-launcher. A GPS cool start is required before every launch. This property makes very flexible the kind of missions to perform by the launcher.

FSIS First stage ignition subsystem. A physical mechanism based on a flight safety pin is installed to avoid remote launch. The launch sequence can only start with a ground ignition in order to save weight and increase the safety of the operation.

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[16]PFC-Burn.xls http://www.nakka-rocketry.net/soft/pfc-burn.xls

PFC-BURN
This spreadsheet computes **Kn** (ratio of burning area to throat area) for a propellant grain with a **Pseudo-Finocyl** configuration. The Kn is computed at 45 equidistant points to simulate recession of the burning surface.
Note that the basis of surface area calculations is solely geometrical.
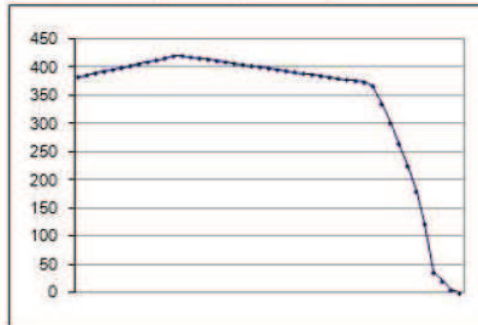Enter data in *blue* font.

Grain length	180	mm
Grain diameter	40	mm
Core diameter	12	mm
Fin width	2,5	mm
Fin depth	8	mm
Number of fins	8	
Nozzle throat dia.	10	mm

SOLVE   RESET

Solution found.

0.148

max Kn   422

Written by: R.Nakka
Date: abr-04
Version: Beta-1

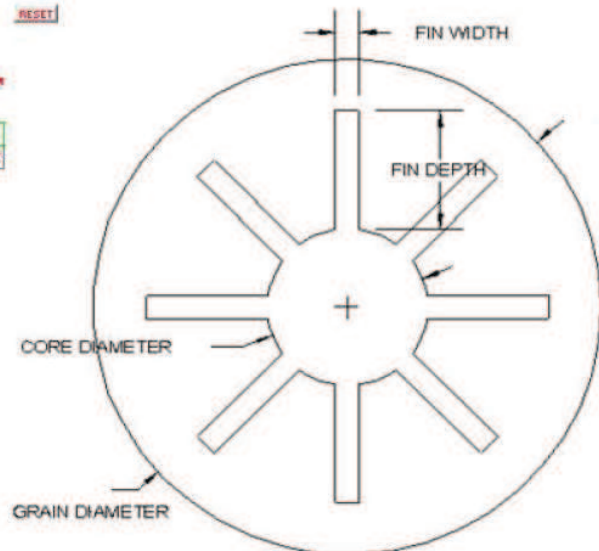| Point	Kn
0	384
1	387
2	390
3	394
4	397
5	400

Figure 3.3: Grain design and thrust curve using PFC-burn.xls spread sheet

GOS Ground operator subsystem. Single human ground operator is required. This person can not interact with the launcher when it is in flight, being responsible for launching, monitoring and tracking the trajectory. Moreover, this person is also responsible for communicating the 'femto-satellite successful deployment' or the 'launcher miss path' to the civil airworthiness authorities and the client.

3.4.2..   Launcher system

The launcher system is composed by a list of subsystems:

SLFS Safe launch and flight subsystem. An autonomous self-destroy mechanism is installed based on the criterion of evaluating if the launch path followed by the launcher is inside a safe corridor. Among other definition criteria, this corridor will take into account population density to minimize the hazards due to possible debris. This mechanism is only used for low altitudes in order to avoid space debris, even if the on-orbit residence would be short due to the low altitude.

PS Propulsion subsystem. The propulsion is based on high specific impulse solid propellant and a multi-nozzle configuration.

TCS Thermal control subsystem. This subsystem is based on passive large thermal range of launcher components. Temperature over-trip is controlled in conjunction with the

electrical power supply subsystem and thermoelectric cells and the structure sub-system as a cooler.

EPSS Electrical power supply subsystem. Low capacity batteries are used for the starting initial phase, while thermoelectric cells are used to feed the launcher electrical needs and the femto-satellite during the launch phase. These are also used in synergy with the thermal control subsystem.

ADCS Attitude determination and control subsystem. This subsystem is composed by an IMU (Inertial Measurement Unit), Barometric sensor, Light sensors and Thrust vector control actuated by electrical servos. Also works in synergy with the Propulsion subsystem.

PDNS Position determination and navigation subsystem. This subsystem is composed by a single GPS and a single MCU[17]. The positioning system is in charge of determine where is the launcher respect to the planet and can be corrected by external inputs apart of the GPS. The navigation part is in charge of follow the predicted trajectory to ensure the success of the mission.

TSS Trajectory simulation subsystem. A simple on-board simulation computer is used for real-time trajectory determination in order to compute the trend of the launcher. Also used in synergy with the Position determination and navigation subsystem.

SS Structure subsystem. Absorbs dynamic physical and thermal loads. Also used in synergy with the thermal control subsystem.

JS Jettison subsystem. Separate the first stage from the rest of the launcher in a safe way.

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[17]MCU Main Control Unit

3.5.. Mini-launcher trajectory and satellite deployment

The Mini-Launcher is a very light, very small, less than 100 kg launcher able to put a small femto-satellite in a LEO orbit of 200 kilometers. The time to orbit is around 6 minutes and 30 seconds in four phases: Fist stage, ballistic flight, Second stage and finally satellite deployment. Figure 3.4 show details about these events.
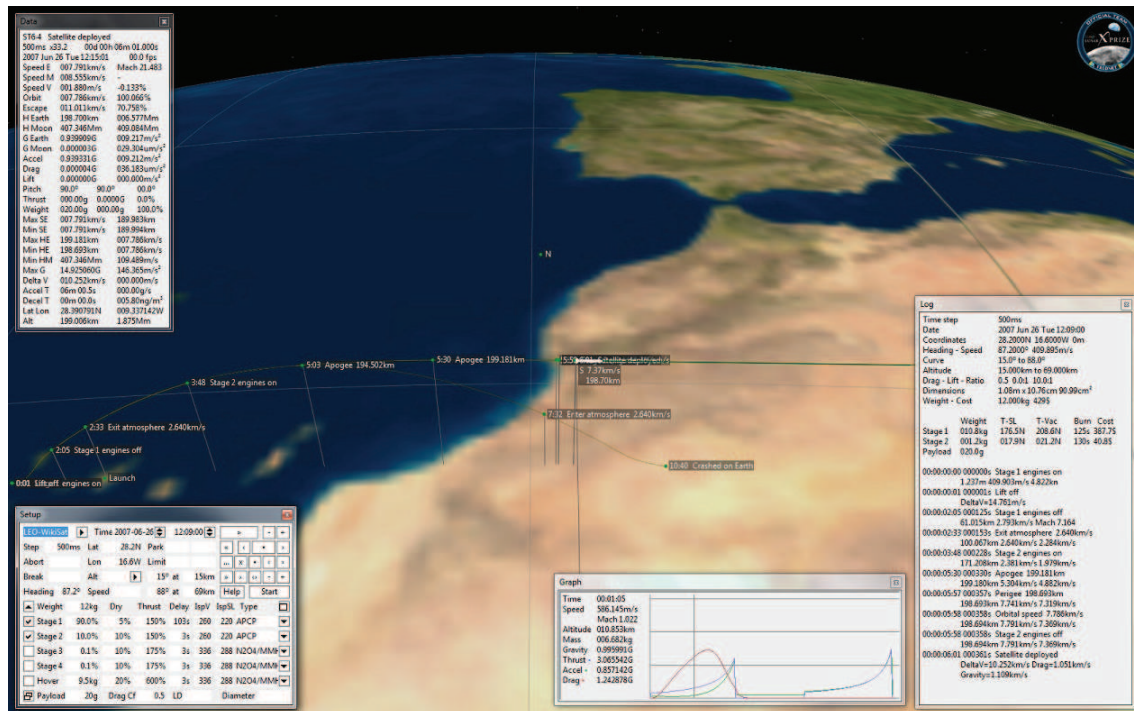
Figure 3.4: WikiLauncher trajectory

3.5.1.. Wiki-Launcher mission analysis

The launcher lift-off at 4:00 am in August 30, 2010. After 0:29 seconds, the sonic boom [18] happens at $4.62\,km$ of altitude and reach the space in 2:03 minutes at speed of $2,575\,m/s$. Three seconds later the first engine burns-out and jettisoned to crash in the Sahara desert in 28.3498N, 009.4049W. This stage burns during the reentry but if a parachute is deployed, could be recovered. The ballistic phase least for $130\,s$, reaching an altitude of $230\,km$ when the second engine starts at 4:06 minutes from the launch moment. In 6:24 minutes the second engine has burn-out reaching to the orbital speed of $7,756\,m/s$ and an altitude of $250\,km$ then the satellite is deployed. The maximum acceleration is $12\,G$, few seconds before the second burn-out. The maximum acceleration in the first stage is $7.9\,G$ few seconds before the burn-out. The period is $5,367\,s$ (1:29:27) and the shift angle is 22.3754 degrees towards the West each orbit.

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[18]Sonic boom http://en.wikipedia.org/wiki/Sonic_boom

3.5.2.. Wiki-Launcher hazard trajectory study

As a matter of fact, if the launcher goes outside the atmosphere (greater than $15\,km$), if something is wrong in the trajectory, no matter the trajectory, the temperature reached in the reentry is so high that it is burned and no debris are expected. Evermore, if a parachute is installed in the first stage, then it is possible to brake the reentry speed in such a way that the temperature due to the drag is no so high. If so, the recovery is feasible thanks to the radio beacon of the tracking system.

Only in the initial path a specially attention has to be taken (at least from the point of view of safety issues). For this reason a safe launch and flight subsystem is installed. If an auto-destroy action is needed, making a hole in the main case, the launcher explodes. This is an autonomous self-destroy mechanism based on the criterion of evaluating if the launch path followed by the launcher is inside a safe corridor. This corridor is both, lateral an vertical profile. Different areas will be taken into account depending on the launching point. This hazard area has a typically radius of $200\,km$. A good approach is launching from the international sea in a boat because different laws are applied respect to launching inside a country. In this sens, international sea makes easier the hazard considerations.

We have selected a late hour ($4am$) because the air traffic is very low so the impact for normal operations are very small. The airspace shall be closed the during duration of the launch window. Special control has to be taken by the government during the launch. The military sector has to be advised in order to coordinate any action. Also in case of recover the first stage, impact area shall be controlled by the corresponding government.

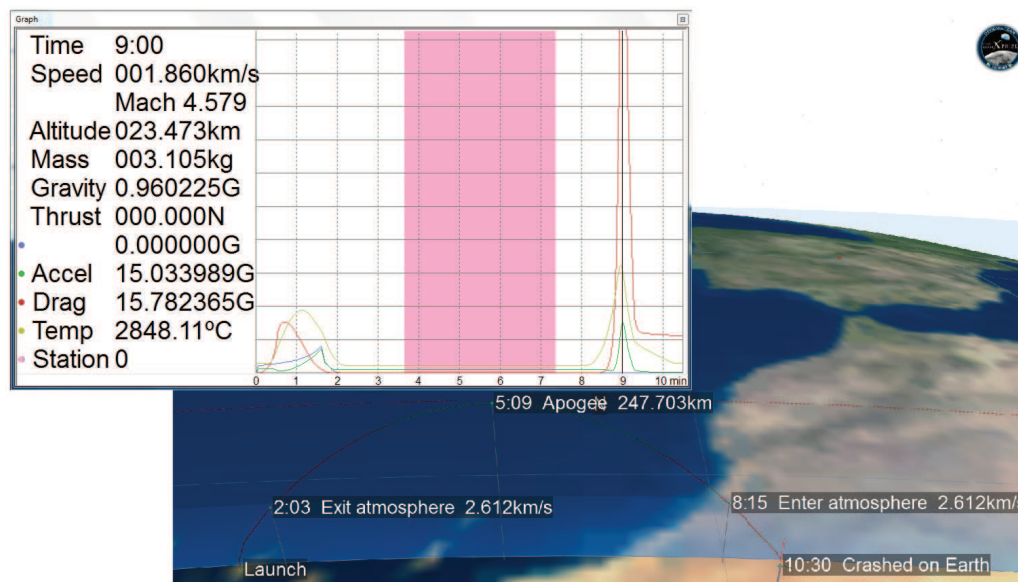Figure 3.5: Wiki-Launcher trajectory in the re-entry. Temperature vs time

Figure 3.5 shows the temperature of the first stage reentry after the second stage jettison reaches a maximum of $2,800\,°C$ and the temperature is very high for two minutes.

1. Ablation speed The ablation speed v_a mainly depends on the ablation temperature of the insulator and the re-entry temperature. The heat flow per unit area is:

$$\bar{\alpha} = 6e_h - 4m_s^{0.905} \int_{t_0}^{t_1} T(t)^{0.3} dt \tag{3.25}$$

Where:
$\bar{\alpha} = 987.52 \, J/(s \cdot m^2)$ Average heat flow per unit area
$e_h = 1.4 \, kJ/kg$ Specific heat
$T(t) < 3,073 \, K$ Re-entry temperature as a function of time
$t_0 = 0 \, s$ Re-entry time when above T_a
$t_1 = 27 \, s$ Re-entry time when is lower than T_a
$m_s = 285.62 \, kg/m^2$ Mass per area

Hence, knowing the average heat flow per unit area, the average ablation speed is:

$$\bar{v}_a = \int_{t_0}^{t_1} \frac{\bar{\alpha}(T(t) - T_a)}{Q_a \cdot \rho} dt \tag{3.26}$$

Where:
$\bar{v}_a = 0.00020 \, m/s$ Average ablation speed
$\bar{\alpha} = 987.52 \, J/(s \cdot m^2)$ Average heat flow per unit area
$T(t) < 3,073 \, K$ Re-entry temperature as a function of time
$T_a = 1,600 \, K$ Ablation temperature
$t_0 = 0 \, s$ Re-entry time when above T_a
$t_1 = 27 \, s$ Re-entry time when is lower than T_a
$Q_a = 4,200 \, kJ/kg$ Ablation heat
$\rho = 1.76 \, kg/m^3$ Insulator density

2. Ablation time for the Wiki-Launcher Now we calculate the time of ablation in order to know if the re-entry time is enough to destroy the fist stage Wiki-launcher.

$$t_{ablation} = \frac{e_e}{\bar{v}_a} \tag{3.27}$$

Where:
$t_{ablation} = 90 \, s$ Wiki-launcher ablation time
$e_e = 0.0180 \, m$ Insulator thickness
$\bar{v}_a = 0.00020 \, m/s$ Average ablation speed

The reentry lasts for 2 minutes and the ablation time for the Wiki-launcher is 90 seconds that means a complete destruction if no parachute is installed in the first stage.

3.6.. Mini-launcher space qualification

A similar process as done for the WikiSat has to be performed in the Wiki-launcher program for space qualification as that which was done for the qualification of the femto-satellite. Many components validated in the Wikisat program are suitable for the launcher. New qualifications are related to the propellant issues which have a specific program of validation and qualification based on progressive advances gained in tests and actual launches.

The specific impulse of a solid propellant depends on the chemical products but also depends on the geometry of the grain, as well as on the nozzle design. At the same time, any change in the specific impulse implies, a redesign of the trajectory and the launcher properties, thus becoming an iterative process. For this reason, a solid propellant characterization is required, and we have done that employing a device called *Constant Pressure Combustion Chamber*. This device uses a high speed thermal camera in order to record the burning sequence of an small specimen. That way it is possible to characterize the performance of a new propellant and greatly speed up the development phase. Moreover, it means a large saving in propellant because it is not required to burn an entire stage. A small quantity is enough to adjust the *Moon2.0* simulator and know the rest of parameters. It is also a way to greatly reduce the environmental impact.

Other validations (as for example those related to the dynamics of the launcher) are only feasible in flight. Initial launches usually imply a high risk because the learning curve of this method. Large safe areas will be required in the first launch, and in this regard we have started negotiations to use the facilities of *El Arenosillo*, a Spanish small-rocket launch base in *Huelva*.

A good example of this problems related to the dynamic test is COTS[19] components validations. As an illustration we can consider the GPS, which can only be tested on ground, but its actual behavior at the typical operational speed can not be tested (or can be so at the cost of a considerable difficulty) until the first launch takes place. For this reason, the trajectory determination for this first launch mainly will be based on inertial means and not on positional means. As soon as this GPS is validated in flight, both methods will improve the accuracy of the launcher trajectory but never compromises the integrity of the system.

We have programed few engine tests: Propellant test using the CPCC[20], real nozzle test and ground engine test (propellant and nozzle). Then, for the late summer, a first launch test will be done to see if all subsystems in the first stage work well in real conditions. This first attempt will replace the weight of the second stabe by a parachute in order to recover the first stage for analyze the effects. When second stage is developed, similar tests will be done at the end of this year. Final launch has to be done and complete their 9th orbit before 19:19:09 (GMT) on the 19th September 2011 as stated by N-Prize requirements.

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[19]COTS Commercial-off-the-shelf
[20]CPCC Constant Pressure Combustion Chamber

CHAPTER 4. CONCLUSIONS

4.1.. General conclusions

We have explored in a practical case a new paradigm for the development of very small space missions, such as a femto-satellite. This new way of unfolding the development can result, if successful, in the birth of a new market for femto-satellites (no satellite of this class has been launched up to now). The missions of this kind of satellites should be very focused on a single goal, but the system can perform most of the typical activities undertaken by its larger mass cousins.

The extremely reduced cost of the satellite development, construction, launch and operation can become a new standard in some space areas, and could act as a facilitator to the entrance into space activities of new actors. After publication and demonstration of these technologies and approaches, a large number of new potential users could appear; these new potential users would not be able to undertake a space mission under the current conditions.

Nevertheless, it must be stated that very small satellites does not directly enter into competition with standard satellites. More interestingly, they could open new markets and opportunities, and could be used for education, remote sensing and even communications on an extremely low cost and complexity. But this does not mean that the new approach must be irrelevant to more usual space systems design, as the demonstration that non-space qualified parts can be usefully exploited in space conditions could allow reductions of mass, cost and complexity in larger satellites, thus enabling more missions, and with greater capabilities.

We have shown that it is possible:

- to design, test, and build a femto-satellite (with less than 20 grams) that is able to perform a short duration mission

- to design a small rocket launcher able to insert a femto-satellite into LEO with a height of 200 km

- that the combination of femto-satellites and mini-launchers have a very short response time, that would be appropriate in several circumstances

- that the new space payload paradigm can contribute to a simpler, faster and much cheaper way of producing very small missions (and perhaps to larger and more complex satellites).

We expect that new markets will appear once the technology demonstrators have shown the feasibility of femto-satellites as capable space systems.

The key factors that make this new approach feasible are:

Space Payload Paradigm The integration of the launcher and the satellite in the design cycle of a space mission. The use of new technologies validating COTS[1] components for space qualification, using the full range of the market to the mission's need.

Open Source Approach The design based on the community and not in the industry; a community of people motivated by the enjoy of a common challenge which is a benefit for the whole community. This approach favors the use of local resources and the competition frame.

Moon2.0 The use of tools like a full cycle simulator from the propellant design, the launcher physical parameters, staging and trajectories, until the ground coverage; with hardware in the loop able to test real hardware. Testing facilities allow us to validate and qualify for the space new components. These facilities can be used for the cube-sat market as well.

CPCC Constant Pressure Combustion Chamber able to characterize new solid propellants using small specimens extracted from the manufacturing procedure. Propellants with a specific impulse greater than 200 seconds have the potential to give us access to the space.

PCBs The possibility of building our own boards with SMD components reduces the cost of electronic devices in 50% and gives us some independence from providers and reduces the extra cost of taxes when we import these components.

4.2..   Environmental impact

The development of these space programs, WikiSat and EPSCSat, based on the community and not in the industry, makes feasible a cheap development with a high sense of satisfaction[7], collaborating for the growing of the global and open knowledge of new technology. The participation of the WikiSat team in prizes and competitions, permits to explode some synergies and feed the spirit of the team.

A good saving of power and efficient use of resources is reached respect to the industrial I+D frame, having by definition a less dramatic environmental impact. This situation is met when the investment of knowledge and innovation required is not easy to make profitable in the beginning (only in long term, more than 10 years) and is not justified to be handled by governments.

4.3..   Future work

Technology transfer from the university and the amateur community to the catalonian industry is expected as well as a few international patents. The knowledge of these technologies will remain open but in contrast, in order to guarantee a good implementation in the new market, exploitation of new patents and finally establishing a competitive market.

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[1]COTS Commercial-off-the-shelf

A local industry will export these new technologies to other countries until a stable market is reached. A similar growing model as that of the mobile phones and automotive sectors is expected because both are the promoters of these COTS[2] products.

As a future work we want to complete the *WikiSat space program*, the *EPSCSat space program* and also extend this market to the Moon exploration with the strategic *PicoRover to the Moon* program.

4.4.. Acknowledgment

We want to acknowledge the support that *Team FREDNET*, contestant for the *Google Lunar X-Prize*, has awarded us since the beginning of the PicoRover program and the parallel development that this *Open Source* community maintain. Also we want to recognize the labor of the X-Prize foundation through its *Revolution in competition* philosophy in behalf of a cheaper and accessible space exploration.

We want to acknowledge the unconditional support performed by the Catalonian university (UPC) and the *Escuela Politècnica Superior de Castelldefels*. The work of every student have made this thing happens. I want thanks the innovative mind of Victor Kravchenko, a college from the Master in Aerospace Science and Technology in the UPC[3].

The preliminary engine study for the Wiki-launcher was done by *Carlos Ivan Vera*. The author wants to tanks his orientations in this subject. The design of the *Constant Pressure Combustion Chamber* (see Annex D) is mainly by *Victor Kravchenko*. *Albert Moga* collaborates with him in the implementation of this device. The hazard trajectory study was began by *Esteve Bardolet*. Juan Martínez has programed the *Moon2.0* simulator under my design recommendations.

Also we want to thank the help and economical support given by families, friends, collaborators and partners; their donations and collaborations make feasible this *Open Source* approach. Only this support makes feasible a cheap, fast and amateur approach in comparison to an industrial approach.

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[2]COTS Commercial-off-the-shelf
[3]MAST http://mastersoficials.upc.edu/mast/

CHAPTER 5. GLOSSARY

NOTE: Wikipedia references were revised for accuracy in the scope of this master thesis.

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Accelerator. http://en.wikipedia.org/wiki/Accelerator

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ADXL335. 3D accelerometer. http://www.sparkfun.com/datasheets/Components/SMD/adxl335.pdf

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Amax D12 200938-08-102-e. https://shop.maxonmotor.com/

APCP. Ammonium Perchlorate Composite Propellant. http://en.wikipedia.org/wiki/APCP

APRS. http://en.wikipedia.org/wiki/Automatic_Packet_Reporting_System

Asbestos. http://www.matweb.com/search/DataSheet.aspx?MatGUID=1bfa107092794f5eaf1aef9b82801528

ASTRONAUTIX. http://www.astronautix.com/props/index.htm

AVR. Augmented Virtual Reality. http://en.wikipedia.org/wiki/Augmented_reality

BAiE. Barcelona Aeronautics and Space Association. http://www.bcnaerospace.org/

balloonTraj. http://weather.uwyo.edu/polar/balloon_traj.html

BSD license. http://creativecommons.org/licenses/BSD/

Carbon-fiber. http://www.matweb.com/search/DataSheet.aspx?MatGUID=ca447b2e5d934e9b8ea1636e71a6

CCC. Carbon-Carbon composites http://www.substech.com/dokuwiki/doku.php?id=carbon-carbon_composites

Copper. http://www.matweb.com/search/datasheet.aspx?MatGUID=9aebe83845c04c1db5126fada6f76f7e

COTS. Commercial-off-the-shelf

CPCC. Constant Pressure Combustion Chamber by Victor Kravchenko

CR2450. http://www.houseofbatteries.com/pdf/CSO-CR2450

DeltaV. Change in velocity. http://en.wikipedia.org/wiki/Delta-v

ECmax D16 283825-08-173-e. https://shop.maxonmotor.com/

EIRP. Equivalent isotropically radiated power. http://en.wikipedia.org/wiki/Equivalent_isotropically_radiated_pow

Embedded. http://en.wikipedia.org/wiki/Embedded_system

Emissivity. http://en.wikipedia.org/wiki/Emissivity

Emissivity, Table of. http://www.infrared-thermography.com/material-1.htm

Equation of state for an ideal gas. http://www.grc.nasa.gov/WWW/K-12/airplane/eqstat.html

ESA. European Space Agency. http://www.esa.int/

eZ430. http://focus.ti.com/lit/ug/slau227e/slau227e.pdf

FAR-101. http://www.sscl.iastate.edu/far101

Femto-satellite. A less than 100 grams satellite

Fiberglass. http://en.wikipedia.org/wiki/Fiberglass

FLOP. FLoating point Operations Per Second. http://en.wikipedia.org/wiki/FLOP

FOAM. http://www.matweb.com/search/DataSheet.aspx?MatGUID=91d44cae736e4b36bcba94720654eea

FPGA. Field Programmable Gate Array. http://en.wikipedia.org/wiki/FPGA

G1-1.0-127-1.27 datasheet. http://www.tellurex.com/pdf/G1-1.0-127-1.27.pdf

Genso. Global Educational Network for Satellite Operations. http://www.genso.org/

GEO. Geostationary Orbit. http://en.wikipedia.org/wiki/Geostationary_orbit

Glass-fiber. http://www.matweb.com/search/DataSheet.aspx?MatGUID=d234c9575f63487cbd69a0df8bfc

Google Earth. http://earth.google.com/

GPS. http://www.sparkfun.com/commerce/product_info.php?products_id=8936

Gyro. http://en.wikipedia.org/wiki/Gyroscope

HD camera 1300x1040 pixels. http://www.sparkfun.com/commerce/product_info.php?products_id=8668

HIM-800 weather balloon. http://www.meteorologyshop.eu/Radiosonding_balloons/ENG_276_EUR_38_632

Hohmann transfer orbit. http://en.wikipedia.org/wiki/Hohmann_transfer_orbit

IMU 6DOF http://www.sparkfun.com/commerce/product_info.php?products_id=9431

Isentropic Nozzle. http://en.wikipedia.org/wiki/Isentropic_process http://www.grc.nasa.gov/WWW/K-12/airplane/isentrop.html

ITAR. International Traffic in Arms Regulations. http://en.wikipedia.org/wiki/ITAR

KINGMAX 4G memorystick. http://www.kingmax.com/material/download/3/superstick.pdf

KML file format. http://en.wikipedia.org/wiki/KML

L-1 Lagrangian point. is the region where Earth and Moon gravity field are equal.

LEO. Low Earth Orbit. http://en.wikipedia.org/wiki/Low_Earth_orbit

Li-Ion batt. http://en.wikipedia.org/wiki/Lithium-ion_battery

LNA. Low Noise Amplifier

LPR530AL. XY-Axis gyro. http://www.sparkfun.com/datasheets/Sensors/IMU/lpr530al.pdf

LY530AL. Z-Axis gyro. http://www.sparkfun.com/datasheets/Sensors/IMU/LY530ALH.pdf

MAST. Master in Aerospace Science and Technology. http://mastersoficials.upc.edu/mast/

MCU. Main Control Unit

MEO. Medium Earth Orbit. http://en.wikipedia.org/wiki/Medium_Earth_orbit

Mini-launcher. A less than 100 kg launcher

Modulus of Elasticity. http://en.wikipedia.org/wiki/Elastic_modulus

Moon2.0 project. http://code.google.com/p/moon-20/

N-Prize. http://en.wikipedia.org/wiki/N-Prize

NOTAM. Notice to AirMan

NASA SP-3094 http://hdl.handle.net/2060/19750016776

NMEA. National Marine Electronics Association. http://en.wikipedia.org/wiki/NMEA_0183

ONAVI 23503-400-0100-A. http://www.o-navi.com/Gyrocube3A_4.pdf

PFC-Burn.xls. http://www.nakka-rocketry.net/soft/pfc-burn.xls

Photodiode. http://en.wikipedia.org/wiki/Photodiode

Polyester. http://en.wikipedia.org/wiki/Polyester

Pyrolytic Graphite. http://www.matweb.com/search/DataSheet.aspx?MatGUID=07d26ab4d0e349aea3a200f4

Refractory sheet cylinder. RS-101 http://www.zrci.com/rs101.htm

RS-101. Refractory sheet cylinder. http://www.zrci.com/rs101.htm

SAR. Synthetic Aperture Radar. http://en.wikipedia.org/wiki/Synthetic_aperture_radar

SEN-09088. Light sensor. http://www.sparkfun.com/datasheets/Sensors/Imaging/SEN-09088-datasheet.pdf

SiRF. http://en.wikipedia.org/wiki/SiRF

SM-0103. Permanent magnet. http://www.eamagnetics.com/library/EAM-Standard-Alnico-Magnets.pdf

SMA661AS. Low noise amplifier. http://www.sparkfun.com/datasheets/Components/SMD/sma661as.pdf

SNR. Signal to noise ratio. http://en.wikipedia.org/wiki/Signal-to-noise_ratio

SONCEBOZ 6415. http://www.sonceboz.com/medias/produits/fiches-techniques/6415.pdf

Sonic boom. http://en.wikipedia.org/wiki/Sonic_boom

SP-3094 http://hdl.handle.net/2060/19750016776

Space Payload Paradigm. Is the engineering process of designing a space mission around its payload and not around the space industry.

Specific heats. http://www.grc.nasa.gov/WWW/K-12/airplane/specheat.html

SPENVIS. http://www.spenvis.oma.be/

STK. Satellite Tool Kit. http://www.stk.com/

Thermal conductivity. http://en.wikipedia.org/wiki/Thermal_conductivity

Thermal expansion, Table of. http://www.wisetool.com/designation/te.htm

Trial and error methodology. http://en.wikipedia.org/wiki/Trial_and_error

Universal gas constant. http://www.grc.nasa.gov/WWW/K-12/airplane/eqstat.html

UTJ. http://en.wikipedia.org/wiki/Multijunction_solar_cell

WikiSat organization. http://code.google.com/p/moon-20/wiki/WikiSat_Engineering_Management_Plan

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      http://code.google.com/p/moon-20/

ANNEXES