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TÍTOL DEL TFC: Integració d'una càmera en el cicle de disseny d'un femto-satèl·lit i valoracio

TITULACIÓ: Enginyeria Tècnica Aeronàutica, especialitat Aeronavegació

AUTOR: Aina Pallarès Albaladejo

DIRECTOR: Joshua Tristancho Martínez

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Título: Integración de una cámara en el ciclo de diseño de un femto-satélite y valoración
Autor: Aina Pallarès Albaladejo
Director: Joshua Tristancho Martínez
Fecha: 31 de enero de 2014

Resumen

Un femto-satélite tiene una masa de entre 10 y 100 gramos. Los satélites comunes se basan en cajas de diferentes fabricantes. La consecuencia de esta arquitectura es un incremento de la masa debido a las estructuras, cableado y conectores. En este trabajo proponemos seguir lo conocido como "Thinking outside the box", un planteamiento diferente que se refiere a diseñar el satélite completo en una única placa PCB. Pero al final, la carga de pago es una caja; el cliente protege el sensor con una caja; en contraste, nosotros proponemos integrar la carga de pago en el ciclo de diseño del satélite. Por supuesto el cliente debe seguir una serie de reglas: la arquitectura de la carga se basa en tecnología Surface Mounted Device (SMD) y la creciente nanotecnología en forma de Micro-Electro-Mechanical Systems (MEMS).

Este trabajo de final de carrera (TFC) pondrá a prueba este concepto de "Integración de la carga en el ciclo de diseño de un femto-satélite". Una cámara del tipo SMD será usada en una misión de Gestión de Catástrofes (DM) como el impacto del tsunami en Sri Lanka en el 2004. El satélite tomará fotografías y las enviará directamente a los teléfonos móviles de los First Responders (FR) mientras se establece la ayuda. Al mismo tiempo, una estación de tierra programada en el femto-satélite antes del lanzamiento recibirá las imágenes con tal de valorar la calidad de las imágenes del sensor.

Se realizarán una serie de simulaciones como: simulación de la propagación de la trayectoria, estudio térmico, valoración de la radiación, valoración del riesgo de colisión con Satélites/Basura espacial/Asteroides, estudio de la cobertura del sensor, estudio de la gestión de la batería y valoración de los datos del sensor.

Palabras clave: Femto-satélite, MEMS, COTS, SMD, Ciclo de diseño

Title: Integration of a Camera in the Femtosatellite Design Cycle and Assessment
Author: Aina Pallarès Albaladejo
Director: Joshua Tristancho Martínez
Date: January, 31st 2014

Overview

A femtosatellite has a mass between 10 and 100 grams. Current satellites are based on boxes owned by different manufacturers. The consequence of this architecture is an increment of mass due to structures, wiring and connectors. In this work we propose to follow the well known "Thinking outside the box" where in our case it will mean to design the whole satellite in a single PCB board. Nevertheless, payloads are boxes; the client covers the payload sensor inside a box; in contrast we propose to integrate the payload in the satellite design cycle. Of course some design rules must be followed by the client: Payload architecture is based on the Surface Mounted Device (SMD) technology and the growing nanotechnology in form of Micro-Electro-Mechanical Systems (MEMS).

This final bachelor work (TFC) will put to the test this concept of "Payload integration in the femtosatellite design cycle". A SMD camera will be used in a Disaster Management (DM) mission like the Sri Lanka tsunami impact in 2004. The satellite will take pictures and send it back directly to the cell phone First Responders (FR) until the help is established. At the same time, a ground station programmed in the femtosatellite before launch will receive the pictures in order to assess the picture quality of the sensor.

A series of simulations and studies will be performed like: Trajectory propagation simulation, Thermal study, Radiation assessment, Satellites/Space Debris/Asteroids collision risk assessment, Sensor coverage schedule study, Battery power management study and On-board sensor data assessment.

Keywords: Femtosatellite, MEMS, COTS, SMD, Design cycle

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Als meus pares, per haver confiat sempre en mi i haver-me recordat lo orgullosos que estan quan més ho necessitava.

Al Marc, per estar sempre al meu costat i donar-me forces.

Al meu avi, per animar-me quan més falta em feia.

Al Joshua, que ha fet possible que aprengués i em divertís al mateix temps, gràcies per la teva dedicació i ajuda constant.

A l'equip WikiSat, gràcies per orientar-me i ajudar-me durant aquest projecte i fer que em sentís com una més de l'equip.

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ACRONYMS, ABBREVIATIONS AND DEFINITIONS

AESA	Agencia Estatal de Seguridad Aérea
CAD	Computer-aided design
CLK	Clock
COTS	Commercial off-the-shelf
DAS	Debris Assessment Software
DM	Disaster Management
ESA	European Space Agency
EMI	ElectroMagnetic Interferences
FOV	Field of view angle
I^2C	Interconnected Integrated Circuit (Two wire interface)
IC	Integrated Circuit
ICAO	International Civil Aviation Organization
IMU	Inertial Measurement Unit
ISS	International Space Station
LEO	Low Earth Orbit
LET	Linear Energy Transfer
MCU	Multipurpose Control Unit
MEMS	Micro-Electro Mechanical System
NASA	National Aeronautics and Space Administration
NEMS	Nano-Electro-Mechanical Systems
NOTAM	Notice To Airmen
ORSAT	Object Reentry Survival Analysis Tool
PA	Power Amplifier
PCB	Printed Circuit Board
SAA	South Atlantic Anomaly
SCL	Signal Clock of the I^2C bus
SDA	Signal Data of the I^2C bus
SMD	Surface-Mount Device
SMT	Surface-Mount Technology
SPI	Serial Peripheral Interface
SSA	Synthetic Aperture Antenna
SPENVIS	Space Environment Information System
TFC	Trabajo final de Carrera (Final Bachelor Work)
TID	Total Ionizing Dose
TMA	Terminal Maneuvering Area
UDP	User Datagram Protocol
UN	United Nations
USSTRATCOM	United States Strategic Command

INTRODUCTION

A femtosatellite is a small satellite having a mass between 10 and 100 grams. To achieve such a reduction of mass, and still be functional, new philosophies must be applied to the satellite design.

Current satellites are based on boxes owned by different manufacturers where certification responsibility is bounded in each box. The consequence of this approach is an increment of mass due to the structures, wiring and connectors.

One new philosophy proposed in this work is the so called "Thinking outside the box", a different approach where in our case means to design the whole satellite in a single PCB board (Satellite on-a-board) including the payload. Usually, payloads are boxes; the client covers the payload sensor inside a box. To simplify even more we have proposed a second philosophy that is **to integrate the payload in the femtosatellite design cycle if same design rules are followed by client and the satellite manufacturer.** Then, the payload architecture must be based on the Surface Mounted Technology (SMT) which is the same technology that the satellite is based. Obviously, not all the payloads can be manufactured in a SMD format, in example a telescope application or a high power communication application but many of them are available in the market in form of Commercial-off-the-shelf (COTS) thanks to the growing nanotechnology in form of Micro-Electro-Mechanical Systems (MEMS).

This final bachelor work (TFC) will put to the test this concept of "Payload integration in the femtosatellite design cycle" placing a SMD camera like the *TOSHIBA TCM8230* and the related components. The required size, mass and power consumption are in the scale of this femtosatellite. We emulate a client that want to test this camera in a real application like a Responsive Space mission for Disaster Management (DM) and we generate a report as the output for this client- To do the assessment we will run some simulations. The mission proposed for this test will be based in a real catastrophe: the *Sri Lanka* tsunami impact in 2004. Four hours after the impact, a femtosatellite will be injected into orbit through a *Rockoon* that in the operation campaign will be launched in the *Gran Canaria Spaceport* in *Spain*. For our simulations the balloon will be released over the land in *Zaragoza* in *Spain* to have the worst conditions of risk, air traffic, bad weather, etc.

The satellite will take pictures and send directly to the First Responders (FR) until the help is established by the United Nations few days later. Due to the disaster, basic infrastructures are down for few days, like electric supply, water supply, communications, etc. The satellite will send the pictures directly to the cell phone of the affected people through a wireless connection (In example an UDP protocol). At the same time, a ground station programmed in the femtosatellite before launch will receive the pictures in order to assess the picture quality of the sensor. Then, it is assessed the idea of integrate the payload in the engineering cycle instead of to assemble the satellite from boxes.

Objectives

- To integrate a typical MEMS payload in the engineering cycle of a femtosatellite in order to see if this approach is valid.
- Propose improvements to the femtosatellite platform when the payload is integrated in the engineering cycle if this improvement is a common need for SMD payloads.
- To study the *TOSHIBA TCM8230* sensor like an example of payload.
- To study a catastrophe adequate for a Responsive Space Mission.
- To use an Open Source tool to simulate a Responsive Space Mission in which is used the *TOSHIBA TCM8320*, taking into account the efficiency of the sensor and the risks.
- To analyze the risks that the satellite can suppose on the environment in the studied mission, simulating it in the *Moon2.0*.
- To study the conditions that the satellite and the sensor will be exposed to in the mission, and analyze if they can work well in this case.
- To assess the efficiency of the sensor in the proposed mission, the impact in the satellite and the suitability of the obtained data, proposing alternatives in the case of not satisfying results.

Student work plan

This bachelor work consists of a minimum of 360 hours done by the student. There was a weekly meeting with the tutor. I managed many tools like Microsoft Word as a word processor, and Microsoft Excel as the most used statistics tool. Also I learned about some NASA risk analysis tools like DAS, ORSAT, USSTRATCOM and SPENVIS and other *WikiSat* tools like *Moon2.0*, Wikimedia *moon-20* (*Moon2.0* web) and Wiring schematics in CAD format like *Eagle*® PCB.

The bachelor work (TFC) requires a number of 360 hours. Following, number of dedication hours is detailed:

- 7 h in learning about the automaton *Arduino*. I had to learn about how to program it, so the *WikiSat* group had provided me 4 h of formation.
- 10 h in knowing the payload, in this case the camera proposed to be integrated in the design cycle of the mission to test the concept. I had to read the datasheet and make some experimental test with the camera. I had known how it will be integrated in the satellite to check if it follows the rules of the engineering cycle. I had to learn the I2C protocol and use it through the *Arduino*.
- 20 h reading associated works and TFC of other students to know the studies about the LiPoly batteries in the empty, the effects of the radiation over the MEMS components (like the camera used in this work) depending on the orbit and the duration, and the thermal effects on the components in conditions of exposition to the Sun and the Eclipse.
- 5 h in knowing the operation of the lens that can be putted like payload, the specifications that can be useful for the mission and if some problems are expected when these components done for the domestic

market are exposed to the near space conditions (empty, high thermal changes, moderated radiation, etc).

- 5 h in knowing the satellite in the wiring diagram level (Wiring diagrams, Wiring schematics, etc), the parts of the satellite, the subsystems of it, and the area designated to the payload and its limitations.
- 5 h looking for an example of disaster mission and studying it to understand what are the requirements of the payload and what is expected from the satellite to reduce the effects of this catastrophe.
- 20 h learning about how the *WikiSat* group wants to put into orbit a femtosatellite, the details of the mission, the duration of an orbit, the type of orbit used, how is the satellite programmed to activate the camera, to do the broadcasting of the pictures or downloading the data and which utility can has the satellite when the group insert the payload.
- 15 h in learning to use the *Moon2.0* simulator, from which I have received 6 hours of formation from the *WikiSat* group.
- 4 h of formation about the *WikiSat* team and their facilities.
- 5 h in learning to use statistic tools of the Excel and some statistics functions used in this work.
- 2 h learning about analyzing tools like DAS, ORSAT, USSTRATCOM, SPENVIS.
- 15 h in doing an Excel file with the thermal control calculations done for the old and the new satellite design which incorporate the lens.
- 45 h in doing mission simulations with the *Moon2.0*. I have done simulations that lasted several days in order to capture realistic pictures of what the satellite may see. In this way, I could do an assessment of the results expected of the studied camera.
- 7 h in writing a tutorial which will be added to the original documentation of the *Moon2.0* program. The tutorial is about how to ask for a NOTAM and I have done it in English. To do this I had to work with the programmers and editors of this tool.
- 76 h in weekly meeting with the tutor.
- 15 h doing quality audits with the *WikiSat* group for assess if I follow the established program and be sure that I achieve the quality standards expected by the organization. Some tools have been used to control the activity and manage the security copies.
- 90 h redacting the memory. I have do near of 70 revisions of the document verified by my tutor.
- 20 h doing the TFC presentation and practicing the oral defense.

Total 372 h

Written work distribution

To achieve the objectives some simulations and studies will be performed like: Trajectory propagation simulation, Thermal study, Radiation assessment, Satellites/Space Debris/Asteroids collision risk assessment, Sensor coverage schedule study, Battery power management study, and On-board sensor data assessment.

The first chapter is an introduction to the *WikiSat* space program as well as the technologies that will be used in this work.

The second chapter contains an example of payload integration in the design cycle based on a SMD camera.

The third chapter consists of a real Disaster Management mission and a series of studies/assessments in order to test the SMD camera.

The chapter four has a risk assessment around few topics like: Airspace, Satellites, Space debris and Asteroids.

The chapter five is a satellite thermal study for the previous mission of Disaster Management. Improvements will be proposed for the future satellite version.

The chapter six contains an assessment of the satellite minimum valid inclination and the data quality downloaded to the ground station from the onboard sensor.

The chapter seven has the conclusions, the future work and the environmental impact study.

Finally, there are the bibliography and the annexes.

CHAPTER 1. SATELLITE DESCRIPTION AND RELATED TECHNOLOGIES

The first chapter is an introduction to the *WikiSat space program* for femtosatellite platforms and launched with *Rockoons*. In addition, the concept of payload integration in the femtosatellite design cycle is presented as well as the technologies that will be used through.

1.1 Definitions and related technologies

This section introduces some definitions related with technologies or radiation that are implicit in other chapters.

1.1.1 MEMS – Micro-Electro Mechanical Systems

The Micro-Electro-Mechanical Systems are the technology of very small devices; it merges at the nano-scale into nano-electromechanical systems (NEMS) and nanotechnology. MEMS are made up of components between 1 to 100 micrometers in size, and MEMS devices generally range in size from 20 micrometers to 1 millimeter. They (see Figure 1) usually consist of a central unit that processes data (the microprocessor) and several components that interact with the surroundings such as microsensors.[1]

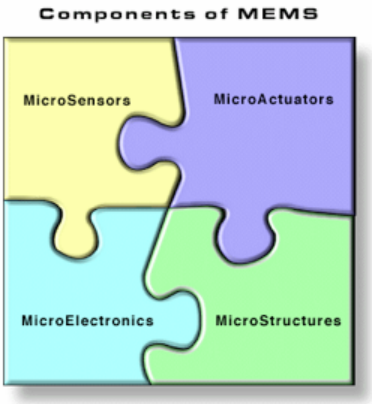


Figure 1 – Micro-Electro-Mechanical Systems components. Source: MNX[2]

The materials used for MEMS manufacturing are: silicon (it has good mechanical properties), polymers (cheaper than silicon because it can be produced in huge volumes), metals (they can exhibit very high degrees of reliability), and ceramics (advantageous combinations of material properties).

[1] http://en.wikipedia.org/wiki/Microelectromechanical_systems
[2] https://www.mems-exchange.org/MEMS/what-is.html

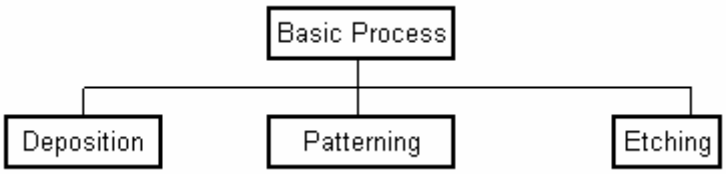


Figure 2 – MEMS basic processes

MEMS application is categorized by type of use: sensor, actuator and structure. The basic building blocks in MEMS processing are showed in the Figure 2:

- The *Deposition process* consists of the ability to deposit thin films of material with a thickness anywhere between a few nanometers to about 100 micrometer.
- The *Patterning* building block is the transfer of a pattern into a material.
- The *Etching processes* can be done by two methods: wet etching and dry etching. In the first one, the material is dissolved when immersed in a chemical solution. In the second one, the material is sputtered or dissolved using reactive ions or a vapor phase etching.

1.1.2 COTS – Commercial-off-the-shelf

Commercial off-the-shelf components (COTS) are usually components that are commercially available to the general public and which require no special modification or maintenance over its life cycle. COTS were used for software [1] but we are extending the concept to the hardware. They have a well defined architecture. For example MEMS or SMD devices are COTS.

COTS provide some advantages [2] like applications are provided at a reduced cost, the application is higher quality because competition improves the product quality, and the maintenance is easier because the systems documentation is provided with the application.[3]

1.1.3 SMD – Surface Mounted Device

This is an electronic device so is made following the Surface-Mount technology (SMT), which is a method for making electronic circuits in which the components are mounted or placed directly onto the surface of printed circuit boards (PCBs). In the industry it has largely replaced the through-hole technology construction method of fitting components with wire leads into holes in the circuit board.

An SMD is usually smaller than its through-hole counterpart because it has either smaller leads or no leads at all. It may have short pins or leads of various styles, flat contacts, a matrix of solder balls (BGAs), or terminations on the body of the component.[4]

This type of device have some advantages in front of the through-hole technique: they are smaller, lower initial cost, simpler and faster automated assembly, better mechanical performance under shake and vibration conditions,

[3] http://en.wikipedia.org/wiki/Commercial_off-the-shelf
[4] http://en.wikipedia.org/wiki/Surface-mount_technology

better EMC compatibility, components can be place on both sides of the circuit board.

Nevertheless, they have some disadvantages in front of the through-hole technology: they are more complex devices so they need a more complex manual, their reparations are more difficult and they are unsuitable for large, high-power, or high-voltage parts.

For these reasons, it is common to combine SMD and through-hole construction.

1.1.4 Radiation definitions

To understand better this section, it is important to explain some definitions:

- The *flux* is the amount of energy or number of particles (that can be photons) through a unit area per unit time. The flux is measured in number of particles/cm^2/s, units of the centimeter-gram-second system.
- The *fluence* is the number of particles (that can be photons) through a unit area. It is measured in number of particles/cm^2, units of the centimeter-gram-second system.
- The Linear Energy Transfer (LET) is the amount of energy incident over a material per area and mass dedicated to the ionization of this material. This parameter is measured in MeV/cm^2/g and usually its values are tabulated in different types of incident particles and materials.
- The Total Ionizing Dose (TID) is the integral of the radiation dose that a component receives during an interval of time. This consists in the integral, as function of the energy, of the product of the LET and the energy spectrum. The unit of this parameter is Rad, the absorbed radiation dose.

1.1.5 Ionizing particles

The most frequent particles that produce ionizing radiation are:

- *Alpha* (α) radiation consists of helium nuclei fast moving, which is formed by two protons and two neutrons. It can be stopped by a sheet of paper.
- *Beta* (β) radiation consists of electrons and can be detained by an aluminum plate.
- *Gamma* (ɣ) radiation is the highest frequency electromagnetic radiation. It consists of energetic photons. It is eventually absorbed following an exponential law as it penetrates a dense material.
- *Neutron* (n) radiation consists of free neutrons that are blocked using light elements (like hydrogen) which slow and/or capture them.

1.2 WikiSat space mission program

The *WikiSat* team was born in the 2009 like an open international group formed by people keen on the space but from different fields. Its main finality is to provide access to the space for everyone[5]. The *WikiSat* team participates in *N-Prize* competition, which consists of put into orbit a small satellite of less than 20 grams by private means and the price cannot be more than 1,200 Euros. Nowadays, no team has succeeded.

The *WikiSat Space Program* consists of a femtosatellite design, its launcher and the mission management platform, so that it can be adapted to the *N-Prize* competition rules but it can be reused. The design of the space access platform was thought under the *Space Payload Paradigm*[6]. The satellite controls autonomously the trajectory of the rocket, the orbit injection as well as the payload schedule and management. The design of the satellite is open and based on technologies accessible for the general public. This facilitates the payload physical integration in the satellite and also in the mission. It will allow the reuse of the design for many missions changing the payload and the sensor application. For many Earth observation missions almost all remains constant, only the payload changes. Several subsystems are the same different kind of missions.

Responsive Space mission for Disaster Management (DM) have the advantage that the satellite is launched when the catastrophe is known. This fact saves resources respect to other kind of missions while maintaining the satellites in the orbit waiting for a catastrophe. This method can revisit any point over the planet (Except for the poles) in less than 13 hours thanks to its circular very Low Earth Orbit with 40º of inclination.

1.3 Femtosatellite description

This section describes the main components of the old *WikiSat* femtosatellite version 4.1 and their consumption.

Table 1 includes a list of the subsystems of the satellite specifying their mass, consumption, operational temperature range and their size. These aspects are really important to design the satellite and to assure the correct operation of this. The total consumption has to be less than the total capacity of the battery and the total of temperature is the most restrictive of all the margins of the components; in this way we will be able to ensure the good operation of all the elements. For our payload, the mass budget is about 1 gram and the power budget is 150 mW.

[5] http://www.wikisat.org/?page_id=2
[6] http://code.google.com/p/moon-20/wiki/Space_Payload_Paradigm

Table 1 – WikiSat v4.1 mass, power and temperature budget (Source: Lara Navarro)

Subsystem	Mass	Max. consumption	Idle	Temp. [ºC]	Size [mm]
Power supply	6.6 g	Batt. 3.3 V 600 mAh	-	-30 to 60	D24.5x5
Communications	0.1 g	100 mW	0.1 mW	-40 to 85	171x34
Structure	2.0 g	-	-	-116 to 204	30x25x7
Attitude determination	1.4 g	5 mW	0.1 mW	-30 to 70	D5X2
Position determination	1.2 g	65 mW	0.1 mW	-30 to 60	30x16x3
Attitude control	0.4 g	40 mW	30 mW	-150 to 550	1x0.25x0.25
Tracking	7.0 g	500 mW	25 mW	-40 to 85	140x25x1
Camera	1.0 g	150 mW	-	-20 to 60	6x6x4.5
TOTAL	19.7 g	860 mW	55.3 mW	-20 to 60	171x34x7

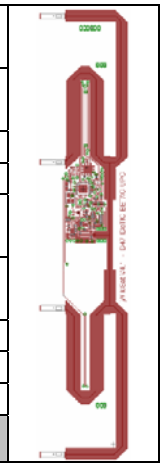


Figure 3 represents all the components of the satellite and after that all of these are explained.

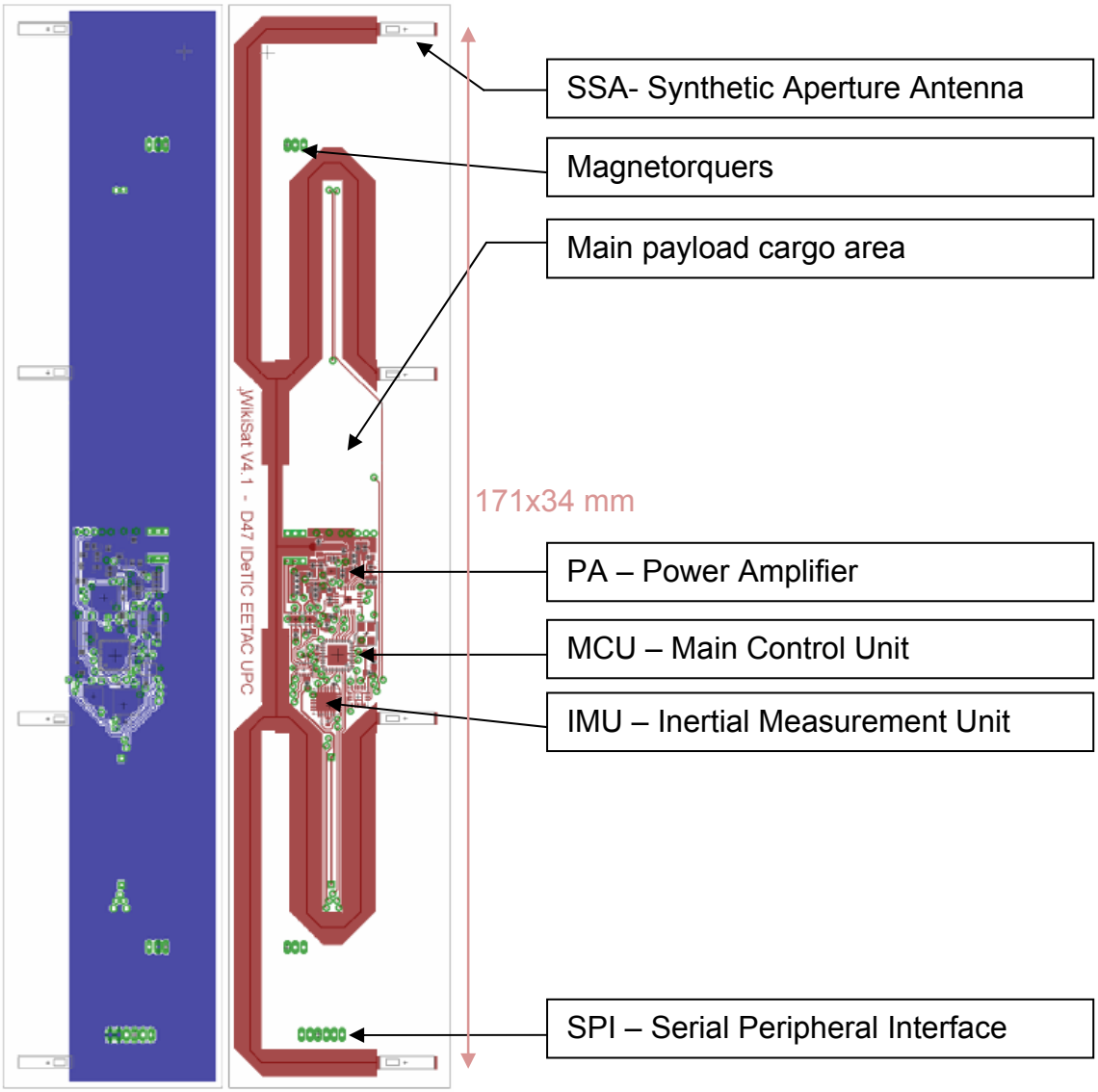


Figure 3 – Main components of the WikiSat femtosatellite (Adapted from: Lara Navarro)

1.3.1 SSA – Synthetic Aperture Antenna

In the *WikiSat* there are four ceramic antennas that are distributed like the Figure 3 shows. In this way, they form an antenna array with synthetic aperture and with the benefits of a high directive beam and a high gain as proposed **Fernandez-Murcia** in [18].

1.3.2 PA – Power Amplifier

This component is used to feed the micro-strip injection point through the circuit adapted in the *WikiSat*. The group formed by the Power Amplifier, the transceiver and the Main Control Unit is called System-on-Chip (SoC).

1.3.3 MCU – Main Control Unit

This area is shielded by a LiPoly battery that supplies 600 mAh. The Main Control Unit is based in an 8051 compatible processor, with 32 kb of RAM and 2 kb of SRAM and a frequency of 16 MHz.

1.3.4 IMU – Inertial Measurement Unit

This unit is compounded of 3 Axes accelerometers with high accuracy and 16 bits of data resolution, 3 gyros and 2,000 degrees per second range and a temperature sensor for the calibration in real time. The bus used for data and configuration is the I^2C.

1.3.5 SPI – Serial Peripheral Interface

This component is used for the communication with the launcher, because its main function is to control the launcher vector.

1.3.6 Magnetorquers

Two degrees of freedom for attitude control and for antenna pointing are provided by four coils. This configuration is based on the Earth's magnetic field. Depending on the orbit and the position some corrections should be done in real-time.

1.4 Main payload cargo area

This section introduces the payload areas and de design rules of the client.

1.4.1 Payload cargo areas

In the WikiSat there are three payload areas, like is shown in the Figure 4.

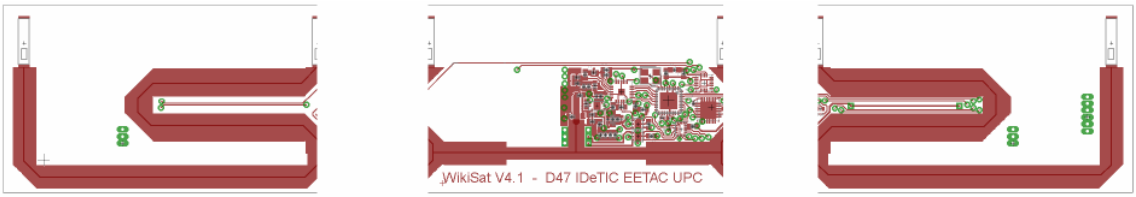


Figure 4 – WikiSat payload areas of the old version, the WikiSat v4.2

The central section has the most often used payload area. This section is divided in two parts, the onboard computer and the main payload area. Between both parts there is a control bus that connects the payload with the onboard computer. Lateral payload areas could be used only if necessary when two payload components are mandatory to be splitted, i.e. to test optic links or similar applications. In these three areas can be adapted surface-mounted devices (SMD) components.

The central area has the following benefits in front of the lateral ones:

- The main payload area is prepared for surface mounted devices without physical connectors or holes. These SMD components are integrated in the satellite wiring design. It has the largest payload area.
- The central area is shielded by the battery that is placed backwards.
- Through an I^2C bus the state of the component situated in this payload area can be monitored by the satellite while lateral areas have not this bus.
- The electromagnetic compatibility is well known in this area.

1.4.2 Design rules

There are some design rules that the client must follow because the satellite is based in the architecture of a single PCB board. **Navarro** has presented these rules in [5]. The *WikiSat* team developed the satellite schematics in a CAD design tool called *Eagle PCB*. Dimensions were extracted from the *WikiSat* file.

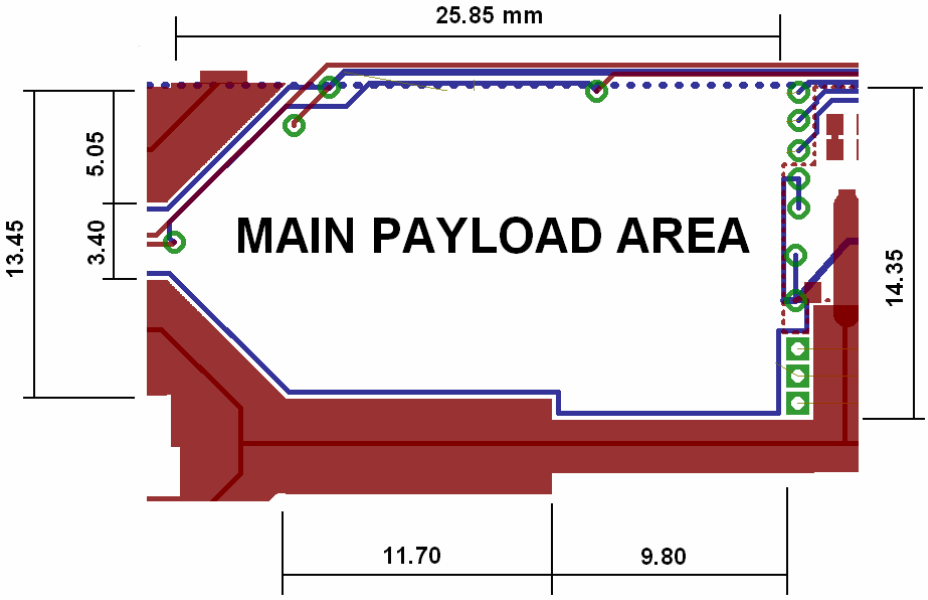


Figure 5 – Main payload area dimensions in mm

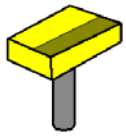
Payload SMD components must be placed inside the available area. The dimensions are detailed in Figure 5. The most important rule is that the components have to be SMD with a size of 0402 (1.00 x 0.50 mm). The separation between them has to be of “8mil”. Another important thing is to analyze the Electro-Magnetic Compatibility (EMC) between all the components. In this work, the payload is mainly a SMD camera. The integration of this camera followed these rules.

1.5 New satellite version budgets

Through the work we will see that we have to modify the old satellite version. In the Table 2 we present the new mass, power and temperature budgets that will be justified in the following chapters. Comparing this budget with the one of the previous version satellite, we can observe that the maximum consumption decreases a lot and the size of the satellite is reduced too.

Table 2 – New satellite mass, power and temperature budget. WikiSat v6

Subsystem	Mass	Max. consumption	Idle	Temp. [ºC]	Size [mm]
Power supply	6.6 g	Batt. 3.3 V 1,000 mAh	-	-30 to 60	60x37x5
Communications (Radiometrix option)	0.1 g	92 mW	0.1 mW	-40 to 85	32x12.5x3.8
Structure	2.0 g	-	-	-116 to 204	30x25x7
Attitude determination	1.4 g	5 mW	0.1 mW	-30 to 70	D5X2
Position determination	1.2 g	63 mW	0.1 mW	-30 to 60	30x16x3
Attitude control	0.4 g	40 mW	30 mW	-150 to 550	1x0.25x0.25
Tracking	7.0 g	0 mW	25 mW	-40 to 85	140x25x1
Camera	1.0 g	150 mW	-	-20 to 60	6x6x4.5
TOTAL	19.7 g	350 mW	55.3 mW	-20 to 60	60x40x75



The new Radiometrix radio will has less bandwidth (it will not send videos but it will be able to download high definition photos). The frequency is decreased to have more range. The antenna is changed too, instead of a directional antenna it will be used a dipole, which is omnidirectional. The battery is changed to one with more margin of operation where the effective capacity is 60% of the total. The payload includes the lens (telephoto lens) which affect significantly to the thermal balance. The thermal control system is adjusted to the new requirements of the satellite.

CHAPTER 2. EXAMPLE OF PAYLOAD INTEGRATION

The second chapter presents an example of payload integration in the design cycle. The client will be a manufacturer of MEMS that want to test a SMD camera. They want to test this camera in the space environment for Space Responsive applications.

2.1 Payload description

The prepared payload is an optic sensor, the *TCM8230*. This camera was used in mobile phones, micro cameras or toys. Table 3 includes the general specifications of the *TOSHIBA TCM8230* camera obtained from the datasheet[7]. The module size is important for the integration of the camera in the satellite. The operational temperature is really important because in the space the conditions are very different than in domestic situations, so we have to assure that the camera will work well in the space.

Table 3 – Camera and lens specifications (Source: *TOSHIBA TCM8230* datasheet)

Camera specifications			
Power supply	Photo diode I/O	2.5 V +/-0.2 V	
	A/D converter Digital	1.5 V +/-0.1 V	
Temperature ranges	Storage	-30 to 85 ºC	
	Operational	-20 to 60 ºC	
Module size	6x6x4.5 mm (WxDxH)		
Camera mass	0.18 grams		
Lens specifications			
Optical format	1/6 inch		
Total pixel numbers	698(H)x502(V)		
Field of view	Horizontal	57.4 degree	
	Vertical	44.5 degree	
	Diagonal	69.1 degree	
Map resolution h=250 km	Pixel size	344 meters	
	Area	240x173 km	
Structure	Double lens		
Optional telephoto lens specifications			
Reference	MC5970P-C		
Field of view	8.0 degrees		
Map resolution h=250 km	Pixel size	50 meters	
	Area	35x25 km	
Cylinder size	Diameter 21.4x70 mm		
Cylinder mass	0.82 grams		

[7] Datasheet http://kreature.org/ee/avr/cmos_cam/TCM8230MD.pdf

The lens specifications are important to assure the enough quality of the pictures obtained during the mission; the large distance and the different inclinations that will have the satellite during the mission makes more difficult the obtaining of good photos.

Figure 6 shows the size of the camera and the schematics. The left photo shows how small the camera is. The right schema details the dimensions of the camera, which is really important to know if this camera can be used in the WikiSat satellite, because there is a fixed area designated to the payload. Finally, the third schema (bottom) is about the schematics of the camera (left) and the serializer (right), necessary to know how to do the electric connections.

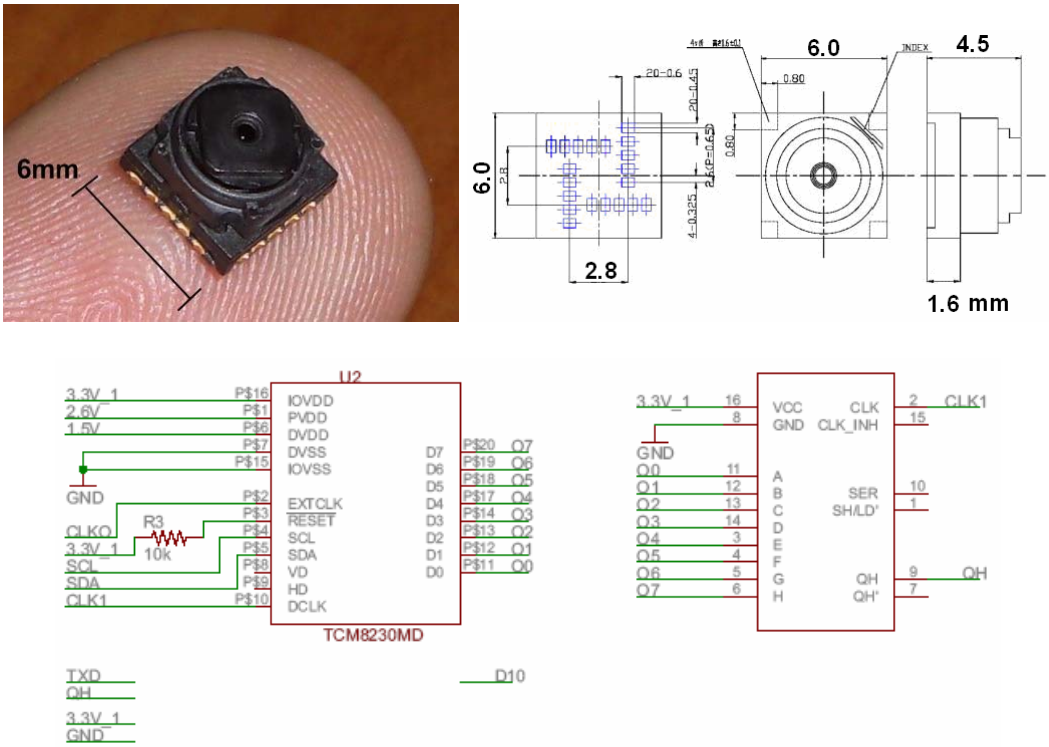


Figure 6 – Camera dimensions and schematics

2.2 Payload integration

The satellite will be used in a Responsive Space Mission, which has the main objective of observing the Earth and taking photos for the first week of a catastrophe; in the next chapter the mission will be described. For this mission, the payload (in this case the camera) has to accomplish the following requirements:

- SMD encapsulation
- MENS based design
- Low mass
- Low consumption
- Impact resistance
- Resolution at least 640x480x8 bits of color
- Pixel size smaller than 500 meters at 250 km distance

When the camera is integrated in the satellite, that has a given orbit, it is possible to define the map resolution of the pictures. The minimum pixel size of the *TOSHIBA TCM8230* camera is 206 meters [11] in the original lens configuration of 57.4º and 50 meters of pixel resolution in the 8.0º lens configuration. The closer catastrophe distance in the orbit of the next chapter example is 250 km. The best pixel size at this altitude for each lens will be:

$$s_{pixel57^{\circ}} = \frac{250\,km \cdot \sin\left(\frac{57.4^{\circ}}{2}\right)}{\frac{698}{2}\,pixels} = 344\,meters \qquad s_{pixel8^{\circ}} = \frac{250\,km \cdot \sin\left(\frac{8.0^{\circ}}{2}\right)}{\frac{698}{2}\,pixels} = 50\,meters$$

These pixel sizes in the map correspond to an area of 240x173 km for the 57.4º lens and an area of 35x25 km for the optional 8º lens when inclination is 90º also called orthophoto. Annexes A.3 provide extra details for different distances or satellite inclinations.

The payload mass budget is 1 gram but the real mass is 0.18 grams and the optional telephoto lens has a mass of 0.82 grams. The power budget is 150 mW while the power consumption of this camera is 132 mW.

In order to integrate the payload in the engineering cycle of the satellite, the Main Payload Area from the *WikiSat* is selected as showed in Figure 7. This area provides power and data communications.

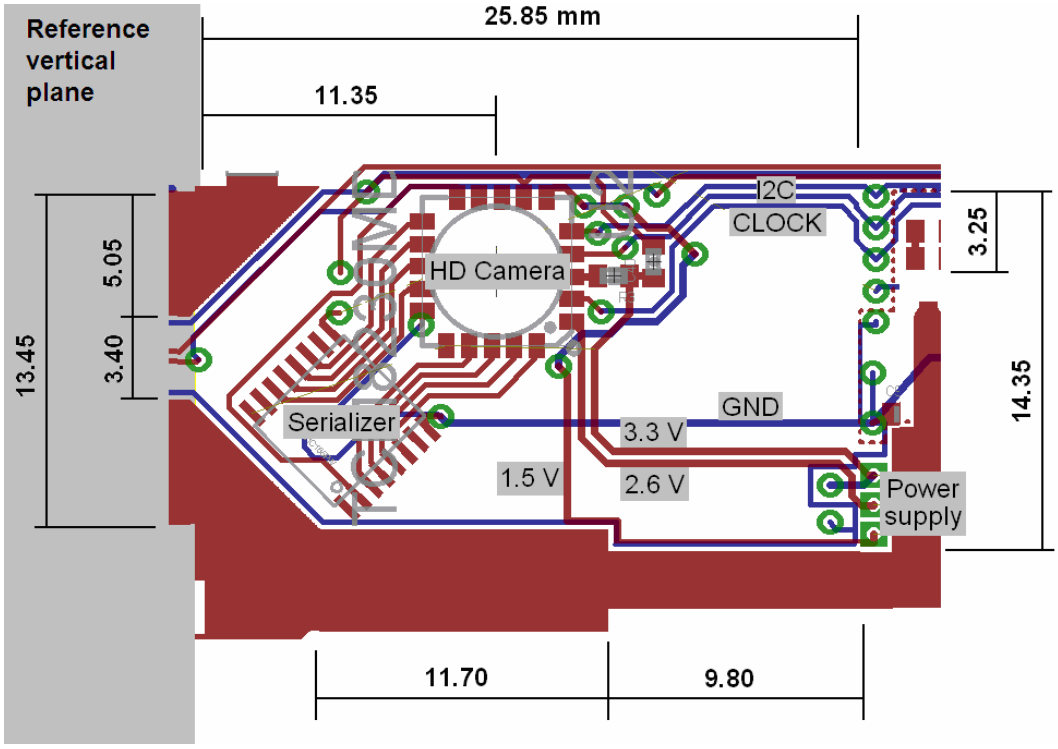


Figure 7 –Camera integrated in the Main payload area. Dimensions and location

During the satellite manufacturing process, our camera and the serializer are installed with the rest of satellite components. SMD components are placed with a soldering paste then the satellite goes to a reflow oven and heated up to 230 ºC for few seconds. Components are soldered and cooled to room temperature. The *TOSHIBA TCM8230* accomplishes with the mission requirements. The assessment of the picture quality is pending and will be covered in Chapter 6: Onboard assessment.

The camera is placed at 11.35 mm from the vertical reference side and 3.25 mm from the horizontal reference side of the Main Payload Area. The radiolink transmits or the memory stores the picture in a serial way. Since the colors are codified in 8 bits, a serializer is required as design **Navarro** in [15]. The Texas Instruments *SN74HC165* is a standard 8 bits serializer as depicted in Figure 7 and is feet by 3.3 volts. The output signal is called *QH*. The serializer input signals are 8 (*O0* to *O7*) that are connected with the camera outputs (*D0* to *D7*). The camera control is done by the I^2C bus (Two wires: *SCL* and *SDA*) which is connected with the onboard computer, the *Arduino*. The camera and the serializer have two clock signals (*CLK0* and *CLK1*). The camera is feet by three sources: 3.3, 2.6 and 1.5 volts.

2.3 Functional tests

Navarro performed functional tests of the camera as explained it in the page 41 of the Chapter 6 of her work[15]. These tests consist in confirm the good operation of the camera alone and then the good operation of the group formed by the serializer and the camera, because it is important to assure the good functioning of these two elements together. First the validation of each device was done and after that, the entire subsystem was validated by the same procedure.

The procedure first of all consists in sending a start-up sequence through the bus I^2C to the camera to activate it. To do this it is necessary to define a library containing the main registers of the camera and then send the required start-up sequence by a code. To test the camera output, it was used a Logic Analyzer.

The *TOSHIBA TCM8230* camera can be provided by *sparkfun* who also has a wide forum about this camera topic available in the web. Other authors evaluated and/or tested this camera. Some web references are:

- https://www.sparkfun.com/products/8667
- http://sigalrm.blogspot.com.es/2011/03/tcm8230md-breakout.html
- http://diydrones.com/m/blogpost?id=705844%3ABlogPost%3A1045800&maxDate=2012-12-02T17%3A10%3A04.693Z
- http://tim.cexx.org/projects/tcm8230/

CHAPTER 3. A DISASTER MANEGEMENT CASE

This chapter is an example of a Disaster Management mission in a real case. The payload component will be integrated in a satellite in order to help in this disaster.

3.1 The 2004 tsunami

The 2004 tsunami was one of the deadliest natural disasters on record. Because it occurred in the Christmas season and hit many resort area beaches. Its death toll of almost 250,000 was indiscriminate, taking not only South Asians but many visiting vacationers. People everywhere were affected by it.

The tsunami was caused by tectonic activity beneath the *Indian Ocean*. A fault twenty miles below the ocean surface ruptured, forcing one of the plates to be thrust upwards by as much as 40 feet. The ocean above was forced upwards and the displaced water moved out as a series of giant ripples. From the land, the first sign of a tsunami is the water being dragged out to sea. The vertical wall of the tsunami destroyed everything in its path. The undersea mega-thrust earthquake occurred at 00:58:53 UTC on Sunday, 26 December 2004, with an epicenter off the west coast of Sumatra, Indonesia and a magnitude of 9.3 on the *Richter* scale. The coordinates of the epicenter are: latitude 3° 19' N and longitude 96° E.

The size of a seaquake depends above all on the extent of the fault on which it occurs and the vertical shift of the sea bed. The tsunami took different amounts of time to reach different countries. The speed increases in relation to the depth of the sea, so in deeper waters the wave traveled more quickly.

Figure 8 – Tsunami of 2004

After 15-20 minutes the tsunami had already hit the island of *Sumatra*, after one hour and a half it hit *Thailand* and after about two hours it had reached the coasts of *India* and *Sri Lanka*, causing a total of around 290,000 deaths.[8]

Figure 8 and Figure 9 show examples of the devastation caused by the tsunami and a representation of the most affected area of *Sri Lanka*.

[8] http://www.scienzagiovane.unibo.it/English/tsunami/5-tsunami-2004.html

Figure 9 – Tsunami consequences

The earthquake and resulting tsunami affected many countries in Southeast Asia and beyond, including *Indonesia, Sri Lanka, India, Thailand, the Maldives, Somalia, Myanmar, Malaysia, Seychelles* and others. Many other countries, especially *Australia* and those in *Europe*, had large numbers of citizens traveling in the region on holiday. *Sweden* lost 543 citizens in the disaster, while *Germany* had 539 identified victims.

3.2 Example of Disaster Management mission

The method for femtosatellite operations based on a list of coordinates as proposes **Tristancho** in [3] simplifies the operation almost in any disaster management mission if each femtosatellite has a minimum number of subsystems. Then, a mission is easy to program using a few default functionalities in everyone of a cooperative satellite swarm giving a list of coordinates. The satellite will send real-time pictures to the first responder agents until the help arrives.

The satellites used in Disaster Management (DM) have the disadvantage that they need a continuous maintenance during the time in which nothing happens. Because of this femtosatellites can be a way of decreasing the operating costs in Disaster Management, because they are launched to a single emergency and the first hours after the disaster are essential to minimize the consequences.

The femtosatellite has a quite limited energy and will only be active while taking images of the disaster and downloading them. The rest of time, the femtosatellite will stay in and idle state.

Each femtosatellite uses a high power wireless modem so a ground station does not require a huge installation. The information to be transmitted is not encrypted because it is of global interest.

The basic process of a femtosatellite is to know where the satellite is relative to a coordinate and when the interest point is in sight it could perform a tracking maneuver, activate the sensors and acquire information. This information can be downloaded immediately to first responders, and/or stored to be subsequently downloaded to the scheduled ground stations.

Each femtosatellite will have a list of eight geographical coordinates; the first coordinate will be the point of interest that has absolute priority over other coordinates. The rest of coordinates are ground stations to download the information. The number of ground stations and the places should be considered. The higher number of ground stations increment the volume of information downloaded, but also increment the power consumption and decrease the mission time. Ground stations should be located after the disaster in the line of flight of the swarm (Usually Eastward) separated at least 1,000 km from the disaster area to have the information available as soon as possible to the decision makers. More than one ground station placed inside the 50 km footprint can receive the same information.

Changes on the objectives during the mission can affect the energy use for a given femtosatellite. These changes, usually done by the decision makers during the emergency are unpredictable.

3.3 Launch trajectory and orbit propagation

In this section some calculations are presented: The inertial matrix of the *WikiLauncher*, its trajectory and the *WikiSat* orbit propagation.

3.3.1 WikiLauncher inertial matrix estimation

The *WikiLauncher* is still in development but it is mandatory to fix the main specifications as real as possible. However, this is a possible estimation but future computations must be done. The *WikiLauncher*, as calculated by **Beizaee** in [6], has a total mass of 3,370 grams and it is composed by the payload section, the vector control section and a light stage1 with a 4 mm nozzle. The inertial matrix is showed in (3.1) marked as I_1 where the center of gravity is located at $I_{CG1} = \begin{pmatrix} 0 & 297 & 0 \end{pmatrix} mm$ from the rocket nozzle throat.

$$I_1 = \begin{vmatrix} 0.659 & 0.071 & 0 \\ 0.071 & 0.009 & 0 \\ 0 & 0 & 0.666 \end{vmatrix} \tag{3.1}$$

Trajectory simulation was programmed with these parameters in order to simulate the vector control subsystem using the so called hardware-in-the-loop technique as presented by **Navarro** in [5].

3.3.2 Trajectory computation

Main parameters are based on the simulation run by **Tristancho** in [3] launched from the *Gran Canaria Spaceport*. For our calculations, the launching point will be *Zaragoza* to provoke a worst case surrounded by ground instead of sea. The trajectory has an apogee of 250 km as showed in Figure 10. The first stage, the vector control and the fairing will reenter as predicted by the thin pink line trajectory. These particles will be burned during the reentry. The second stage and the satellite will achieve the orbital speed instead, while the second rocket

stages will follow the satellite. This phase of the mission is covered by the following section 3.3.3 Orbit propagation.

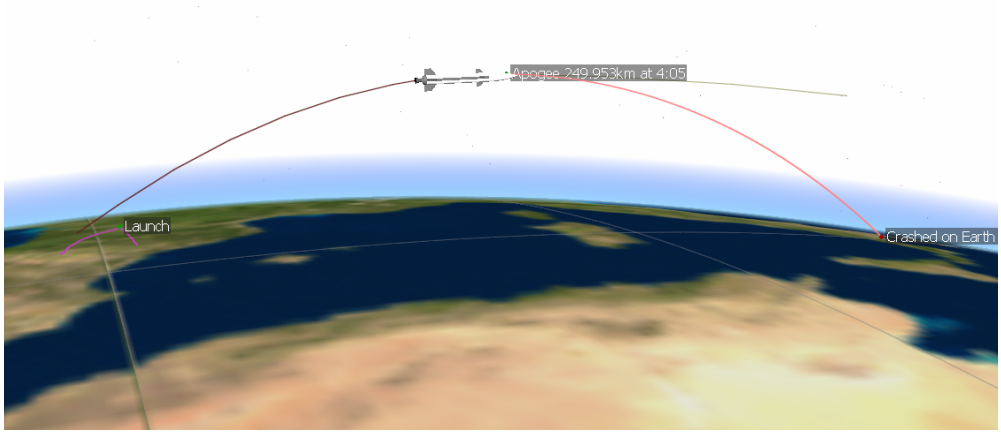


Figure 10 –WikiLauncher trajectory

3.3.3 Orbit propagation

The *WikiSat* orbit propagation was presented by **Tristancho** in [4] where only few ground stations receive the information. Figure 11 shows how only two footprints are available due to the simplified list of coordinates.

- The epicenter in *Matara* (*Sri Lanka*) and
- The *Maspalomas* ground station in (*Spain*).

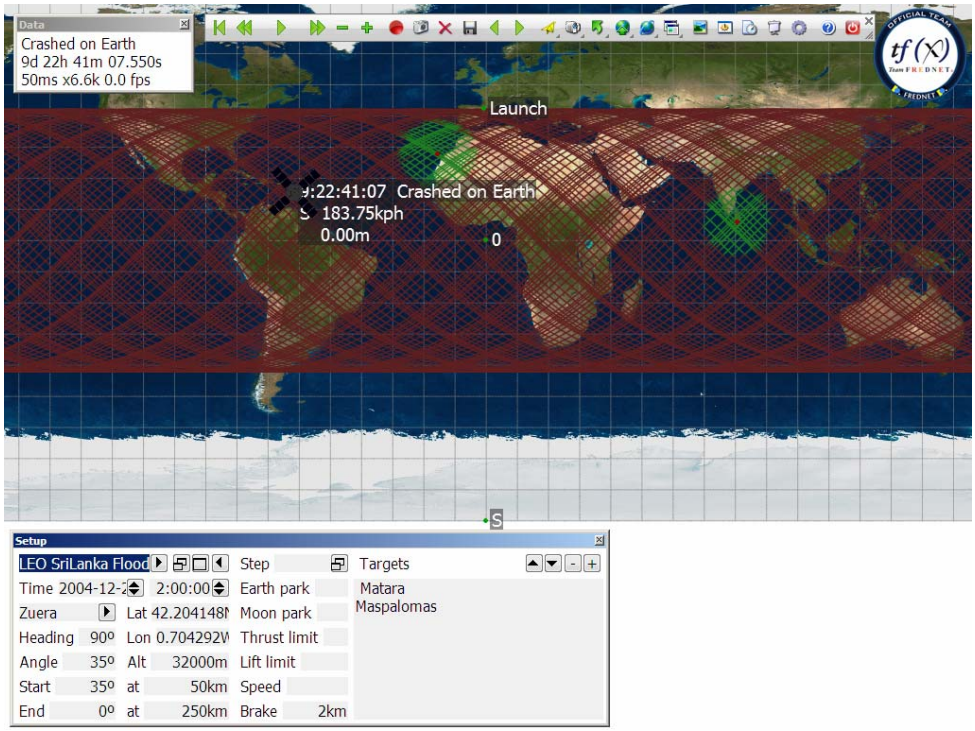


Figure 11 –*WikiSat* orbit propagation and ground coverage

The phases of the trajectory of the *WikiSat* are the following:

1. Ascension of the balloon
2. Ignition or liftoff
3. Apogee maneuver
4. Satellite injection
5. Orbit with Sun and Eclipse
6. Reentry

3.4 Sensor coverage schedule study

Our mission lasts 9 days and the satellite takes 24 hours to revisit the zone of interest, so the number of times that the satellite will fly over the affected area is:

$$visits = \frac{mission\ time}{revisiting\ \ period} = \frac{9\ days}{1\ day} = 9\ visits \tag{3.2}$$

In these 9 visits, many of them will be invalid because of the lack of light; this is due to the pass of the satellite over the affected area during the night. Another parameter that can suppose the invalidation of a photo is the incorrect inclination; the best photo will be the orthonormal one, the others will have some inclination in the highest point and the magnitude of this inclination can suppose the acceptance or not of a photo.

Another parameter to analyze is the pixel size that is very important in the quality of a photo. If the pixel size is too large, the photo will be poor and will not show clearly the site photographed. A pixel size bigger than 500 meters is not considered acceptable as proposed by **Izquierdo** [11]. The minimum pixel size of the *TOSHIBA* TCM8230 camera is about 200 meters.

3.5 Battery power management study

To study the management of battery power it is necessary to analyze the consumption of all the components of the satellite. Table 4 is a table that contains all the elements of the satellite and their consumption:

Table 4 – Mass, Power and Temperature budget (Source: Lara Navarro)

Subsystem	Device	Mode	Current consumption	Current consumption (mA)	Total mission consumption (mAh)
Microcontroller interfacing	Atmega168	Active	200 [uA]	0,20000	1,27500
		Power-down	0.1 [uA]	0,00010	0,02160
		Power-save	0.75 [uA]	0,00075	
Communication	nRF24L01P	Power-down	900 [nA]	0,00090	0,19440
		Standby - I	26 [uA]	0,02600	
		Standby - II	320 [uA]	0,32000	
		RX	13.5 [mA]	13,50000	25,81875
		TX	11.3 [mA]	11,30000	50,42625
		Crystal Startup	400 [uA]	0,40000	
	PA2423L		80 [mA]	80,00000	357,00000
Sensor	LIS331HH	Normal	250 [uA]	0,25000	0,01667
		Low-power	10 [uA]	0,01000	
		Power-down	1 [uA]	0,00100	0,21600
	ITG-3200	Normal	6.5 [mA]	6,50000	41,43750
		Sleep	5 [uA]	0,00500	1,08000
Power management	TCA6408	Operating	6.5 [uA]	0,00650	1,40400
		Stand by	1[uA]	0,00100	
	TPS192615	Active	0.5 [mA]	0,50000	3,18750
	TPS193333	Active	0.5 [mA]	0,50000	3,18750
Payload	TCM8230MD	VGA (15fps)	40 [mA]	40,00000	25,50000
	SN74HC165	Active	80 [uA]	0,08000	0,51000
	Magnetorquer	Active	100 [mA]	200,00000	12,75000
			Total mission consumption		524,02517
			Average consumption (mAh)		2,42604

The consumption of the satellite during the download of the photos is calculated using the contact duration average with a minimum inclination of 45º (52 seconds) and the consumption of the radio-link system.

The original radio-link system is called *nRF24L01P* and its consumption is 92 mA. The maximum range of this system is about 500 km, which is enough for our mission because the maximum distance between the satellite and *Matara* is about 350 km, but the margin is small and a rain condition can cause some problems. This system has the problem of the small margin of the duration of the battery too, which is about 6.6 hours. However, it exist another radio-link proposal called *NTX2* as presented **Izquierdo** in [12] with a consumption of 18 mA, a maximum range of 1,800 km and a battery life of 33,9 hours in this way we can make more contacts, i.e. we can add new ground stations in the list of interest points. So using the original communication system, the average consumption is:

$$Average\ consumption\ picture = 92\ mA * 52\ s = 1.3\ mAh \tag{3.3}$$

With the *NTX2* we can use a COIN battery which is lighter as informed **Izquierdo** in [12] and it allows a battery drain up to 50 mA. Of course, this is enough for the radio-link subsystem but not enough to feed the other subsystems that have a total drain of at least 330 mW which is 100 mA. If the satellite is composed by a single camera and this transceiver broadcasting continuously, the total consumption can be adapted to this power budget and we can use a different approach, i.e. a swarm of these satellites without attitude control. So using the *NTX2* system, the average consumption is:

$$Average\ consumption\ picture = 18\ mA * 52\ s = 0.26\ mAh \tag{3.4}$$

The number of pictures that the satellite will be able to take and download depends on the consumption of the camera during this process and the battery capacity. The next formula shows the calculation of the maximum number of pictures that the satellite can download with the old battery:

$$Max\ num.\ pictures = \frac{Effective\ battery\ capacity}{Average\ consumption\ picture} = \frac{600\ mAh}{1.3\ mAh} = 461\ pictures \quad (3.5)$$

It is not possible to use all the battery capacity when the satellite is cold. In the calculations, a 40% of margin not used that allows an effective battery capacity of 600 mAh instead of 1,000 mAh that the new LiPoly battery provides. This is another bad design from the old satellite version that is fixed in the new version.

3.6 Radiation assessment

The radiation assessment presents the main radiations sources: Cosmic ray and Trapped radiation. The effect of the radiation over the satellite is considered taking into account the layers and materials of the satellite.

In this study, it is used the distinction between two types of radiation: the cosmic ray radiation (X ray and gamma) and the trapped radiation.

3.6.1 Cosmic ray: X and Gamma

The Cosmic ray sources correspond to the electromagnetic radiation throw out in determinate radioactive nucleus decay, which is produced during some of the most energetic phenomenon of the universe. Because of the high energies that are characteristics of these rays, this is a type of ionizing radiation able to penetrate in the materials deeper than the alpha radiation (helium nucleus) or beta (electrons or positrons) that belongs to the trapped radiation.

The materials that can absorb better these rays are the ones that have a high electronic number and an elevate density. Also, when the gamma rays pass through the materials, the probability of absorption in a layer is proportional to the thickness of the layer. This supposes an exponential decrease of the intensity with the thickness. The exponential adsorption only occurs in a thin beam of radiation, if a thick beam of gamma radiation passes through a thick bloc of concrete, the dispersion by the faces reduces the absorption. When the physical matter is crossed, the gamma rays can ionize it, so they can pull out particles charged by the atoms.

3.6.2 Trapped radiation

The solar wind is a continuous radiation of particles from de Sun, which presents variations depending on its state of activity. It is emitted in all directions in different intensities. The particles that forms it are fundamentally protons, electrons, helium nucleus and, in less quantity, ions of other heavier elements like C, O, N, etc. Protons can produce secondary radiation when collide with first shield layers; electronic components are protected against this

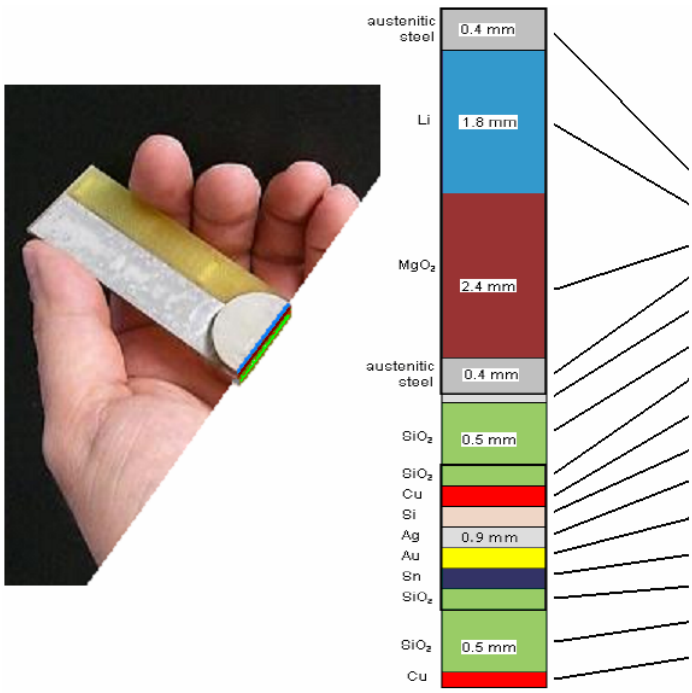
type of radiation due to its rapid scission inside the shield layers. In the Earth orbit level, the velocity is about 400 km/s but it has a lot of variations.

It exist a magnetic field around the Earth, which is more intense near the planet than the solar magnetic field. The particles of solar wind are forced to move following the lines of this field. The belt of radiation is formed by protons and electrons with high energy trapped in the field, due to the magnetic effect in which the force lines are joined in the proximities of the magnetic poles. There are two doughnut-shaped magnetic rings surrounding the Earth, which are called the *Van Allen* radiation belts. The particles trapped in the *Van Allen Belts* follow three different types of movement: a rapid rotation around the magnetic field lines defining a spiral movement, a slow rocking forward and backwards and a slow drift around the magnetic axis of the Earth.

The flux of trapped particles increases with the altitude, for this reason in the LEO orbit the radiation is less harmful to the electronic components than in the higher orbits. In the LEO orbit, the radiation only affects during the pass through the South Atlantic Anomaly (SAA) zone.

3.6.3 Effects of radiation over the femtosatellite

The WikiSat satellite radiation environment was studied by **Molas** and **Bermejo** [13, 14]. Figure 12 is a graphic explanation of the different layers of our satellite. Depending on the material and the thickness of a layer, the particles will penetrate easier or not. From layer 1 to 4 correspond to the satellite battery and the rest of them correspond to the electronic circuits, which are more fragile and are more important to be protected. Layers 5 and 6 are a thermal shield that protects the electronic circuits. From layer 7 to 13 is the Integrated Circuit like a MEMS. From layer 14 to 15 is the satellite board.



Layer	Material	Thickness (mm)
1	Austenitic steel	0.4
2	Li	1.8
3	MgO_2	2.4
4	Austenitic steel	0.4
5	Al	0.02
6	SiO_2	0.5
7	SiO_2	0.13
8	Cu	0.13
9	Si	0.13
10	Ag	0.13
11	Au	0.13
12	Sn	0.12
13	SiO_2	0.13
14	SiO_2	0.5
15	Cu	0.1

Figure 12 –Button battery system. Satellite layer definition (Source Esther Bermejo)

As we had explained, the first layers correspond to the battery and these change depending on the type of battery. Radiation is coming from the battery side because the satellite points toward the Earth. This COIN battery (Which is a CR2450 as depicted in Figure 12) protect better than the LiPoly one against the radiation. Therefore, using the *NTX2* radio-link system, the satellite will be more protected against the radiation because it allows use the COIN battery, whereas the *nRF24L01P* system does not allow it. But using all the subsystems, a COIN battery is not enough. The LiPoly used in our satellite protects less than the COIN battery, but the damage can be assumed. Those problems generated in the circuits are protected by redundant means like latching mechanism or software integrity monitoring.

Figure 13 shows two simulations obtained with the SPENVIS, a simulator of the propagation of the particles; both with 100 incident electrons and simulating an orbit of 250 km of altitude.

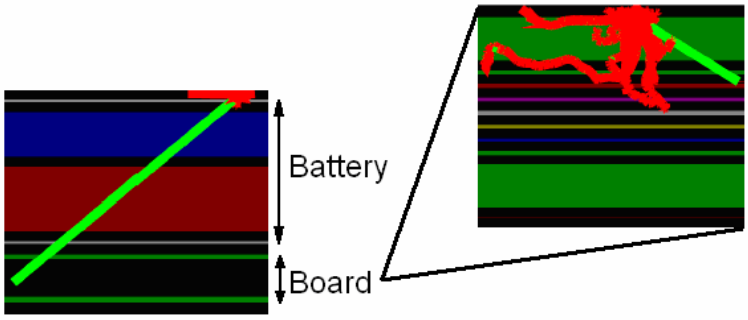


Figure 13 –Button battery system. 100 electron as incident particles. 250 km altitude (Source Esther Bermejo)

The left photo is the simulation obtained using a system composed by a battery and the electronic circuits, so the electronic devices are more protected against the radiation. In this case, we can observe that the more intense radiation (red traces) only affect the two first layers, and then some low radiation (green traces) penetrate to the rest of battery and electronic components. However the right photo is the simulation of particle propagation with 100 incident electrons obtained with a system without battery. In this case, the intense radiation, represented in red, penetrate in a lot of layers of the electronic circuits. In conclusion, the battery gets its additional objective of protection, the only radiation that penetrates inside the electronic components are the gamma particles. These gamma fluxes cause the Total Ionizing Dose[9] (TID) in the layers. All the electronic elements without the protection of the battery are exposed to higher radiation doses and can have more problems of ionizing. But both cases are possible because in low orbits the ionizing dose will not cause important damages to the devices. Both radiation and time of exposure are small.

[9] TID http://en.wikipedia.org/wiki/Total_Ionizing_Dose

CHAPTER 4. RISK ASSESSMENT

This chapter is a risk assessment around few topics like: Airspace, Satellite, Space debris and Asteroids involved in the mission. In order to guarantee the aircrafts safety inside the Airspace where the launch will take place, a simulation in the *Moon2.0* is performed. Our satellite can not disturb others satellites, because of this we check possible conflicts with near satellites in the simulator. Finally, we have to assure the integrity of our satellite, due to this we have to test the risk of collision with Space debris or Asteroids.

4.1 Introduction to the risk assessment

To analyze the different collision risks of our satellite, we use a similar method to the one used by the NASA and the ESA. Therefore it is necessary to explain these two methods.

Most space debris detection processes are iterative and involve several risk criteria [7]. The risk criteria depend on the quality of positional data available. The ISS debris avoidance process is initialized by the *USSTRATCOM* screening on the entire catalog over a 72 hours window. Any conjunction within a ±10 x ±40 x ±40 km box is notified to NASA. The box dimensions are radial x downtrack x crosstrack. If after additional orbit numerical processing, the event falls into a ±2 x ±25 x ±25 km box, collision probability is assessed. If the conjunction falls within a ±0.75 x ±25 x ±25 km “pizza box” avoidance maneuvers are considered. Similarly, ESA’s approach combines a ±10 x ±25 x ±10 km exclusion ellipsoid completed with a collision probability assessment.

Two competing quantities have to be balanced when designing such a criterion:

- The alarm frequency
- A measure of the residual collision risk

An effective risk criterion has to establish a good trade-off between alarm frequency and collision risk reduction.

4.1.1 Commercial Risk Assessment software

There are two NASA methods to compute the reentry survivability of spacecraft components. The Debris Assessment Software (DAS) is a conservative, easy-to-use tool; the Object Reentry Survival Analysis Tool (ORSAT) is a more accurate, higher fidelity model requiring operator expertise and training.

The DAS tool is periodically updated by NASA to reflect the space debris in Earth orbit.

The ORSAT tool is the primary NASA computer code for predicting the reentry survivability of satellite. This prediction is required in order to determine the risk to humans on the ground. According to [8] the process based on the predicted

total debris casualty area, orbit inclination, and year of reentry, shows that this risk should be less than 1:10,000.

4.1.2 Conditions of the Experiment

To simulate the possible risk situations of our satellite we use public databases of airspaces, satellites, space debris and asteroids.

We follow a similar method to the used by the NASA; the difference is that we use a sphere's geometry around the satellite to delimitate the zone of risk, in front of the box's geometry used by the NASA. In this way, we analyze if there is some object inside a 100 km radius sphere around our satellite. We use this distance because is more than the double of the most demanding in the distances used by the NASA that is 40 km.

Even the satellite is in the space, we can consider the case if a country do not like the pass of our satellite over them. We will analyze in section 4.2 the total time when the satellite is over a controversial country. We use the example of *North Corea*, so we add a virtual ground station located at their capital *Pionyang*. To do this, we have added the parameters of this station in the *Stations.csv* file of the *Moon2.0*.

```
NORTH COREA,Pionyang,39.0333,125.7500,27
```

Thanks to the parameter *Station visibility angle* we define the detection range of this virtual ground station changing this value from 5º to 45º that corresponds to a radius of 100 km for an altitude of 250 km. For a correct operation of the simulator, this parameter should be reset to 5º when the simulation is completed.

4.2 Airspace risk assessment during the launch trajectory

To analyze the airspace risk during the launch trajectory, it is necessary to divide the launch in different phases.

The first stage is the ascension of the balloon, in which the risks are: the interference with aircraft performances, the fall of fragments of the balloon onto the people responsible of the launch or the fall of some parts of the balloon over the sea. For these reasons we simulate for how long will the balloon be flying over the land and in concrete inside an aerospace. Also we analyze if it will pass over highly populated areas. Figure 14 shows the simulation of the rocket pass over the land. The rocket flies over not much populated areas, only around *Girona*, and stays a very short time over the land, only during 2 minutes and 11 seconds.

Figure 14 –Simulation of the flight over populated areas and time remaining over the land

To realize this stage of the mission, in which a free balloon of Light category is dropped, it is necessary present to the airworthiness authority (in *Spain*, the AESA agency) a study with the prediction of the balloon trajectory and the impact to the air traffic control with a month of advance. To make this, the simulator *Moon2.0* has an option that shows the moment when the balloon enters to an airspace and when the balloon leaves it. By this method, we can calculate exactly the time while the balloon is in a concrete airspace, because this information is necessary to be specified to the air traffic controller. How to use this tool to generate a report is explained in the ANNEX that will be also included with the *Moon2.0* official open source code.

When the airworthiness authority has received this information, consults to the airways operator and to the general staff if the balloon can cause conflicts. If the response is negative, the airworthiness authority generates a NOTAM that is published three days before the launching. The day of the launching, it is necessary to call to the air traffic controller to coordinate the maneuver. It is possible that the air traffic controller closes the airspace that the balloon crosses. Then it is important to know for how long the balloon will be inside this. The airspaces have an altitude of 18 km and the Spanish territory has an altitude of 30 km. Due to the launching is realized in the stratosphere, the Decree of the 4th of May of 1968 about the “Launching of space engines of private character” [9] is not applied. The appendix S of the “Regulation of Air Circulation” RD57/2002 [10] establishes the regulations about the activity of unmanned free balloons of *Light category*.

The study about the pass of our balloon through some airspaces shows that it only enters in two airspaces: the airspace corresponding to the TMA of *Zaragoza’s airport* and the corresponding to the airway *W-851*. It stays in the TMA of *Zaragoza’s airport* airspace during 18 minutes. In the airspace of the airway *W-851* it stays during 16 minutes. And in global, it stays in the Spanish territory during 98 minutes until it arrives at an altitude of 30 km. Figure 15 and Figure 16 shows these steps of the balloon trajectory.

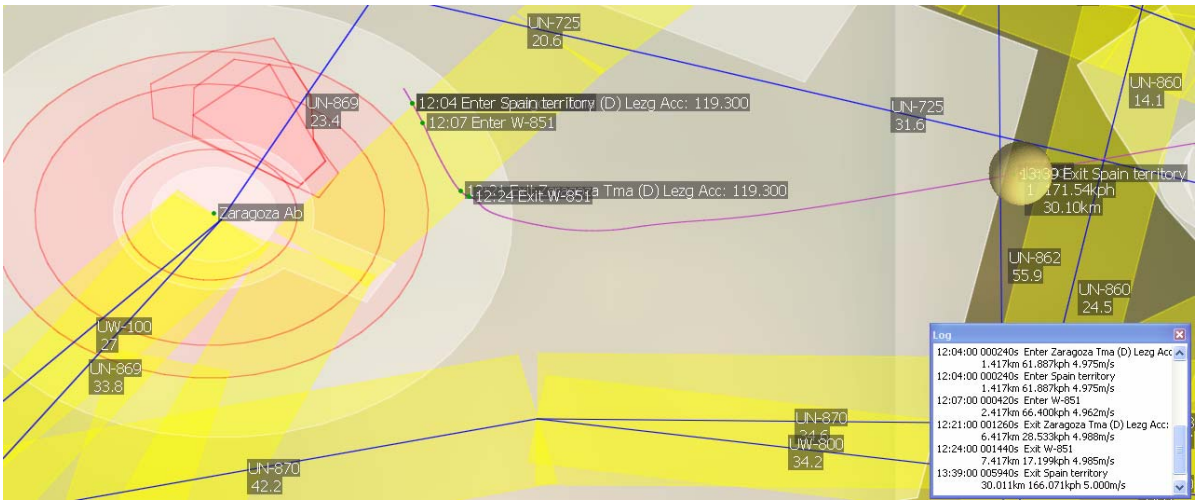


Figure 15 – Top view of the balloon trajectory

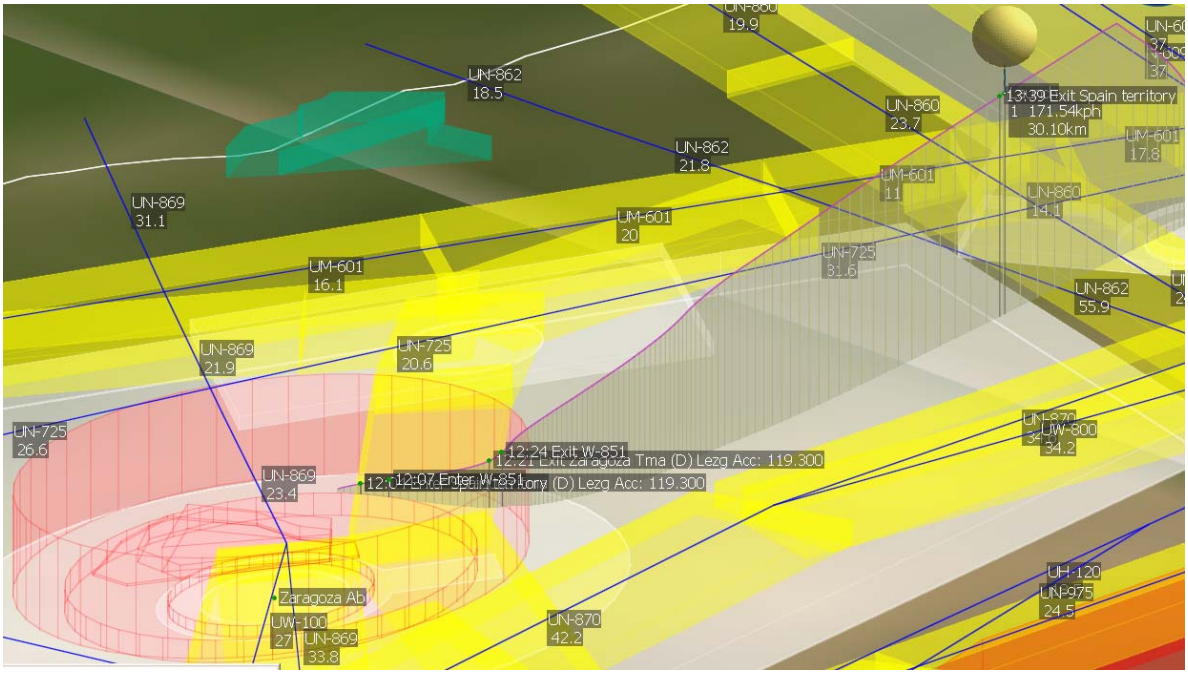


Figure 16 – Isometric view of the balloon trajectory

The next phase is the ignition, in which the risk is the failure of the combustion. If the ignition fails, the balloon will continue ascending until it bursts and its remains will fall down to the sea. If the ignition is produced properly, there is the risk that all explodes, but after that the fire will be extinguished by itself and the parts of the balloon will fall down to the sea. And the other risk after the good ignition is the deviation from the trajectory; in this case the consequence is the total burning due to the atmosphere drag. It is necessary to say that all these risks are not very dangerous because these stages of the launch take place outside any airspace and onto the sea and burned during the atmosphere reentry.

The following step is the apogee maneuver. If this maneuver and the satellite injection fail, there is a risk of total burning if the orbit is not circular and the satellite returns to the atmosphere.

The last phase is the orbital fly. In this step there is only the risk of the degradation of the orbit, that in which case will result in the reentry of the satellite to the atmosphere and the burning of it.

Apart from the trajectory stages of the satellite, it may be the possibility of the disagreement of some countries about the pass of our satellite over them. Assuming a not peaceful use of this amateur satellite where it is used to take photos of a country in conflict we study the magnitude of a menace. Because of this, we simulate the pass of our satellite over one country to demonstrate that the time while the satellite will be over it is really short and a hypothetical spy action cannot be maintained for long periods. We take the example of *North Corea*.

To analyze this case we calculate for how long time a virtual ground station situated in the capital of *North Corea, Pionyang*, is available. Configuring the simulator as explained in section 4.1.2 we placed a 45º cone in *Pionyang city.* We observe that the satellite crosses 13 times for an average period of 40 seconds. In global, this station is available during 509 seconds in all the trajectory of our satellite.

The mission lasts 806,984 seconds (9 days 8 hours 9 minutes and 44 seconds), so the virtual *Pionyang station* is available less than the 0.1% of the total time. The proposed client camera (TOSHIBA TCM8230) has a resolution of 660x492 pixels and 8 bits of pixel color resolution in a normal mode. The pixel size projected on ground is about 408.77 meters, enough to see a disaster but not for spy a country. A single picture is less than 318 kbytes. The satellite download rate is 250 kbps hence it requires less than 11 seconds. It is possible to download a picture to the Corean agency 12 times but in any case, this action is hold only for one week. The satellite cannot spy a country for a long period.

Figure 17 shows the simulation of the *Pionyang station* availability, so that the green lines represent the segment of the trajectory in which this station is available and the red lines represent the unavailability of the *Pionyang station*.

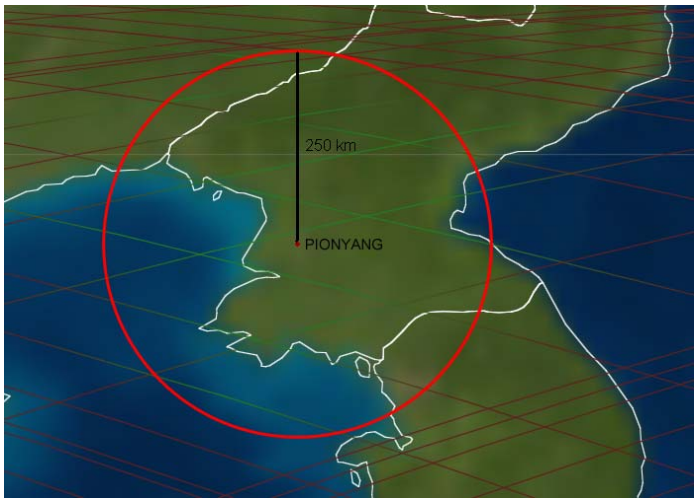


Figure 17 – Simulation of the Pionyang station availability

4.3 Satellite, space debris and asteroid risk assessment

To analyze the risk of collision of our satellite with others satellites, space debris or asteroids we use a similar method that the used by the NASA, like we explain in section 4.2.1. Therefore we simulate the complete trajectory of our satellite and we observe the maximum proximity with other objects.

With respect to other satellites, we use a limit distance of 100 km, which is more than the double of the maximum distance assumed by the NASA, 40 km. During all the trajectory of our satellite, there are 8 cases when the distance between our satellite and other satellite is less than 100 km. During our mission there were 1,488 satellites, so only the 0.5% of satellites are nearer than 100 km.

The nearest satellite is the *H-2A R/B*, so the minimum distance between our satellite and other satellite is 96.37 km. This distance is so higher than the considered by the NASA to require notifications about objects proximity; because of this we can neglect the probability of collision of our satellite with other satellites. The simulation about this is shown in the Figure 18 where the worst case is shown.

The accumulative total conflict time is 140 seconds. The mission lasts 806,984 seconds (9 days 8 hours 9 minutes and 44 seconds). The probability of conflict in this simulation for the whole mission is $P_{conflict}=1.7\cdot10^{-4}$. This result is a bit greater than the considered by NASA in the Debris Assessment Software that is 10^{-4} and this is due to the conjunction area is taken as 100 km instead of 40 km.

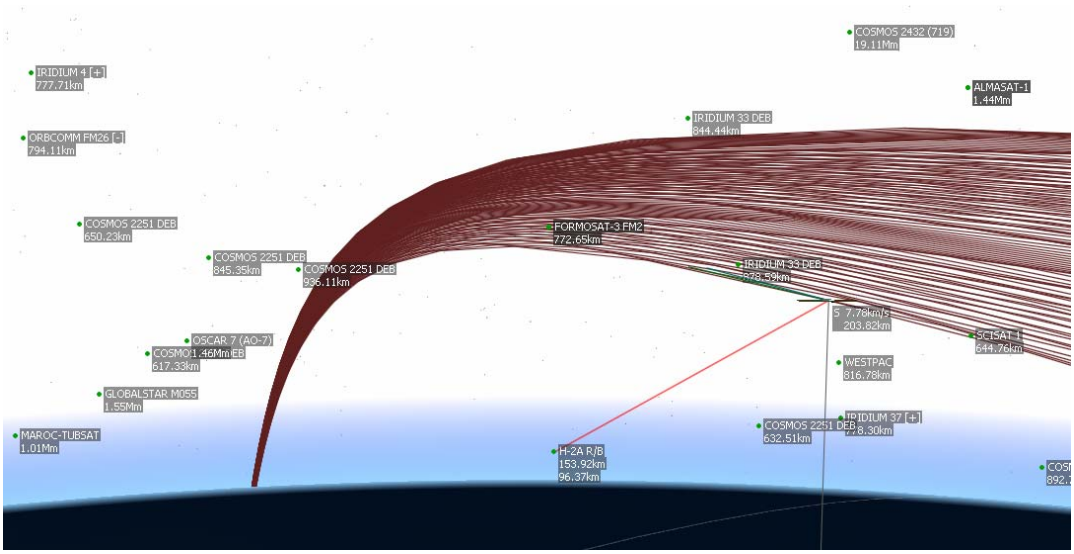


Figure 18 –Simulation of the nearest satellite

Regarding the space debris, there were 1,831 units during our mission and 62 units of these were nearer than 100 km from our satellite, which represent the 3.4% of the total.

The minimum distance between our satellite and any space debris is 88.05 km. This distance is much higher than the considered by NASA (See section 2.1, page 2 from **Péret** *et al.* [7] considered as 10x40x40 km^3 box), like happens with the case of satellites, so we can be sure that there is not risk of collision of

our satellite with space debris. Figure 19 shows the simulation of the nearest space debris during the complete trajectory of our satellite. The total conflict time is 858 seconds. The mission lasts 806,984 seconds (9 days 8 hours 9 minutes and 44 seconds). The probability of conflict in the whole mission is $P_{conflict}=1.06\cdot10^{-3}$.

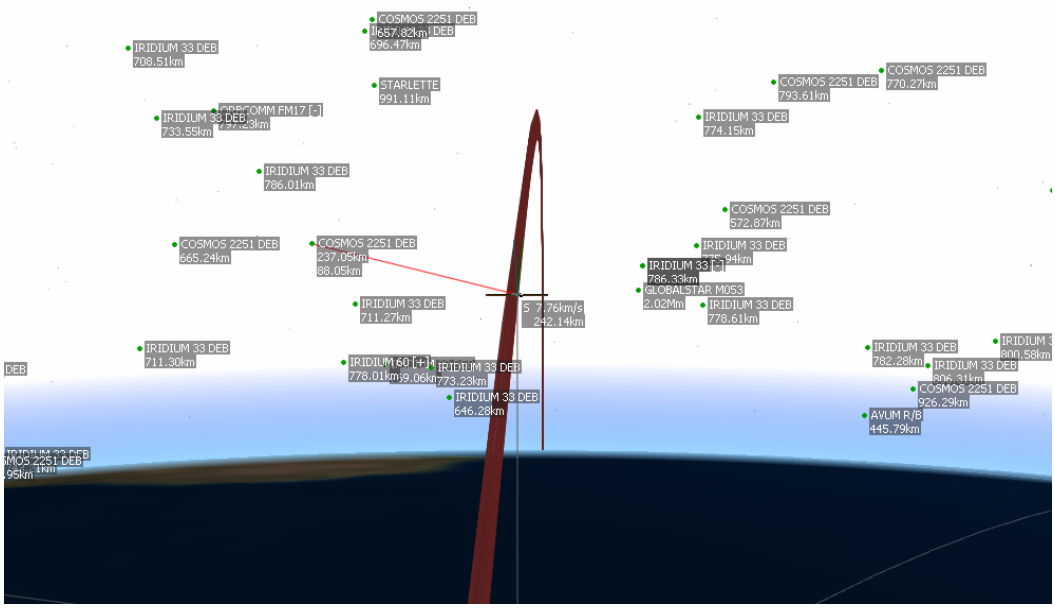


Figure 19 –Simulation of the nearest space debris

Finally, with respect to the asteroids, we simulate the proximity with our satellite using the maximum limit distance available in the simulator *Moon2.0*, which is 100 Gigameters (100 Gm). We test there is no one closer than this distance, so the probability of a collision between our satellite and an asteroid is practically negligible.

To summarize the results, the Table 5 reports the basic information about the statistics for every assessment: Satellites, Space debris and Asteroids.

Table 5 –Summary of WikiSat probability of conflict

Celestial body	Population	Sphere radius	Conflict number	Nearest	Worst distance	Total conflict time	Probability
Satellites	1,488	100 km	8(0.5%)	H-2A R/B	96.37 km	140 s	$1.7\cdot10^{-4}$
Space debris	1,831	100 km	62(3.4%)	COSMOS 2251 DEB	88.05 km	858 s	$1.06\cdot10^{-3}$
Asteroids	-	100 Gm	-	-	-	-	-

CHAPTER 5. THERMAL STUDY

This chapter is a satellite thermal study for the previous mission of Disaster Management in order to know the temperature range. The satellite has not a dynamic thermal control system. It should be studied in advance. Improvements are proposed for the future satellite version.

5.1 Thermal environment

Some simulations were run in the *Moon2.0* simulator in order to see the temperature evolution that the *WikiSat* is exposed to. This evolution is summarized in Figure 20 for every phase or step of the mission: Ascending of the balloon, Launch, Orbit and Reentry.

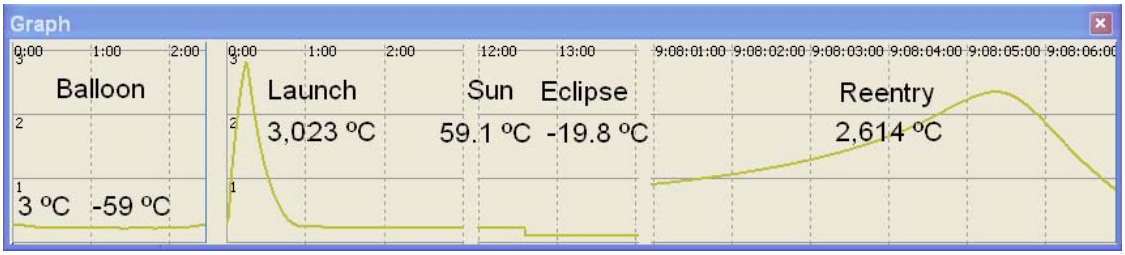


Figure 20 – Temperature evolution

In the ascending of the balloon the temperature starts from the ground temperature, which in the simulation was 3 ºC. In the tropopause reaches to -59 ºC but increases in the stratosphere up to -3 ºC. After the ignition, in the launch phase, temperature starts from -59 ºC but rapidly increases due to the tiny atmosphere drag and the huge speed. It reaches a peak of 3,023 ºC for few seconds and descends up to 59.1 ºC. In orbit, after the injection, the satellite has a temperature of 59.1 ºC and during the Eclipse the temperature lowers up to -19.8 ºC. Finally, in the reentry, the satellite heats up to 2,614 ºC for few minutes results in the satellite burn.

5.2 Old version satellite thermal study

The thermal environment is determined by the heat flow received by the satellite and the heat flow that the satellite dissipates. It was initially studied by **Borras** in [16] in Chapter 7.3 more focused in femtosatellites, **Ferre-Ponsa** in [17] studied the radiation effects over the *WikiSat* onboard computer in vacuum. The satellite receives heat flow from three different sources: the Sun, the Earth Albedo (Sun radiation reflected by the Earth) and the infrared (radiation emitted by the physical matter). Figure 21 shows and scheme of the heat flows entering and leaving the satellite.

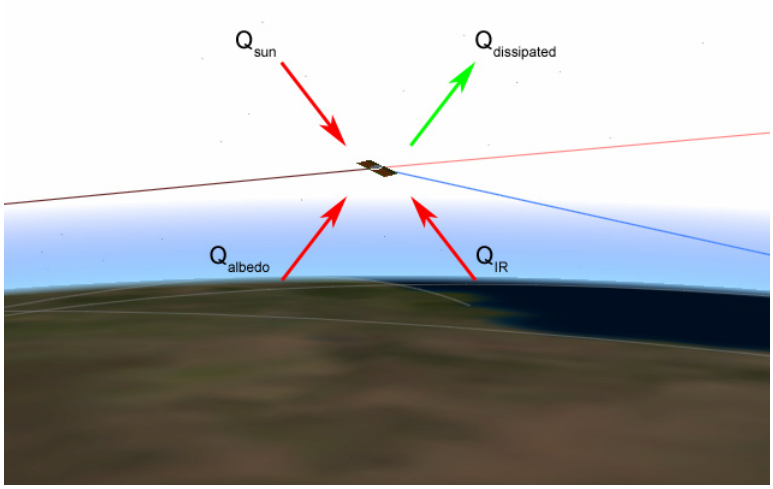


Figure 21 – Heat flow balance

The old *WikiSat* satellite has a passive method of thermal control, which is based in the equilibrium between the emissivity[10] of a material and the heat retention of another. A layer of copper irradiates heat, while a fiberglass layer retains it; in this way, we can control the temperature of all the board of the satellite adjusting the amount of copper and fiberglass. The dimensions of this passive thermal control design were based in a layer of copper of 25x170.75 mm and a fiberglass layer of 9.45x170.75 mm. These dimensions were limited by the antennas, because these materials situated too near to them can produce Electromagnetic Interferences (EMI). Calculations to know the heat flows and the temperature in different cases were done and the Table 4 summarizes the results.

Table 6 –Thermal budget overview of the old WikiSat (Adapted from Tristancho)

Maximum cooling heat flow						
	Q_{Sun}	Q_{Albedo}	Q_{IR}	$Q_{Dissipated}$	Q_{TOTAL}	Temp. K (ºC)
Cooling	-	-	-	-	3.911 W	333.15 (60ºC)
Sun case						
	Q_{Sun}	Q_{Albedo}	Q_{IR}	$Q_{Dissipated}$	Q_{TOTAL}	Temp. K (ºC)
Idle case	3.086 W	0.771 W	0.621 W	0.055 W	4.532 W	346.19 (73.04)
TX case	3.086 W	0.771 W	0.621 W	0.860 W	5.337 W	360.63 (87.48)
Eclipse case						
	Q_{Sun}	Q_{Albedo}	Q_{IR}	$Q_{Dissipated}$	Q_{TOTAL}	Temp. K (ºC)
Idle case	0.000 W	0.000 W	0.621 W	0.055 W	0.676 W	215.12 (-58.03)
TX case	0.000 W	0.000 W	0.621 W	0.860 W	1.481 W	261.73 (-11.42)

Observing the Table 6 we can see that the most restrictive temperatures that the satellite will acquire are 87.48 ºC and -58.03 ºC, so all the components have to be able to operate between these margins. In Chapter 1.2 we analyzed the components of the satellite, their characteristics and their temperature margin of operation. Only the structure and the attitude control can work in this margin of temperatures; the others have a more restrictive margin of operation. The most restrictive component in this sense is the camera, which has the margin of correct operation between -20 ºC and 60 ºC. In conclusion, for the old satellite

[10] Emissivity list: http://www.zytemp.com/infrared/emissivity.asp

budget most of the components will not be able to operate in the range of temperature that the satellite will be exposed to. The conclusion is that the satellite was not well designed related to the thermal control issue.

5.3 Improvements for the satellite thermal control

After this work, the *WikiSat* team decided to improve the satellite keeping the old electronics design but improving other physical aspects. This satellite will be called *WikiSat v6* and referred as the new satellite.

We propose a smaller satellite version adapted to the modifications done by the team who proposed to move to a more adequate radio band for this kind of applications; UHF instead the 2.4 GHz Wifi frequency. Substitute the old ceramic array antennas and the Microstrip adaptation lines by a dipole antenna. Replace the old small battery by a larger one. The radio generates less heat since the radiation power is smaller than the previous version.

The satellite size is constrained by the larger battery (LiPoly 60x38x3 mm). The dipole antenna do not constrains the satellite size since two wires are integrated inside the satellite and deployed in the orbit.

The satellite thickness will be 5 mm. The satellite size will be 60x40x5 mm (LxWxH). Thus for the passive satellite thermal control, the copper layer in the Sun exposed surface will be the *A* parameter and the fiberglass layer will be the *B* parameter. All surfaces will be made of polished copper with an emissivity of 0.015 and the black fiberglass layer will have 0.95 of emissivity. Figure 22 shows the minimum and the maximum temperature equilibrium for different values of the *A* design parameter. The range between 22 and 26 mm of copper layer provides a fine manufacturing adjustment of 1 ºC. Outside this band, the temperature could be out of the satellite operational range between -20 to 60 ºC. The best value in the yellow range of the Figure 22 is 23 mm of copper layer.

The camera may incorporate a Lens (Telephoto lens) in order to improve the field of view (FOV) angle. The *MC5970P-C* lens[11] will be placed in a cylindrical tube of 70 mm in length (I parameter) and a diameter of 21.4 mm (J parameter). The surface will be painted with bronze paint of 0.74 of emissivity. This lens provides a FOV of 8º as depicted in figure 22.

11 http://3rdee.en.alibaba.com/product/1005807083-200277650/8deg_VOA_pinhole_lens_10m_clear_video_520TVL_Mini_CCTV_Camera_MC5970P_0_008LUX_small_size_with_2_install_holes_.html

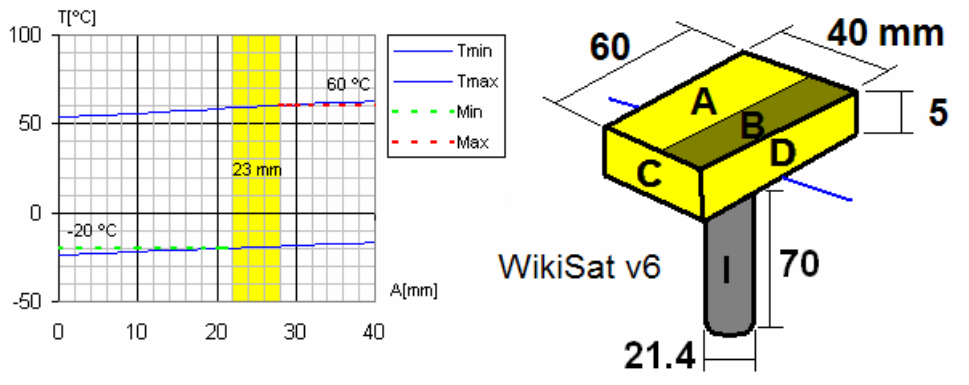


Figure 22 – Temperature range of the new WikiSat version and dimensions

5.4 Satellite thermal control design

The thermal control is based on 3 adjustments. The main adjust is based on the selection of emissivities of the satellite faces which gives a passive temperature range of 150 degrees. The second adjustment is based on a face exposed to the Sun that has two surfaces with opposite emissivity and it is possible to choose the size, providing a passive temperature adjustment of 1 degree. The third adjustment is a dynamic control of the satellite temperature with a range of 4 degrees. This dynamic temperature control can be only used in the Eclipse case.

5.4.1 Average emissivity calculation

The passive thermal design in the old *WikiSat* as calculated by **Navarro** in [15] is based on the correct choose of face material selection. The new satellite is mainly a box of 60x40x5 mm where a large face is exposed to the Sun. In the opposite face which is exposed to the Albedo and the Earth IR sources there is a telephoto lens of 70 mm in length and 21.4 mm of diameter.

When the satellite is exposed to the Sun, large amount of heat is incoming through but when Eclipse, only the Earth exposed surfaces are warming the satellite. The satellite is made of polished copper (Faces *A*, *C*, *D*, *E*, *F* and *G*) which sets the minimum temperature range of 74 degrees between the minimum during the Eclipse and the maximum exposed to the Sun.

The average satellite emissivity is $\bar{\varepsilon}_{Sat}=0.637$ when the *A* face defined by the *A* parameter is 23 mm that is justified later:

$$A\ face = S_A \cdot A = 0.060 * 0.023 = 0.0014\, m^2$$
$$B\ face = S_B \cdot B = 0.060 * 0.017 = 0.0010\, m^2$$
$$C\ face = 0.040 * 0.005 = 0.0002\, m^2$$
$$D\ face = 0.060 * 0.005 = 0.0003\, m^2$$
$$E\ face = 0.040 * 0.005 = 0.0002\, m^2$$
$$F\ face = 0.060 * 0.005 = 0.0003\, m^2$$
$$G\ face = 0.060 * 0.040 = 0.0024\, m^2$$
$$I\ face = 0.25 * \pi * (0.021)^2 * 0.070 = 0.0252\, m^2$$

$$Sun\ face = A + B = 0.0024\, m^2$$
$$Earth\ face = G + I = 0.0276\, m^2$$
$$S = A + B + C + D + E + F + G + I = 0.0310\, m^2$$

$$A_\varepsilon = A * 0.015 = 0.000021$$
$$B_\varepsilon = B * 0.95 = 0.000969$$
$$C_\varepsilon = C * 0.015 = 0.000003$$
$$D_\varepsilon = D * 0.015 = 0.000005$$
$$E_\varepsilon = E * 0.015 = 0.000003$$
$$F_\varepsilon = F * 0.015 = 0.000005$$
$$G_\varepsilon = G * 0.015 = 0.000036$$
$$I_\varepsilon = H * 0.74 = 0.018631$$
$$\sum A \cdot \varepsilon = 0.0197$$

$$\bar{\varepsilon}_{Sat} = \frac{\sum A_n \varepsilon_n}{S} = \frac{0.0197}{0.0310} = 0.635$$

5.4.2 Coarse emissivity adjustment calculation

The following step is to reduce the telephoto lens emissivity, initially made of fiberglass, through selecting the correct paint color (See *ZyTemp* emissivity list in the previous page). This emissivity selection can set the satellite from -34 ºC of minimum to 368 ºC of maximum and a temperature range of 152 degrees.

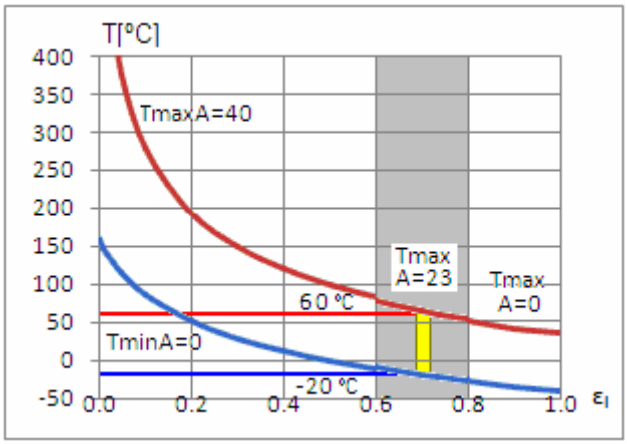
Calculations were done in an EXCEL sheet and solved by iteration and extra information is available in the Annexes A.4 where the temperature range is adjusted initially by changing the telephoto lens paint ε_I .

The graph of the Table 7 shows the whole temperature range when the telephoto emissivity is studied for all the range. For values closer to zero, the satellite is really hot and the temperature range is wide but when is closer to one, the temperature range is smaller as the one supported by the satellite components. The yellow area is the valid for the coarse adjustment.

5.4.3 Fine emissivity adjustment calculation

When the coarse adjustment is done, the *A* parameter provides a fine adjustment. The bronze paint of the telephoto lens has an emissivity of 0.74. When the telephoto lens is painted with it, provides a temperature range between -23.8 ºC of minimum and 63.3 ºC of maximum as presented in Table 7 and a temperature range of 79 degrees.

Table 7 – Temperature manufacture adjustment by telephoto lens paint ε_I (coarse adjustment) and Sun faced copper layer A (fine adjustment)



	Too cold				Small manufacture adjustment						Too hot			
L[mm]	40	40	40	40	40	**40**	40	40	40	40	40	40	40	40
W[mm]	60	60	60	60	60	**60**	60	60	60	60	60	60	60	60
H[mm]	5	5	5	5	5	**5**	5	5	5	5	5	5	5	5
A[mm]	0	10	20	21	22	**23**	24	25	26	27	28	29	30	40
B[mm]	40	30	20	19	18	**17**	16	15	14	13	12	11	10	0
I[mm]	70	70	70	70	70	**70**	70	70	70	70	70	70	70	70
J[mm]	21.4	21.4	21.4	21.4	21.4	**21.4**	21.4	21.4	21.4	21.4	21.4	21.4	21.4	21.4
Tmin[ºC]	-23.8	-22.1	-20.3	-20.1	-20.0	**-19.8**	-19.6	-19.4	-19.2	-19.0	-18.9	-18.7	-18.5	-16.6
Tmax[ºC]	53.9	56.1	58.4	58.7	58.9	**59.1**	59.4	59.6	59.9	60.1	60.3	60.6	60.8	63.3
Tdelta	77.7	78.2	78.7	78.8	78.9	**78.9**	79.0	79.0	79.1	79.1	79.2	79.3	79.3	79.9
Min[ºC]	-20	-20	-20	-20										
Max[ºC]										60	60	60	60	60

5.4.4 Maximum cooling heat flow calculation

The maximum cooling heat flow, assuming the satellite in the hotter limit at 60ºC (333.15 K) is able to dissipate the following amount of heat:

$$Q_{Cooling} = \sigma T^4 \sum A \cdot \varepsilon = 5.67 \cdot 10^{-8} \cdot (333.2)^4 \cdot 0.0197 = 13.740 W$$

5.4.5 Heat flow balance calculation

When the satellite is exposed to the Sun $E_{Sun} = 1{,}420 W$, faces A and B are heated and more or less faces from *C* to *F* are heated too, but faces *G* and *I* are in the shadow towards the Earth. Since Sun rays are somehow parallel, the form factor is $k_{Sun} = 1$.

Faces *G* and *I* receive heat from Albedo and IR sources. The infrared energy is $E_{IR} = 244 W$ and the form factor is $k_{IR} = 0.75$ because rays are coming from the near Earth, rays are not parallel. The heat from the Albedo is the Sun heat reflected by the Earth while the infrared heat is due to the Earth temperature. We assumed Albedo as $E_{Albedo} = 0.32 \cdot E_{Sun} = 454.4 W$ and the form factor is

$k_{Albedo} = 0.5$ because rays coming from the Earth reflection are very dispersive. For the Sun case, there are two conditions when Idle and when Transmitting:

$$Q_{Sun} = \alpha \cdot E_{Sun} \cdot k_{Sun} \cdot A_{Sun} = 0.9 * 1{,}420 * 1 * 0.0024 = 3.067W$$
$$Q_{Albedo} = \alpha \cdot E_{Albedo} \cdot k_{Albedo} \cdot A_{Albedo} = 0.9 * 454.4 * 0.5 * 0.0276 = 5.639W$$
$$Q_{IR} = \alpha \cdot E_{IR} \cdot k_{IR} \cdot A_{IR} = 0.9 * 244 * 0.75 * 0.0276 = 4.542W$$
$$Q_{Idle} = 0.055W$$
$$Q_{TX} = 0.350W$$

$$Q_{SunIdle} = 3.067 + 5.639 + 4.542 + 0.055 = 13.304W$$
$$T_{SunIdle} = \sqrt[4]{\frac{Q_{Idle}}{\sigma \sum A \cdot \varepsilon}} = \sqrt[4]{\frac{13.304}{5.67 \cdot 10^{-8} \cdot 0.0024}} = 330.5K = 57.3^{o}C$$
$$Q_{SunTX} = 3.067 + 5.639 + 4.542 + 0.350 = 13.598W$$
$$T_{SunTX} = \sqrt[4]{\frac{13.598}{5.67 \cdot 10^{-8} \cdot 0.0276}} = 332.3K = 59.1^{o}C$$

Opposite, when the satellite is in Eclipse, only faces *G* and *I* receive heat from IR sources. For the Eclipse case, there are another two conditions when Idle and when Transmitting:

$$Q_{EclipseIdle} = 4.542 + 0.055 = 4.597W$$
$$T_{EclipseIdle} = \sqrt[4]{\frac{4.597}{5.67 \cdot 10^{-8} \cdot 0.0024}} = 253.4K = -19.8^{o}C$$
$$Q_{EclipseTX} = 4.542 + 0.350 = 4.892W$$
$$T_{EclipseTX} = \sqrt[4]{\frac{4.892}{5.67 \cdot 10^{-8} \cdot 0.0276}} = 257.3K = -15.8^{o}C$$

As we can see, no cases of these four are greater than the dissipated heat what means a correct satellite thermal design; the satellite never will be hotter than the allowed maximum temperature established by the requirements. The maximum allowed by the requirements (See Table 1 in Chapter 1) is an 80 degrees range (From -20 to 60 ºC) so there is 1 degree of margin. Through the selection of the polished copper layer in the Sun exposed face (Parameter *A*) it is possible to set the minimum to -20 ºC and the maximum to 59 ºC. So, the best choice is when the *A* parameter is set to 23 mm which is the coolest inside the range.

5.4.6 *Summary of heat flow for every satellite mode*

Table 8 summarizes the heat flows through the new version satellite in the cases of maximum cooling heat flow, exposition to the sun and eclipse. Also it includes the temperatures in these situations depending on the state of the satellite: idle or transmission.

Table 8 – Thermal budget overview of the new version of the WikiSat

Maximum cooling heat flow						
	Q_{Sun}	Q_{Albedo}	Q_{IR}	$Q_{Dissipated}$	Q_{TOTAL}	Temp. K (ºC)
Cooling	-	-	-	-	13.740 W	333.15 (60ºC)
Sun case						
	Q_{Sun}	Q_{Albedo}	Q_{IR}	$Q_{Dissipated}$	Q_{TOTAL}	Temp. K (ºC)
Idle case	3.067 W	5.639 W	4.542 W	0.055 W	13.304 W	330.47 (57.32)
TX case	3.067 W	5.639 W	4.542 W	0.350 W	13.598 W	332.29 (59.14)
Eclipse case						
	Q_{Sun}	Q_{Albedo}	Q_{IR}	$Q_{Dissipated}$	Q_{TOTAL}	Temp. K (ºC)
Idle case	0.000 W	0.000 W	4.542 W	0.055 W	4.597 W	253.38 (-19.77)
TX case	0.000 W	0.000 W	4.542 W	0.350 W	4.892 W	257.34 (-15.81)

Observing the Table 8 we can see that the most restrictive temperatures that the satellite will acquire are 59.14 ºC and -19.77 ºC, so all the components have to be able to operate between these margins. In Chapter 1.2 we analyzed the components of the satellite, their characteristics and their temperature margin of operation. The most restrictive component in this sense is the camera, which has the margin of correct operation between -20 ºC and 60 ºC. In conclusion, all the components will be able to operate in the range of temperature that the satellite will be exposed to.

5.4.7 Dynamic thermal control calculation

Additionally, since the satellite has a temperature sensor, it is possible to warm the satellite dynamically if required but it is not possible to cool more than the stated by manufacture adjusted. This dynamic control can be done only in the Eclipse case. If the satellite is designed too cool then there is a battery drain and a shorter battery life. The dynamic temperature range $\Delta T_{Dynamic}$ is at least:

$$\Delta T_{Dynamic} = T_{EclipseTX} - T_{EclipseIdle} = (-15.8) - (-19.8) = 4.0\,^{o}C$$

Smooth variations in the satellite orientation can be tolerated in a passive way because of this passive thermal control system and its wide temperature range.

CHAPTER 6. ONBOARD SENSOR DATA ASSESSMENT

This chapter is an assessment of the data downloaded to the ground station in the simulation from the onboard sensor in terms of picture quality and correct information. The assessment is focused on the effects of the minimum valid satellite inclination over the quality of a photo and the results of the simulated downloaded photos taken by the payload camera.

6.1 Satellite minimum valid inclination assessment

In order to do this study, it is very important to fix a minimum inclination that supposes the limit from which the pictures will be considered with good quality; in the *Moon2.0* this parameter is called *Station visibility angle*. To do this, we made the simulation in three different cases, in which the *Station visibility angle* was 30º, 45º or 60º. But in this choice there are two main factors to consider: the quality of the images and the time of contact. In one hand, a pixel size bigger than 500 meters is not considered acceptable; for example, the *ORGVIEW2* use a pixel with a size of 1.1 km and the *LANDSAT TM5* a pixel of 120 meters. The minimum pixel size of the *TOSHIBA* TCM8230 camera is 200 meters [11]. And in the other hand, it is necessary a time of contact higher than 16 seconds, because the satellite lasts 5 seconds to establish the contact and 11 seconds to download an image. These factors are opposite each other, because a small angle implies a longer contact but the quality of the images is worse. Instead of this, a high angle implies a better quality of images but a shorter contact. So it is necessary to get tradeoff.

In the case that our satellite could not get a pixel size smaller than 500 meters, respecting the time necessary to download an image, we will have to propose to the client to change the optics of the camera. For example, the *3rdEye MC5970P* camera has better optics. The angle of view of this camera[12] is about 8º, which allows more precise images, in front of a 57º angle of view[13] of the *TOSHIBA* TCM8230. Apart from this, the optical size of the camera of our client is of 1/6 inch optical format, but the other camera has an optical size of 1/4.

For a *Station visibility angle* of 30º, the images taken were poor and did not show clearly the site. At this inclination the distance is about 440 km and the pixel size projected on ground is about 596 meters, which is not enough. Using this inclination, in the entire mission the satellite establishes 11 contacts with *Matara*, which implies 1,018 seconds (0.28 hours) of connection. All the contacts last more than 16 seconds. In this time, the satellite can download 31,813 Mbytes, equivalent to 102 pictures. Figure 23 is an example of a photo after the establishment time, when the satellite detects that the inclination is higher than 30º, but after this 5 seconds the inclination is about 35º.

[12] http://3rdee.com/View/NewsView.aspx?id=3c77717c-ffd6-4785-93eb-fd0ba1b52e2a
[13] https://www.sparkfun.com/datasheets/Sensors/Imaging/TCM8230MD.pdf

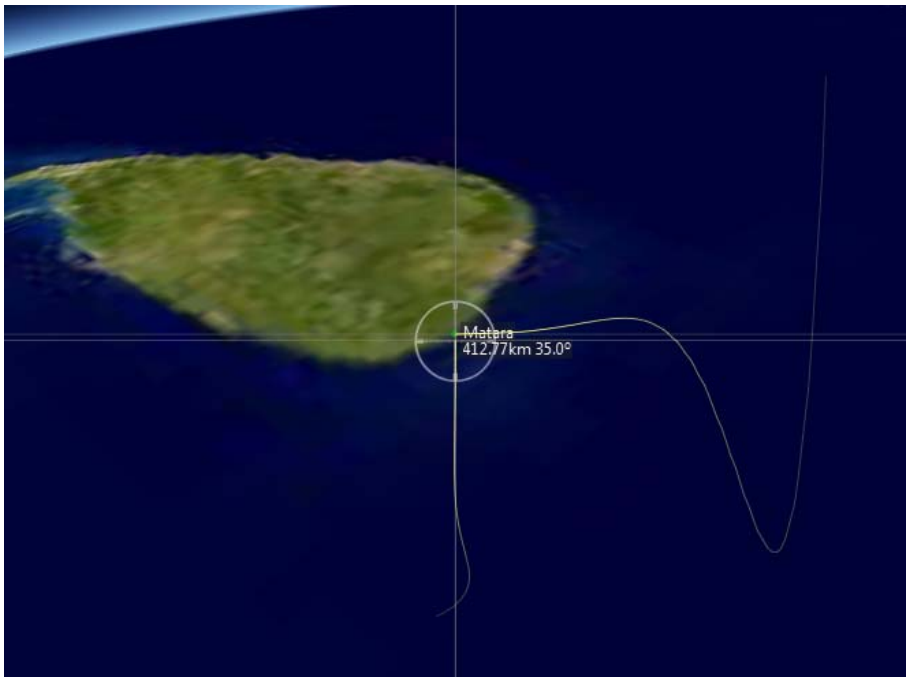


Figure 23 –30º inclination. Pixel size 596 meters

With a 45º *Station visibility angle,* the pictures were acceptable and allow see clearly the site. The distance to the ground station is about 339 km and the pixel size projected on ground is about 426 meters, which is enough for disaster management. The satellite establishes 9 contacts, all longer than 16 seconds. In total, the connection between our satellite and the ground station lasts 468 seconds (0.13 hours), which allows download 14,625 Mbytes, equivalent to 47 pictures. Figure 24 shows an example of a picture taken after the establishment time when the satellite detects that the inclination is higher than 45º, in that moment the inclination is about 53º.

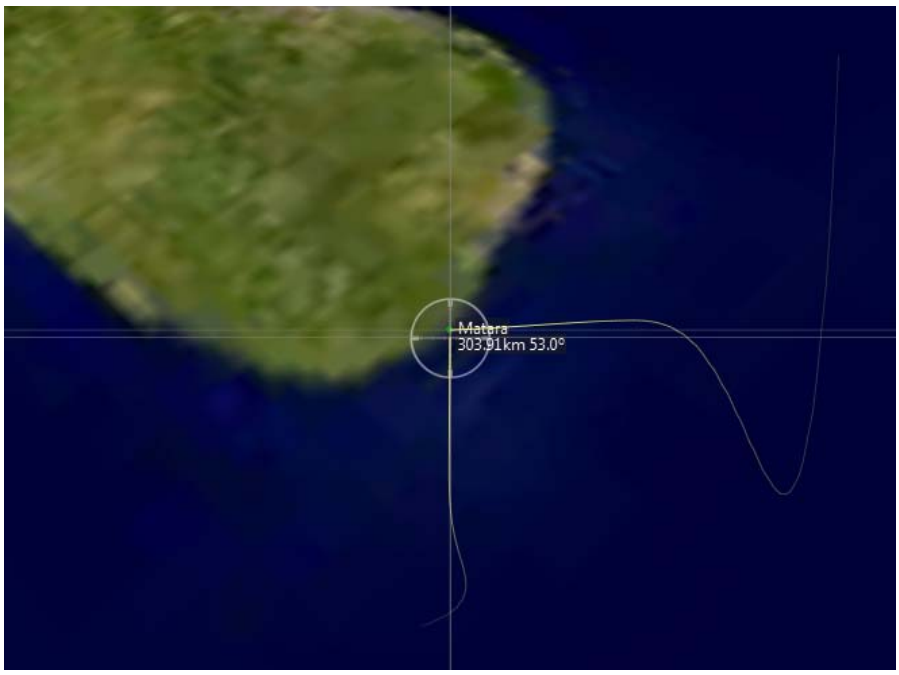


Figure 24 –45º inclination. Pixel size 426 meters

In the case of a 60º *Station visibility angle,* the distance to the ground station is about 280 km and the pixel size is 352 meters. During the entire mission, the satellite establishes 6 contacts, but only 5 contacts last more than 16 seconds. The total time of connection is about 173 seconds (0.05 hours). In this case, the satellite can download a maximum of 5,406 Mbytes, equivalent to 17 pictures. In the Figure 25 we can see a photo taken by the satellite after the establishment time when the satellite detects that the inclination is higher than 60º, but after this 5 seconds the inclination is about 70º.

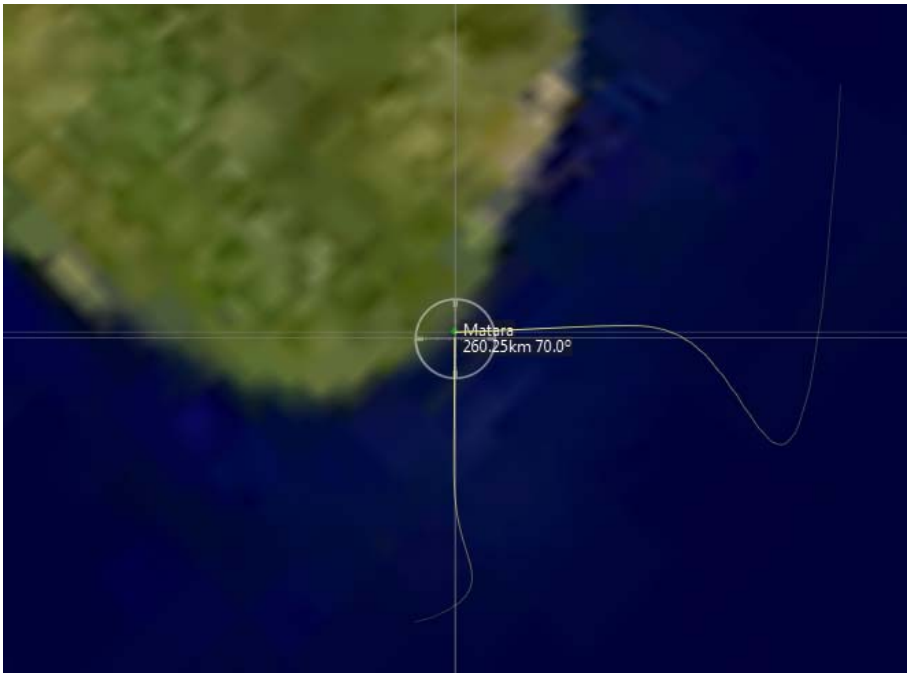


Figure 25 –60º inclination. Pixel size 352 meters

Comparing the three cases, we arrived to the conclusion that the first one is not acceptable because the pictures were too much poor and were not clearly enough for a disaster management. We choose the *Station visibility angle* of 45º, because it allows have good quality pictures, but respecting a long time to download these; due to the inclination of 60º is better for the quality of the pictures but the downloading time is shorter.

The case of *Maspalomas* ground station is different, because in this place the objective is to download images, not to take photos. In this case, the interest is to have the longest contact possible, so an inclination of 5º is enough. Using this *Station visibility angle*, in the entire mission there are 40 contacts between the satellite and *Maspalomas*, which implies 10,077 seconds of connection (2.8 hours). In this time, the satellite can download 314,906 Mbytes, which represent 1,016 pictures.

Following this criterion, we analyze how many times our satellite establish good connections with the ground stations during the complete trajectory, compared with all the connections established. In this way, the connections in which our satellite can take pictures with an inclination higher than 45º will be considered good and all the others will not be accepted.

Table 9 – Assessment of the minimum inclination of the point of interest

Inclination Established	**Pixel Size**	**contacts** >16 seg Average	**Distance to station**	**Maximum download**	**Assessment**	**Broadcasted Photo**
5º 5.4º	1.6 km	40 252 s	1,200 km	314.906 Mbytes 1,016 pictures	The camera is not used. The download time to the ground station is maximum.	
30º 32.5º	596 m	11 92 s	440 km	31.813 Mbytes 102 pictures	The pixel size is too large for a disaster management application like a flood case.	
37º 40.5º	500 m	11 64 s	395 km	22.125 Mbytes 71 pictures	The pixel size is in the limit of pixel size considered good enough for this application.	
45º 49.5º	426 m	9 52 s	339 km	15.625 Mbytes 47 pictures	Pixel size is good and First Responders receive 47 pictures, enough for a flood disaster.	
60º 65.5º	352 m	5 28 s	280 km	5.406 Mbytes 17 pictures	Pixel size is very good for a flood case but only 17 pictures are received by First Responders.	

This study summarized in Table 9 is very useful in order to schedule our satellite. There are two ways in which the satellite can operate. The first one consists in the camera optimization to assure that the pictures taken will follow the criterion of inclination; in this way, when the satellite has a good picture, it sends to the ground station and the camera is turned off. The second one consists in the recording of a video while the satellite flies over the affected site until the inclination of the camera starts to fall and the satellite saves the last image because it assures that the photo saved is the one with the highest inclination and the best characteristics. This allows send it in high quality when it overflies the ground station. Both ways have some advantages in front of the other. The first option allows save on energy, because the satellite only turns on the camera when the requirements of the criterion are followed. This method assures that the images will be acceptable and the satellite can send the picture taken immediately to the ground station that it is overflying. The second option has the advantage of assure the sending of the best photo of all the pictures recorded by the satellite in a video.

Because of the explained reason in the above table, we fixed the minimum inclination valid for the photos taken by the satellite in 45º.

6.2 Downloaded photos assessment

Next a check about the photographed area is realized contrasting a map of the zone affected and a photo taken by the satellite.

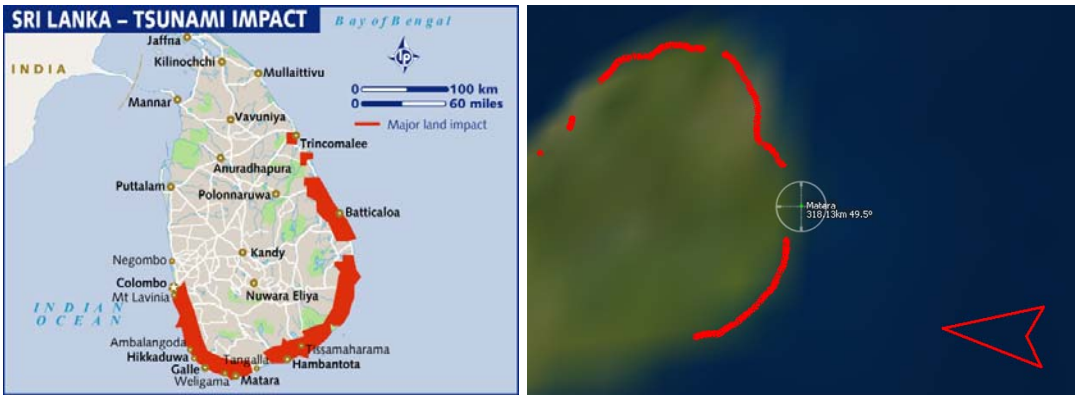


Figure 26 – Affected area and how is photographed by the satellite

In the left photo of Figure 26 we can observe a map of *Sri Lanka* with the affected areas identified in red. The right photo is the photo of *Sri Lanka* taken by the satellite with a minimum inclination of 45º, in which we locate the affected areas like in the map in false red. Contrasting both photos, we can see that the affected areas are photographed by the satellite, which is the objective of our mission. However, there is half photo wasted because it only shows the sea. So we can conclude that in this case the satellite accomplishes its objective, but not optimally. In case of not satisfy the quality of this objective a solution can be taken into account before the real mission.

As a report for the client, the Table 10 with all the valid photos with an inclination higher than 45º and the assessment of these is produced. During the entire mission, for this simulation, *Matara* is photographed 9 times with an inclination of 45º or higher; all these contacts last more than 16 seconds, which is the time necessary to the establishment and the download of an image.

Observing the Table 10 we can see that there are two factors that can convert the photos in useless; one of this is the darkness and the other is the rotation. Two of the nine photos have a light rotation, but this can be solved by a software correction before the launch. However, five photos are dark and this problem does not have any solution. The darkness can be an important problem because at night the satellite will not be able to take photos that show clearly the situation of the site and more than a half of the photos taken are useless because of the darkness.

Table 10 – Quality photos taken by the client sensor with 45º inclination

#	Time **Date**	Inclination **Distance**	Area [km^2] **Pixel[m]**	**Assessment**	**Broadcasted Photo**
1	1,370 s 0:00:22:50	Established at 47.1º 337.44 km	280x204 424.04	Photo not aligned with the horizon. The pixel size does not correspond with the horizon. It can be corrected at ground by software means. Dark photo.	
2	47,259 s 0:13:07:39	Established at 48.3º 328.66 km	273x199 413.74	Good orthophoto like. Good illumination. Photo bearing misaligned; software on ground correction required.	
3	87,264 s 1:00:14:24	Established at 48.4º 326.42 km	271x197 409.94	Photo not aligned with the horizon. The pixel size does not correspond with the horizon. It can be corrected at ground by software means. Dark photo.	
4	133,122 s 1:12:58:42	Established at 49.6º 317.90 km	263x191 398.48	Good illumination. Light rotated photo but it can be corrected at ground by software means.	
5	173,087 s 2:00:04:47	Established at 49.4º 316.67 km	262x191 397.72	Photo not aligned with the horizon. The pixel size does not correspond with the horizon; software solution available. Dark photo; without solution.	
6	218,910 s 2:12:48:30	Established at 48.4º 317.64 km	263x192 398.94	Best orthophoto like. Good illumination. Good photo heading, no software correction required.	
7	258,832 s 2:23:53:52	Established at 48.4º 315.56 km	262x190 396.29	Good inclination but dark photo, solution not available to solve this problem.	
8	736,452 s 8:12:34:12	Established at 50.5º 237.87 km	197x143 298.67	Good illumination and correct inclination but 5 days without a valid photo. Light rotated photo but it can be corrected at ground by software means.	
9	775,800 s 8:23:30:00	Established at 48.4º 223.52 km	185x135 280.77	Good inclination but dark photo, solution not available to solve this problem.	

Excluding the dark photos, there are four photos that show well the affected area. In these four photos the photographed area is not the optimum, because half of this shows the sea but enough for our purposes. In the simulation the first photo is dark but the second photo has a correct view in order to determine the real impact of the disaster. This photo is taken 13 hours after the launch. The second day, the photo number four has a good view of the disaster. In between there is time to update the satellite parameters in order to focus the camera towards an interested area instead of a general photo. Figure 27 shows in the left an example of a magnified photo of the interest area using the Google Earth while in the right there is the simulation picture that is very similar apart of its lower detail.

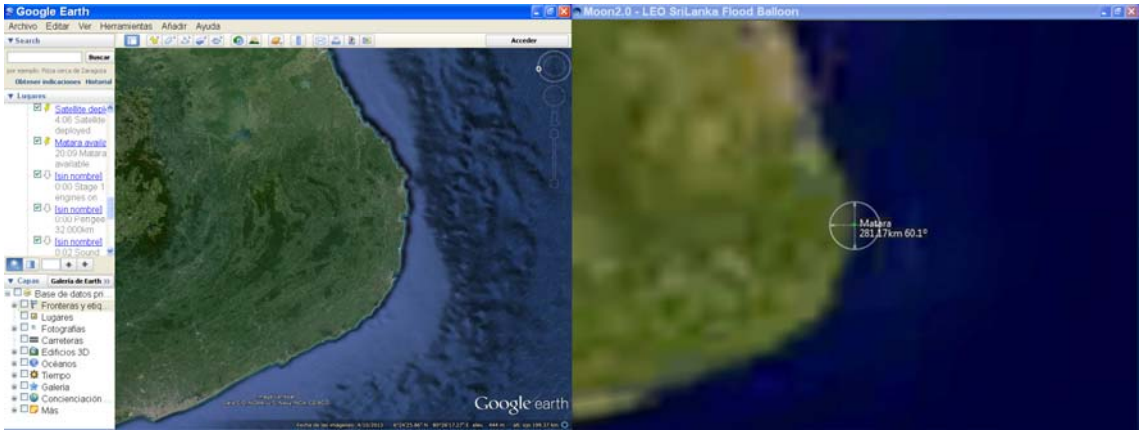


Figure 27 – A real satellite view of the affected area compared with a simulated one

In the three first days take place three of the four good photos; this is an advantage because the client does not have to wait too much to acquire photos that shows clearly the zone of interest. From the third day to the eighth, the satellite is not able to take photos with an inclination higher than 45º. And in the eighth day the satellite takes another good photo. Even the main pictures are obtained in the first days; it is when we most need the pictures in order to take action. For the following days, the UN has established the help and this satellite is no longer useful.

The number of total useful photos is approximately what we expected, so we consider that there are enough photos to inform on time about the situation in a disaster management.

CHAPTER 7. CONCLUSIONS

This Chapter has the general conclusions, the future work and the environmental impact study.

7.1 General conclusions

This final Bachelor work has aimed to study and validate the integration of the payload in the engineering cycle of a femtosatellite.

In Chapter 2 we conclude that the *TOSHIBA TCM8230* accomplishes the mission requirements.

Conclusions for the Chapter 3 are as follow. The satellite is in a Low Orbit of 250 km, which is not Polar and the duration of the mission is about one week, these are the reasons because of the radiation dose received by the satellite is low. The gamma particles produce the TID in the layers of the satellite, but due to the dose is not dangerous and the harmful effects can be detected and corrected. A COIN battery protects the satellite better than the LiPoly battery but it can feed only a camera and a radio. This is because the COIN battery cannot provide enough power to have an attitude control system. It will be useful only in the case of use a swarm of satellite, but it is better to have an attitude control system.

From the Chapter 4 where a risk assessment was done, we can conclude that the risk of collision with other satellites, debris or other celestial bodies is negligible. Since the launcher is released from a stratospheric balloon, the conflicts with the airspace are managed by the air traffic controller. A procedure was published in the *Moon2.0* web for similar Balloon Launch permission reports.

In Chapter 5, with the calculations of the thermal study we observe that the design of the *WikiSat* has a passive thermal control with a wrong design. The ceramic antennas and the lines of microstrip adaptation take up too much surface. The choice of 2.4 GHz is wrong too. If the band UHF is chosen, the range increases significantly and it can be used a traditional dipole antenna which produces less problems, but we lose bandwidth. Also, being realistic, the camera needs a lens which goes in a long cylinder. Because of all these reasons the thermal study has remade using the sizes and the emissivity of the new *WikiSat* design.

As per Chapter 6, from the assessment of the pictures obtained by the satellite, we can conclude that a half of the photos considered valid in terms of inclination are too dark because they are taken during the night. It is not a design problem that has to be improved, it is assumable because we only need the pictures of the first day on time, and because after that the help is already established and the information can be obtained by other ways. Additionally, the satellite minimum valid inclination is 45º. The First Responders are already there and can inform better. In the simulation there are five days without useful pictures. It is important that the period without images cannot happen during the first days

because in these days is when is more interesting to obtain pictures of the catastrophe. It can be achieved doing the launching from a zone near the catastrophe or synchronizing the launching time to make that the first pass of the satellite over the interested area occurs during the day.

The general conclusion is that is valid to integrate this camera as the payload in the design cycle of a femtosatellite, because it accomplishes all the specifications that the designated mission requires and the needed requirements to allow the integration of the camera in the board of the satellite. The camera has an enough resolution and a correct pixel size to assure the obtaining of valid photos during the mission. Its low weight, size, consume and its SMD technology made the camera valid to be integrated in the board of the satellite during the design cycle. Also, the camera has to be able to work in the conditions of the near space, where the mission takes place; with the studies done, we had test that the camera can work in these radiation and thermal conditions. However, some improvements are proposed for the new *WikiSat* version after this work.

7.2 Future work

The version 6 of the WikiSat has been proposed, electrical and thermal budget is done and the radiation study is pending. Anyway, this study will be very similar to the radiation study based on a LiPoly battery, because the layers, the materials and the battery of the satellite have not changed.

7.3 Environmental impact

The *WikiSat Space Program* has an important component of environmental care. These missions reduce a lot the environmental impact comparing with other methods:

- Politics of zero space debris
- Low pollution and
- Very low risk to the population

In chapter 4 - Risk Assessment is studied the impact of the mission over the airspace and the orbit. Using this satellite, we will save energy and pollution in front of the use of large rockets. This is adequate for the environment because it is only launched when it is necessary. The satellite is burned in the atmosphere in only one week, so there is no space debris caused by this satellite. The MEMS components are typically manufactured without lead to not contaminate and the factories follow processes that respect the environment. There are a very low risk for the population due to the low weight of the rocket and the very short time while the rocket is flying over the land. When the satellite is into orbit, the rocket burned in the reentry and there is no space debris after the mission.

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ANNEXES

TÍTOL DEL TFC: Integració d'una càmera en el cicle de disseny d'un femto-satèl·lit i valoració

TITULACIÓ: Enginyeria Tècnica Aeronàutica, especialitat Aeronavegació

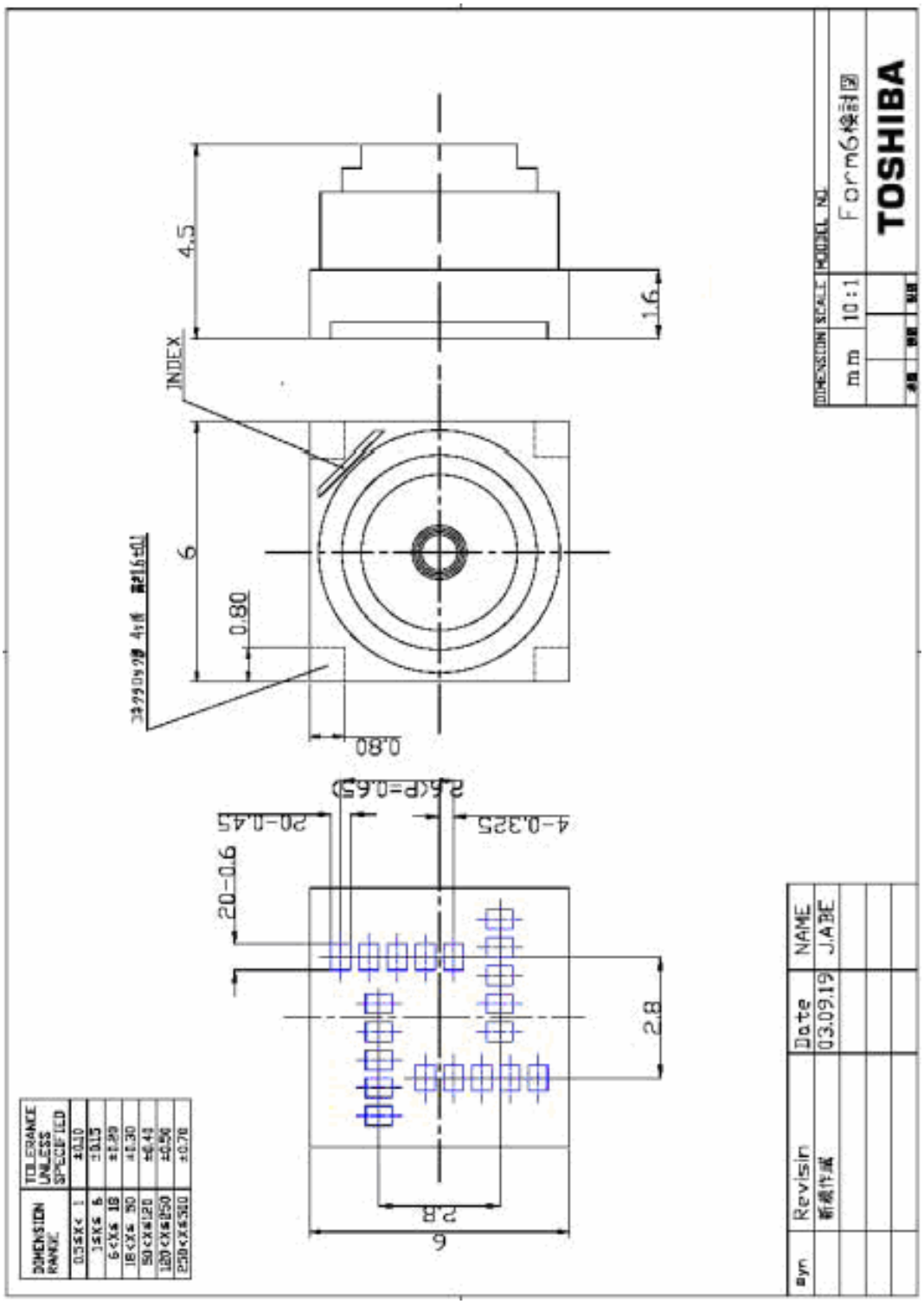
AUTOR: Aina Pallarès Albaladejo

DIRECTOR: Joshua Tristancho Martínez

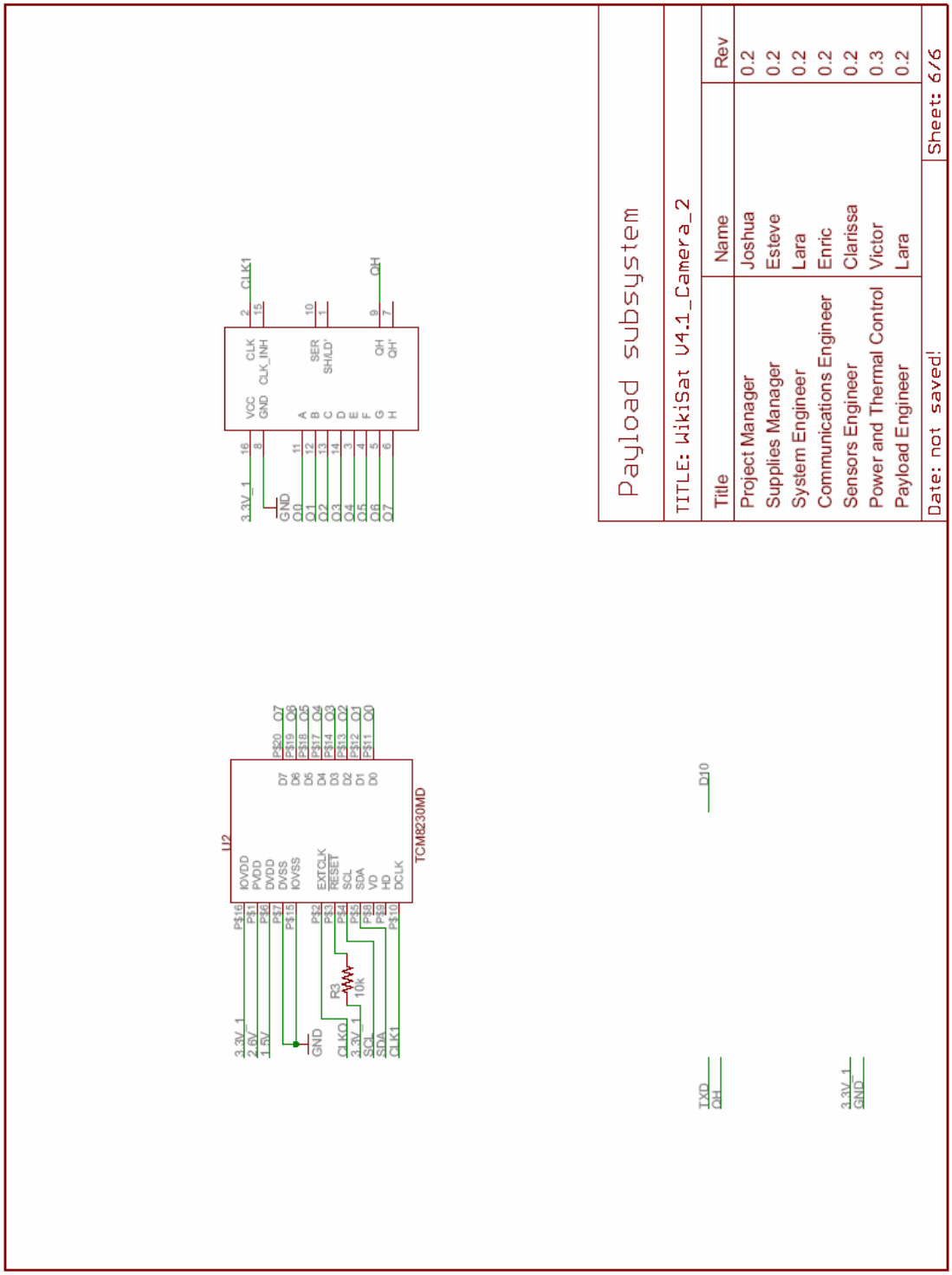
DATA: 31 de gener de 2014

ANNEX A. DRAWINGS FOR THE PAYLOAD

A.1 TOSHIBA TCM8230 camera dimensions

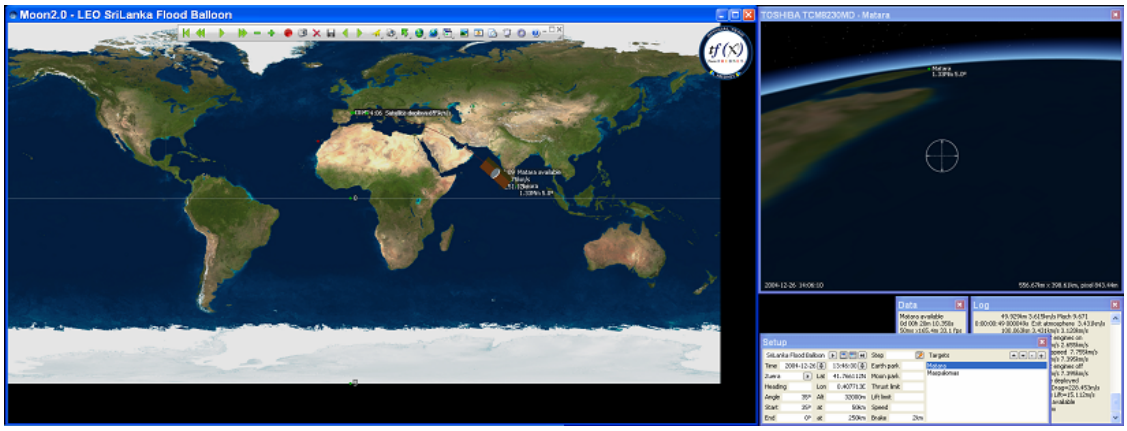


A.2 Connectors and pins of the Payload system (Source: WikiSat team)



Old WikiSat v4.2 with the camera integrated aboard. Source: UPC

Orbit simulation by the open source program Moon2.0 and Ground station. Source: UPC



Satellite propagation tool and the satellite sensor simulator. Source: UPC

Approaching to the interest point for 5.4º, 32.5º, 40.5º, 49.5º and 65.5º satellite inclination

A.3 Map resolution for a given lens option of the satellite camera

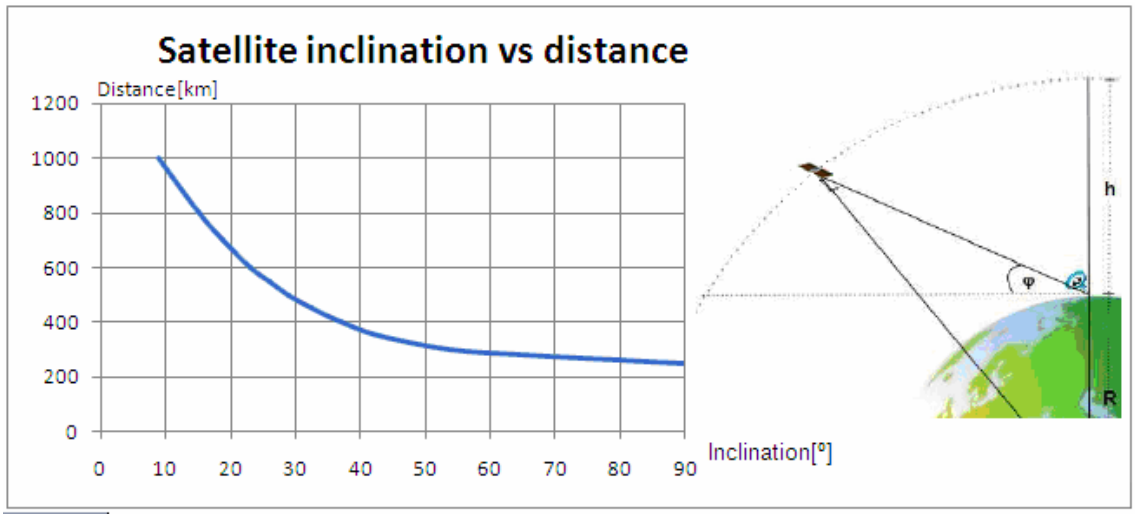
The camera has two lens options where the Field of View (FOV) is 57.4º and 8.0º for the TCM8230 and MC5970 respectively. The inclination of the pictures are valid above 45º when the distance to the interest point is 339 km while for a inclination of 60º the distance is 280 km. An orthophoto is when the angle is 90º that corresponds the minimum distance of 250 km but this situation is difficult that happens.

Map resolution for a given distance (Satellite inclination) and lens options

TCM8230		H: 698			
		V: 502		Area	
h [km]	incl[º]	FOV [º]	pix[m]	H[km]	V[km]
250	90.0	57.4	344	240	173
300	54.0	57.4	413	288	207
350	43.0	57.4	482	336	242
400	37.0	57.4	550	384	276
450	32.5	57.4	619	432	311
500	29.0	57.4	688	480	345
550	26.0	57.4	757	528	380
600	23.1	57.4	826	576	414
700	18.8	57.4	963	672	484
800	15.0	57.4	1101	768	553
1000	8.9	57.4	1376	960	691

MC5970		H: 698			
		V: 502		Area	
h [km]	incl[º]	FOV [º]	pix[m]	H[km]	V[km]
250	90.0	8.0	50	35	25
300	54.0	8.0	60	42	30
350	43.0	8.0	70	49	35
400	37.0	8.0	80	56	40
450	32.5	8.0	90	63	45
500	29.0	8.0	100	70	50
550	26.0	8.0	110	77	55
600	23.1	8.0	120	84	60
700	18.8	8.0	140	98	70
800	15.0	8.0	160	112	80
1000	8.9	8.0	200	140	100

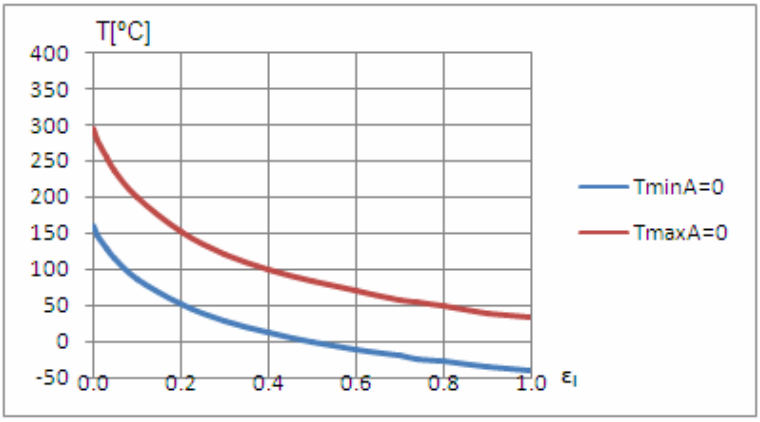
There is an ideal relationship between the satellite inclination and the distance to the interest point for an ideal track that crosses over the interest point. This function was obtained empirically from the simulations and is presented in the following curve.



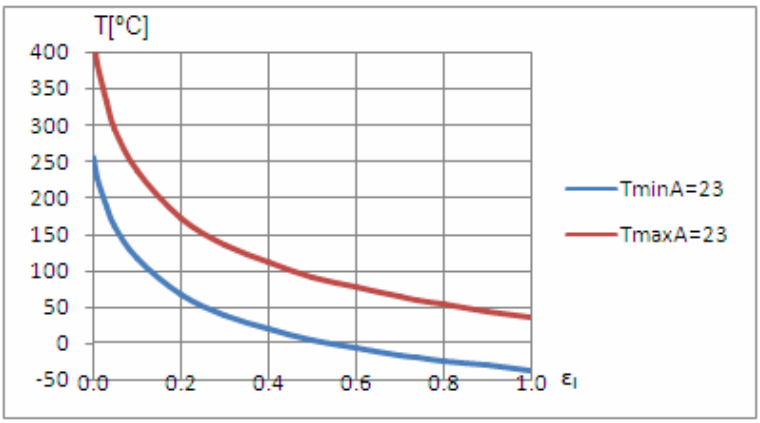
Satellite inclination function vs distance to the interest point

A.4 Emissivity calculations for the passive thermal control system

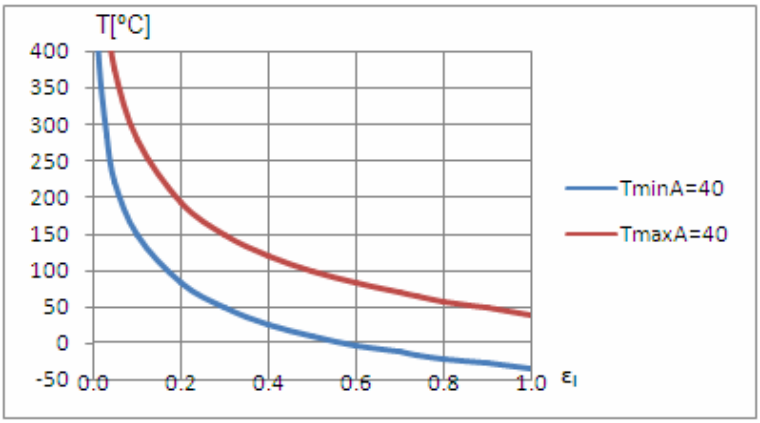
From the EXCEL sheet temperature ranges are calculate for every value of the telephoto lens emissivity paint $\bar{\varepsilon}_I$ combined by the *A* parameter of the copper layer exposed to the Sun. When *A* = 40 mm the satellite is much cooler than when *A* = 0 mm. The optimal is when the *A* parameter is 23 mm as presented in the following curves.



Temperature range vs photo lens emissivity εI and parameter A = 0 mm

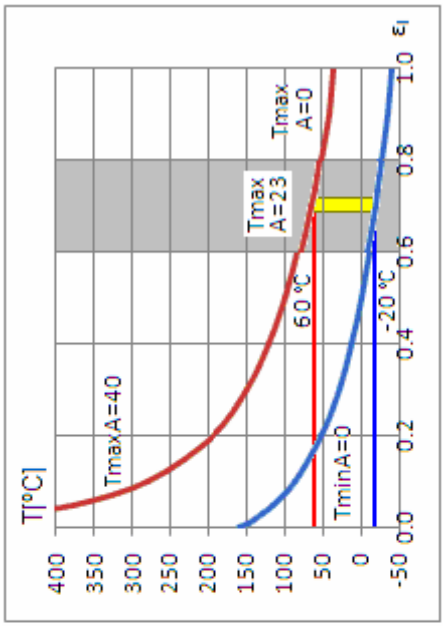


Temperature range vs photo lens emissivity εI and parameter A = 23 mm (Optimal)



Temperature range vs photo lens emissivity εI and parameter A = 40 mm

Numerical values are presented as well in the following table with a graph with the three ranges at the time marking the valid temperature range. The gray band is when the *A* parameter is 23 mm. The hot limit is 60 ºC marked in red and the cool limit is -20 ºC marked in blue. The valid zone for our design is marked as a yellow area. The telephoto lens emissivity paint is narrow and should be selected accurately.



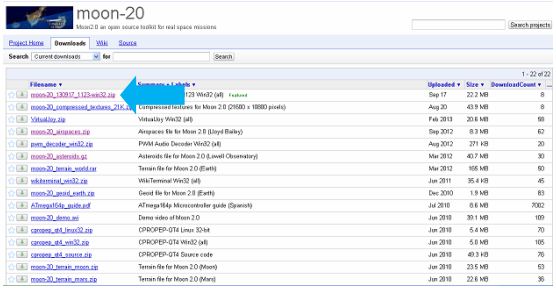
L	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000
W	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000	60.000
H	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000	5.000
A	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
B	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000	40.000
I	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000	70.000
J	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400	21.400
TminA=0	158.7	147.8	129.4	114.5	86.5	50.8	27.8	11.2	-1.6	-12.0	-20.7	-23.8	-28.1	-34.5	-40.1						
TmaxA=0	293.2	278.9	254.8	235.2	198.4	151.6	121.5	99.8	82.9	69.3	58.0	53.9	48.3	39.8	32.4						
Tdelta0	134.5	131.1	125.4	120.7	112.0	100.9	93.7	88.6	84.6	81.3	78.6	77.7	76.3	74.3	72.6						
ε camera	0.000	0.010	0.030	0.050	0.100	0.200	0.300	0.400	0.500	0.600	0.700	0.740	0.800	0.900	1.000						
TminA=23	255.2	227.3	187.8	160.2	115.4	66.7	38.5	19.1	4.6	-7.0	-16.4	-19.8	-24.4	-31.3	-37.3						
TmaxA=23	419.7	383.2	331.4	295.1	236.4	172.6	135.6	110.1	91.1	76.0	63.5	59.1	53.0	44.0	36.1						
Tdelta23	164.5	155.9	143.6	135.0	121.0	105.8	97.1	91.0	86.5	82.9	80.0	78.9	77.5	75.3	73.4						
TminA=40	709.4	426.3	283.9	222.3	146.9	81.5	47.8	25.8	9.7	-2.9	-13.0	-16.6	-21.5	-28.8	-35.1						
TmaxA=40	1015.4	644.1	457.3	376.6	277.7	192.0	147.8	118.8	97.7	81.3	68.0	63.3	56.8	47.3	39.0						
Tdelta40	306.0	217.8	173.5	154.3	130.8	110.5	100.0	93.1	88.1	84.2	81.0	79.9	78.4	76.1	74.1						

TUTORIALS **NOTAM for a stratospheric balloon**

This is the tutorial to generate a report to request the publication of a NOTAM (Notice To AirMen) to the Airworthiness Authority (AESA in Spain) as a result of an air activity, in our case a "non manned light free balloon" launch.

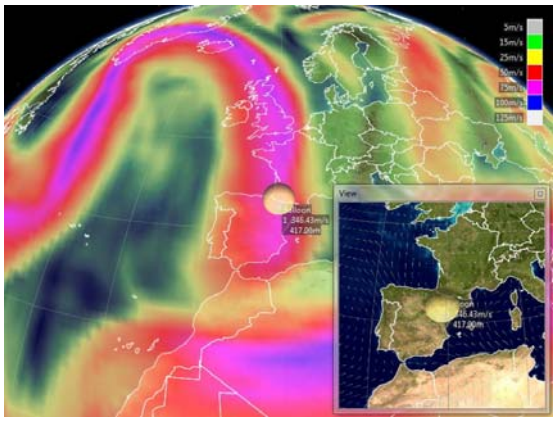
The report has to be sending by e-mail at least one month before the launch day to guarantee the necessary time to establish the contact between the Airworthiness Authority and both: The Airways Operator and The Military Staff. In case of necessity, some of the agencies can request extra information or directly notify the incompatibility of the launch, because of the air traffic or military maneuvers. The e-mail has to include the name of the organization responsible of the balloon launch and the person responsible of the organization (that has not to be the same of the launch) with his contact data. In case of civil responsibility, the contact with each person will be established. Civil Aviation indications breach may result in the prohibition of future launches requested by the organization of the air activity. Also we have to attach to the report: A study of winds before the launch and A compatibility study of the balloon trajectory with airways. Thus, we recommend the use of the tool called Moon2.0, which can be free downloaded here:

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



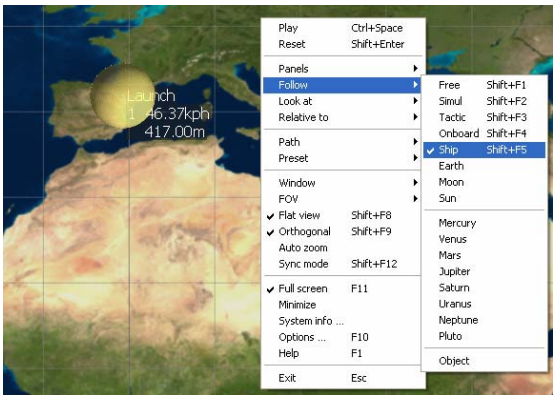
The last version has directly updated the balloon trajectory, which did not happen in old versions of Moon2.0. Another advance of this version is that the airspaces and airways are updated too, so we only have to activate the particular option to see that, like we explain later. But the most important advance is the tool that allows to know the exactly moment when the balloon enters and leave an aerospace or an airway; that allows us to know for how long the balloon is inside it.

The study of winds before the launch is as follows. The Moon2.0 has a wind tool as was explained in the tutorial *Balloon trajectory*.



The compatibility study of the balloon trajectory with airways is as follows. To see the airways, first we will show the top view and for that we have to press the right button of the mouse and select the *Flat view* option, like the next figure shows.

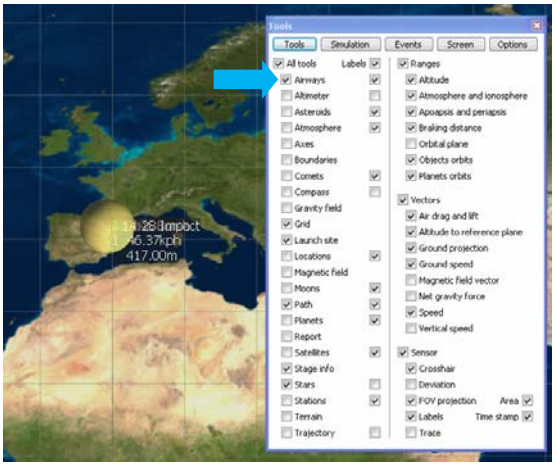
We recommend activate the tracing of the aircraft *(Follow ship)*, which we can found pressing the right button in the mouse and selecting the option showed in the next photo.



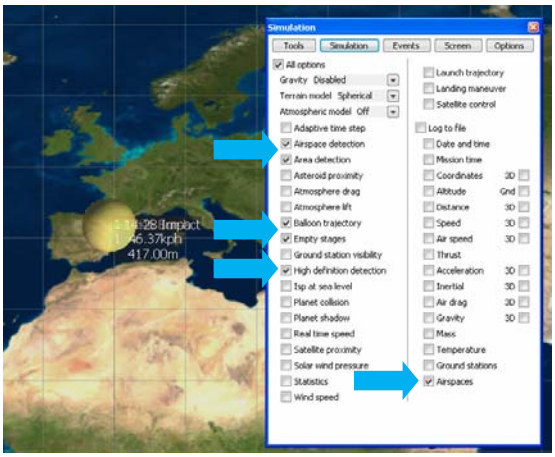
With the wheel of the mouse we can increase or decrease the zoom, which allows see better the interest areas.

After that, we will explain the steps to show the airways and airspaces and to select the option of airspace detection, which show us when the balloon enters and leaves a particular airspace.

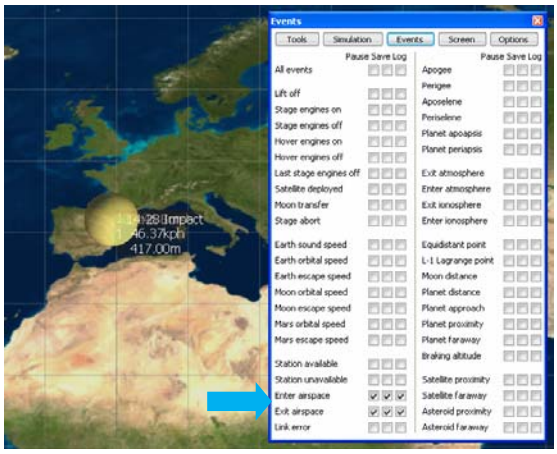
First of all, we have to press the button **Options** *(Viewer options)* of the tools menu and in the tab **Tools** we will select the check box option called *Airways*; like the next photo shows.



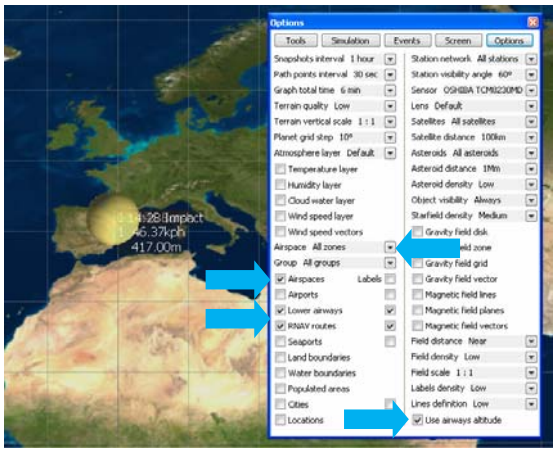
The next step is to click in the tab called **Simulation** and select the following check box options as next photo shows: *Airspace detection, Area detection, Balloon trajectory, Empty stages, High definition detection* and *Airspaces.*



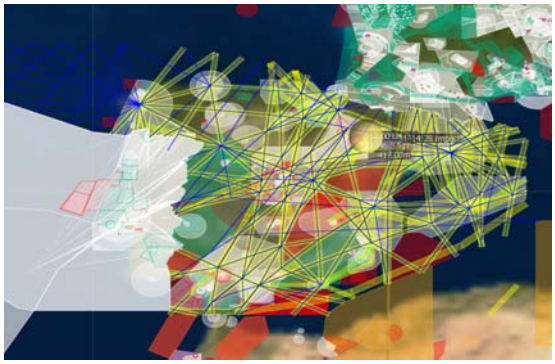
After that, we have to press in the **Events** tab and select the following check box options: *Enter airspace* and *Exist airspace.* In this tab, we can observe that there are three of for which case; the different check box option in front of the other tabs the **Pause**, which serves to stop the image when this event occurs and this allows us see better the characteristics of this event.



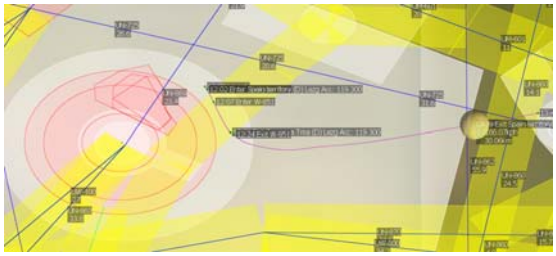
The last step is to press in the **Options** tab and here we will select the next options: *Airspaces, Lower airways, RNAV routes* and *Use airways altitude*, like the next photo shows. If we do not need the airspaces of all the zones, because our balloon does not cross it, we have the option of choose the airspaces that we need; this option is in the list called **Airspaces**, where we will select the zone that we want and this will appear in the place where now there is the name: *All zones*. Next photo shows this process.



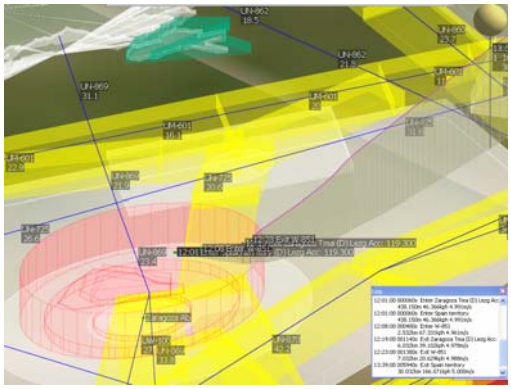
Observe that the Labels of *Airspaces* are not selected otherwise so many names impede a proper vision of the situation. After these steps, we can observe the airspaces and the airways like this:



Increasing the zoom, we observe in more detail airspaces and airways. This is one of the most important photos for the report because the controller will see the possible conflict with real traffic. The photo shows the time to airspace clearance, in this case 23 minutes after the balloon release.



The next photo shows and isometric view that shows the altitude of airspaces and airways. In this example the balloon, starting from the Zaragoza TMA, crosses the W-851 airway in 23 minutes and leave the Spanish territory in 1 hour and 36 minutes when 30 km of altitude is reached. At 32 km the rocket will be ignited autonomously. Apart from this, we can observe the information given by the *Log* panel, in which are detailed the moment when the balloon enters and leave each airspace and airway.



The report has to be send to:

AESA. Servicio de Trabajos Aéreos y Aviación Deportiva

aviaciondeport.aesa@fomento.es

Our NOTAM will look similar to the following photo. In it we can observe the characteristic data of the launch. This information will be read by all the pilots affected before the starting of the flight.

Aena *Aeropuertos Españoles y Navegación Aérea*

Origen/Origin DIRECCION DE OPERACIONES ATM / CAT

De/From COP Fax Tfno.

Fecha/Date 19 de noviembre de 2011 Nº de Hojas 2

Destino/To

Empresa/Company AESA / TRABAJOS AÉREOS Y AVIACIÓN DEPORTIVA

A la atención de/To the attention of

Mensaje/Message

REF: 1608 **CORRECTA VERSION**

ASUNTO: SONDEOS ZUERA (ZARAGOZA)

En contestación a su solicitud del día 19-09/2011, se informa que como resultado del estudio realizado para la ejecución de la actividad solicitada, se ha procedido a su autorización y publicación NOTAM correspondiente, del cual remitimos copia.

- NOTAM:

(D1599/11 NOTAMN
Q)LECM/QWLLW/IV/M /W /000/999/4152N00045W27
A)LECM B)1109191000 C)1109191330
E)RADIOSOUNDING. ASCENT OF FREE STRATOSPHERIC METEOROLOGICAL LIGHT BALLOONS ON 415132N 0004511W
ZARAGOZA/ZUERA

BALLOONS FEATURES
TYPE: SPHERICAL
COLOUR: WHITE
DIAMETER: 2M
WEIGHT: APROX. 2000GR INCLUDING SOUNDING
MAXIMUN ALTITUDE: APROX. 100KM AGL
MAXIMUN DEVIATION: APROX. 50KM
F)SFC G)UNL
)

A PETICION DE ZARAGOZA TACC ANTES DEL LANZAMIENTO DEBERÁ CONTACTAR CON DICHA DEPENDENCIA PARA OBTENER LA APROBACIÓN EN EL TELÉFONO 976 337 XXX.

Telefax

The regulation of non manned free balloons use is in the Regulation of air circulation of the Royal Decree 57/2002. The appendix S contains all the information about this type of activities. It is recommendable the reading of it before preparing a mission to know what we can do and what not.

References:

http://www.mtc.gob.pe/portal/transportes/aereo/regulaciones/docs/rap_rev15/rap101/rap_101_subparte_d_rev15.PDF

http://www.boe.es/boe/dias/2002/01/19/pdfs/C00001-00697.pdf (Spanish)

http://www.boe.es/boe/dias/1963/04/22/pdfs/A06766-06767.pdf (Spanish)

http://www.icao.int/Documents/annexes_booklet.pdf