POLITECNICO DI TORINO

DOCTORATE SCHOOL

PhD in Electronic Engineering XXIII cycle

PhD Thesis

Development of a low cost Sun sensor and star tracker for Aramis (Modular nanosatellite platform)

Jairo Alberto Hurtado

Student ID 152779

Supervisor	**Coordinator of the PhD Program**
Prof. Leonardo M. Reyneri	Prof. Ivo Montrosset

February 2012

Acknowledgements

I want to say thanks to all the people to support me in this process, however, there is not enough space to write them one by one. So I am grateful for all the help that Leonardo (my advisor and my thesis supervisor) gave to me. Also I want to thank to my parents, my family and my friends here and further.

Contents

List of Figures

Chapter 1

1. Introduction

The use of artificial satellites is widely spread in our lives; they are used in various fields such as, astronomy, atmospheric studies, communications, navigation, reconnaissance, remote sensing, search and rescue, space exploration, and weather. All of them provide a great deal of information in the sciences of earth and life.

Usually, a normal sized satellite weighs more than 500kg, it is a few meters long and can cost more than 1million Euro to build and launch it.[1].

In later years, small satellites (MicroSat, NanoSat, CubeSat and PicoSat) have been present as the beginning of the space age.

Nowadays, due to advances in microelectronics (especially in microprocessors and the lower cost on the launching, small LEO (Low Earth Orbit) satellites are an attractive alternative feasible instead of traditional big GEO (Geostationary Earth Orbit) satellites.

Small satellites are cheaper to construct and launch, they can do several specific tasks even better than large satellites, aside from the small size of the satellite. Furthermore, all onboard components are energy thrifty, and their entire surface is used to collect energy from the sun through solar panels.

All these advantages provided for small satellites compared with big satellites can be summarized with the slogan "faster, smaller and cheaper".

The difference between MicroSat, NanoSat and Pico Sat is basically the size of each one. CubeSat is a a type of miniaturized satellite for space research with small volume and mass (less than 1 liter and 1,33 kg). NanoSat is a satellite slightly bigger than a CubeSat, with a mass between 1 and 10 kg [2].

One of the most important parts in a satellite is the Attitude Determination and Control Subsystem (ADCS). It consists of a set of attitude sensors, actuators and a microcontroller with control algorithms.

To determine attitude, several sensors can be used: Sun sensor, Horizon scanner, magnetometer, Star tracker, and Gyroscope. Some of these can be expensive, others must work together because acting as standalone devices they cannot obtain adequate/reliable results, and others are too complex or too big for small satellites.

Sun sensor is one of the most common parts in an ADCS. It determines the satellite's orientation relatively to the sun by measuring the amount of light that hits it. It also can be used for instrument pointing, solar array pointing, and safe mode or sun acquisition systems. There are several types of Sun Sensors, analog or digital, one or two-dimensional position, all of them, with different specifications and cost. In this project, a CMOS sensor based is used as Sun sensor.

1.1. Design Goals and Proposed Solutions

The main goal of this thesis is to estimate the attitude of a small satellite using only image sensors in a low cost satellite.

The main objective is to process the images coming from the image sensor, which is designed with a low cost Sun sensor that can be used also as a star tracker to get the attitude of a small satellite (modular architecture, type ARAMIS) without use any other attitude sensor.

There are two possible ways to solve the problem of the attitude in the satellite using a sun sensor; the first one is based in the knowledge of the orbit parameters, along with the measurements from the magnetometer to estimate the rotation angles. This is the classical solution.

The second approach is presented in this thesis, using only image processing without knowing the orbit parameters or using any other sensor.

Another goal of this thesis is to design and evaluate a real sensor structure for nano satellites using CMOS image sensor, as a attitude sensor and to evaluate the possibility to estimate the position of a small satellite using only image sensors.

Such a direct approach can allow for implementations of single sensors to estimate the attitude, and to reduce cost or space in the satellites.

1.2. Main Contributions

The first period of the PhD program was invested in the study of the literature and state-of-the-art on the field of Sun sensors for nanoSat, complemented with the studies of the physical implantation CMOS sensors and design of the Sun sensor for the satellite. This was an essential part, in order to ensure a proper knowledge base before starting working towards the aforementioned goals.

The following step was to dedicate the time to design, implementation of the circuit and board of the Sun sensor, as a part of the modules in Aramis.

Once the schematics circuit was designed, it was changed and redesigned because the sensor provider (Kodak) in its bankruptcy process cancelled the production of new sensors; consequently, another sensor from a different provider had to be selected.

The work with the circuit and the sensor consisted in capture images from the Sun to calibrate the sensor and to adjust the parameters when neutral filters were used.

The last part of this thesis was done at Space System Laboratory at MIT (Boston, MA. USA), where the mathematical analysis and captured image simulator were developed to estimate the attitude of the satellite based in the measurements with the real Sun sensor (image sensor).

1.3. Contents Organization

This thesis is practically divided into four parts:

Part I is introductory and gives an overview of the contents of this work.

Part II is a description of the models here used and parameters to design the physical assembly of the Sun sensor.

Part III describes the process to take the images by the sun sensor and the work performed in order to improve the quality of the images to have real parameters in the image simulator.

Part IV presents the work done in the simulator, parameters, calculations, errors, and image processing.

The main emphasis is given to solve the problem of the attitude in the satellite. Finally, this work concludes by summarizing the results and suggesting future extensions of this work in the last chapter.

Chapter 2

2.1. What is Aramis

Aramis is the project name assigned to develop modular architecture for cubeSats, it is done in the Electronics Department of the Politecnico di Torino (Turin, Italy). It is motivated to have better use through modularity, knowing that there are common subsystems between cubeSats, for instance, attitude determination and control, telecommunication, power conversion, and management. All this modularity is reached by the use of tiles that contain the subsystems and they are self-bearing the structure.

2.2. Limitations and characteristics

As the sensor must be used in CubeSat, it has some limitations in cost, size, and also it must work in outer space conditions (wide temperature ranges and radiation particles exposure) and using commercial components as much as possible.

Additionally, the sensor will be used in a satellite with modular architecture, which gives size limitations. Furthermore the camera design is not like a normal one this particular camera will aim directly to the Sun constantly and damages in the camera itself because of this exposure are not allowed.

Sun sensor should cover the entire space (at least 90° by face, when a cube shape is used) with maximum error less than 0,01 radians.

2.3. Choosing the sensor

Initially there were two options to choose the type of sensor: PSD (Position-Sensitive Detector) or CMOS. With them, three manufacturers were evaluated to choose the type of sensor, Hamamatsu for PSD, Aptina and Kodak for CMOS.

By comparing PSD and CMOS, it was determined that CMOS has more versatility to use in other applications, so finally CMOS sensor was selected.

Manufactures are very protective of their products and, in the same sense, they are very careful not to give more information than basic specifications. Aptina did not provide programming information about their sensors. On the other hand, Kodak sensor user's and configuration manuals were available; so Kodak was selected.

All the initial design was based, developed, and created on Kodak sensor. Unfortunately, at some point during the testing phase, it came to our attention that Kodak stopped manufacturing these sensors and had abandoned the sensor market. As a consequence, the circuit had to be redesigned entirely based on the Aptina sensor despite the fact that information and data sheet were restricted to the public with low quantity of units to buy.

2.4. Geometric Description

The sensor works with the sunlight, when the sensor faces the light, it goes through a small hole and impacts on the sensor as is shown in Figure 1

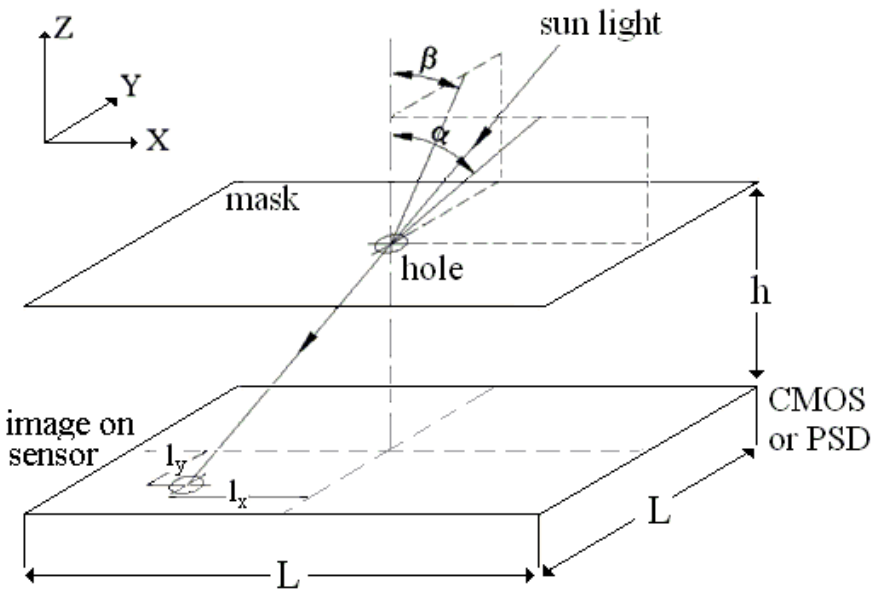

Figure 1. Sun sensor scheme

With the position of the spot beam on the sensor is possible calculating *X* and *Y* angles.

2.5. Physical design of the camera (pin-hole camera)

The criteria used for the first approximation to decide the sensor was the physical design for the optical part. Taking the physical characteristics of each sensor, in terms of size of the active area of interest (Hamamatsu: 9 x 9 mm; Kodak: 4,86 x 3,66 mm; Aptina: 4,51 x 2,88 mm), and the minimum angle of view (90 degrees), then the maximum distance between the pin-hole and the sensor is obtained:

4,5 mm for Hamamatsu, 1,8 mm for the Kodak and 1,35 mm for Aptina.

With the calculated distance, an optical analysis can be done to obtain the hole's size. An optimal value of the pin-hole diameter can be obtained using the Lord Rayleigh equation.

$$d = 1{,}9\sqrt{h \cdot \lambda}$$ Equation 1

Where ***d*** is pin-hole diameter, ***h*** is focus length (distance from hole to sensor) and λ is the wavelength of light (960 nm).

The obtained result shows that an optimal pin-hole ranges between 74 µm y 125 µm

After checking the characteristics of the pin-hole providers, the diameters available were 50 µm ± 5 µm, 100 µm ± 5 µm and 200 µm ± 5 µm, in (Edmund Optics). So the pin-hole diameter chosen finally was d =100 µm.

Pin-hole diameter affects directly the diffraction of the light that pass through it.

2.6. **Diffraction**

Diffraction pattern can be calculated with the Equation 2,

$$I(\beta) = I_0\left[sinc\left(\frac{\pi d}{2\lambda}\sin(\beta)\right)\right]^2$$ Equation 2

Where I_0= light intensity, d = pin-hole diameter, λ = wavelength and β = incidence angle. Thus, taking the values defined with the sensor (d=*3 mm of distance,* pin-hole diameter =*100 µm* and *λ = 960 nm*) and replacing in the equation normalized, diffraction pattern is shown in Figure 2

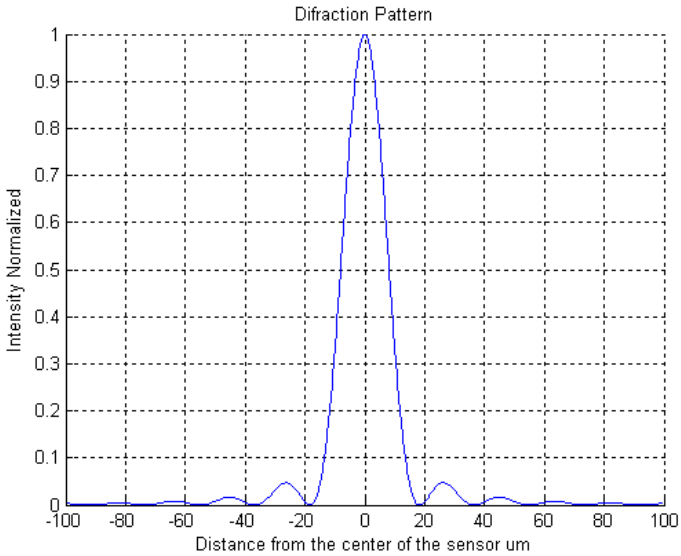

Figure 2. Diffraction Pattern vs incidence angle (β)

2.7. How a CMOS works?

A CMOS, is an array of light-sensitive semiconductors. In our case, the MT9V011P sensor (by Aptina) is made as an array of 480 rows by 640 columns.

At the beginning of the integration time the control circuit will reset every pixel on a row. All pixels in the same row are simultaneously reset, but not all pixels in the array. Then, the pixels on the row are read them in a serial way, as is shown in Figure 3.

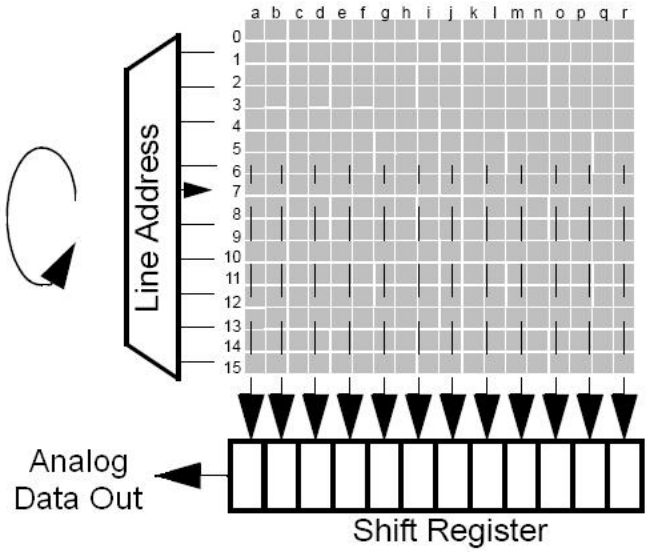

Figure 3. CMOS sensor reading

For better understanding about how this sensor works, it can be compared with an array of caps under the rain. Each row is emptied before starting integration time, rather than at the end of integration time, each cap on the row is “read it” in its level. Level of water is analog to intensity of light. The process is shown in Figure 4.

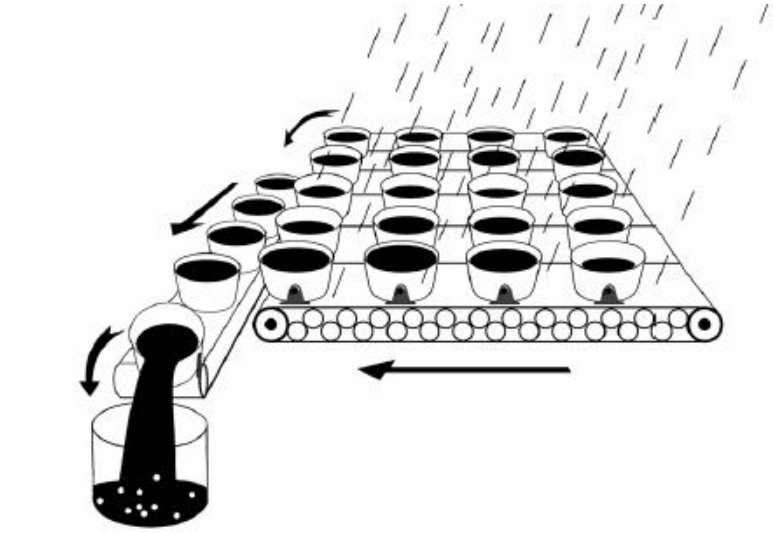

Figure 4. Analogy between CMOS sensor and cups array.

In the sensor, the value of intensity is given in 10 bits

2.8. Calculations of the solar illumination on the sensor

Equation 3 calculates time saturation in the sensor due to the light coming and knowing that the sensor can register up to 30 frames per second, we can estimate if optical filters must be used to avoid damages or saturation values in the measurement.

$$Time\ Sat = \frac{V_{ADC}}{Sensor\ Sensitivity \cdot Ilum\ @\ earth} \qquad \text{Equation } \mathbf{3}$$

Taking the maximum illumination given towards the Sun (130000 lux) and knowing the sensor parameters, sensitivity (1,9 V/lux) and voltage reference (2,8 V), the integration time is calculated in Equation 4:

$$Time\ Sat = \frac{2{,}8\ V}{1{,}9\ V/lux \cdot s \times 130000\ lux} = 11{,}33\ \mu s \qquad \text{Equation 4}$$

Using this value, we can calculate the minimum attenuation factor to avoid saturation in the sensor with Equation 5.

$$Attenuation\ Factor = \frac{\frac{1s}{30\ frames/s}}{11{,}33\ \mu s} \approx 2950 \qquad \text{Equation 5}$$

Hence, the filters used should have an index of at least 2950 of attenuation factor. This is reachable by using an array of Neutral Density filters.

2.9. Neutral Density Filters

To calculate the attenuation given by the filter, we have to apply logarithm to find the Optical Density (OD), obtained the Equation 6:

$$OD = \log_{10} 2950 \approx 3{,}47 \qquad \text{Equation 6}$$

OD required is 3,47, therefore a value of 3,5 will be used; a combination of two filters is required in order to obtain it (0.5 + 3.0, or 1.0 +2.5 or 1.5 + 2.0).

There are two kinds of neutral density filters, reflective or absorptive. Both of them could be used, although absorptive filter can suffer from heat build up and reflective filter must be used carefully to avoid injuries in the eyes when the tests on earth will be done.

2.10. Physical Assembly

Physical design of the sensor mounted should be on the inside one side of the satellite (between pin hole and sensor should be the filters), but there is not enough space for the neutral density filters between the pinhole and the sensor, as is shown in Figure 5.

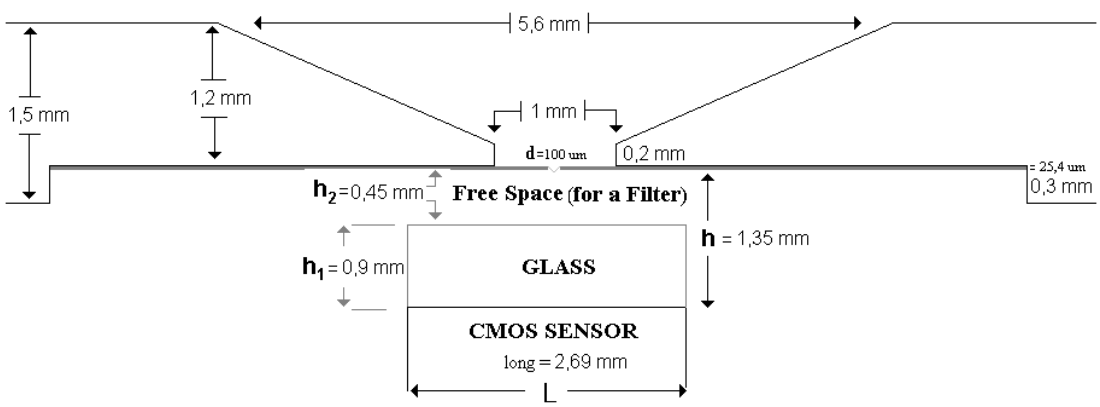

Figure 5. Physical Mounting of the sensor in the satellite

2.11. Physical Mounting Design

As stated previously, the filter should be located outside the assembly; therefore, the filters are located before the hole that allows sunlight to pass through to the sensor as is shown in Figure 6.

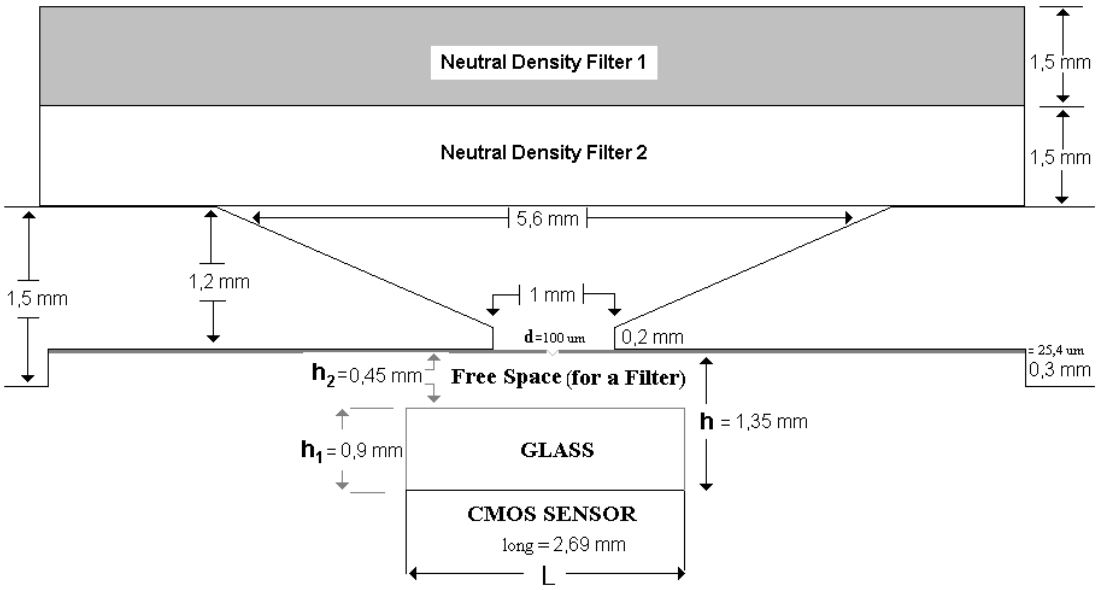

Figure 6. Filters settled outside the face of the satellite

2.12. Physical Mounting Design

Pinhole used was taken from Edmund Optics, with 100 µm diameter. It is shown in

Figure 7, with the physical assembly.

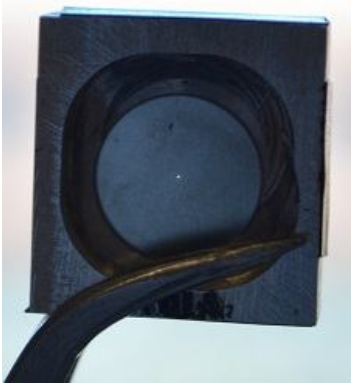

Figure 7. Pin hole with 100 um diameter

To put into perspective the size of the assembly and its components, see Figure 8, a ruler is set to observe the real size of the pinhole cover, the pinhole support and the neutral density filters.

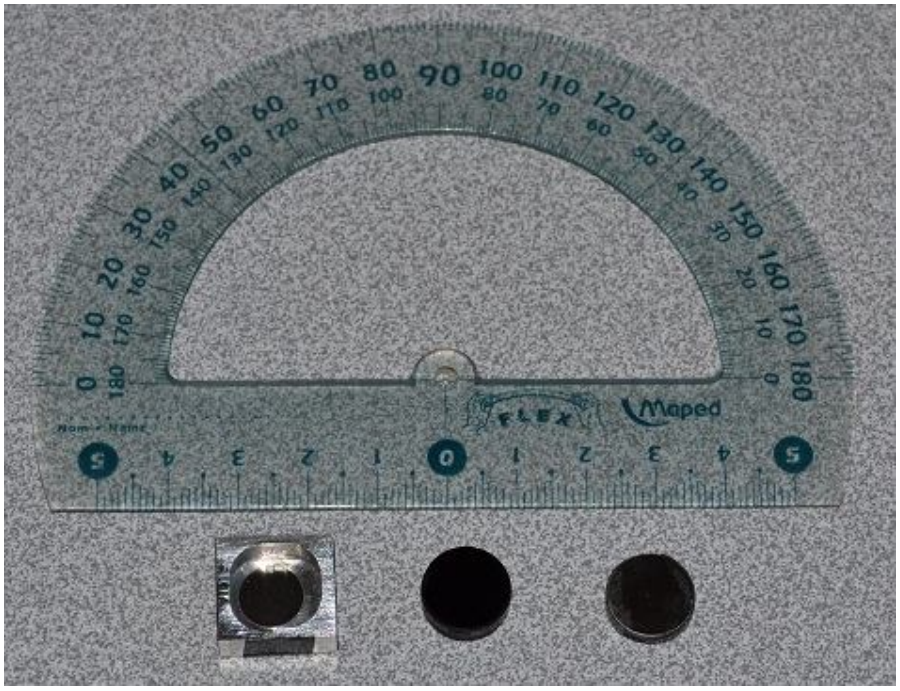

Figure 8. Pin hole support and neutral density filters

Once the physical part of the sensor is defined, we proceed to design the circuit for testing and to capture respective images.

2.13. Schematic design of sensor test circuit

Test circuit starts with the CMOS sensor, connected to an MSP430 microcontroller (which can be used in the outer space applications). MSP receives the command configuration given by the PC and retransmits it to the CMOS sensor. The images taken by the sensor are stored in the SRAM. Later these data are read by the FPGA (Field Programmable Gate Array) board which sends the data to a screen and be able visualize the pictures taken. Schematic circuit is shown in Figure 9.

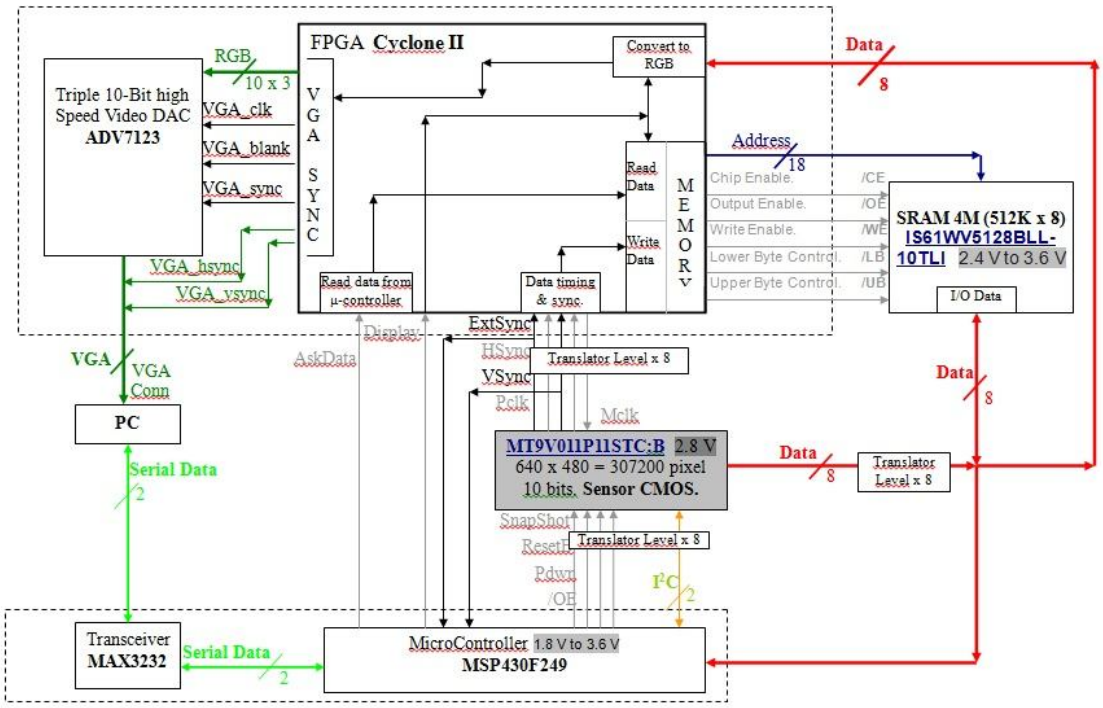

Figure 9. Test circuit schematic

And the test circuit board is shown in Figure 10

Figure 10. Test circuit board

Chapter 3

3.1. Images taken with the sun sensor

Using the test circuit board, several images from the Sun were taken with different kinds of physical configurations (filter sort, filter attenuation) and parameters (exposure time, time integration). All these tests were done to evaluate the systems and our concerns about: image saturation, unwanted reflections in the image, physical installation with the external filters, and choosing threshold level.

3.2. Images taken with different filters

Images were taken with different order in the filters, where the light attenuation is evident, for instance, the image of the Sun in Figure 11 was taken with a filter with OD (Optical Density) equal to 3,0.

Figure 11. Using a reflective neutral density filter with OD = 3,0

In Figure 12, the image was taken with a filter with OD equal to 4,5, the size of the Sun looks smaller than the one in Figure 11, and even the reflections (unwanted lights) are not visible.

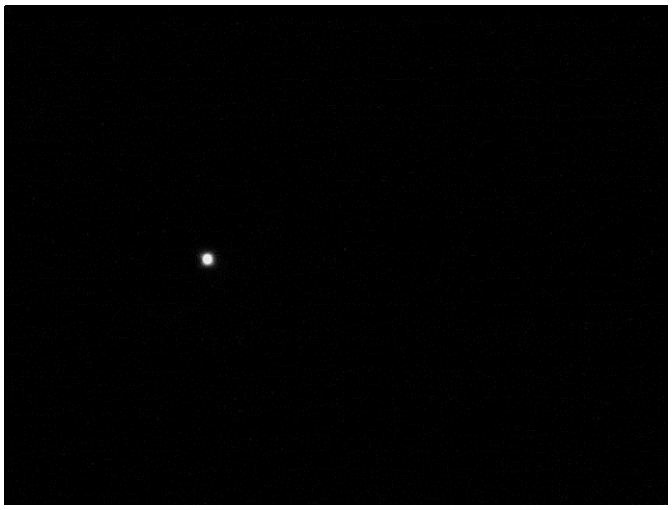

Figure 12. Using a reflective neutral density filter with OD = 3,0 + non reflective neutral density filter with OD =1,5. Total OD 4,5

3.3. Unwanted lights

In the image of the Figure 13, some residual lights are visible. These lights are unwanted because the center of mass in the image will be different to the center of mass of the Sun.

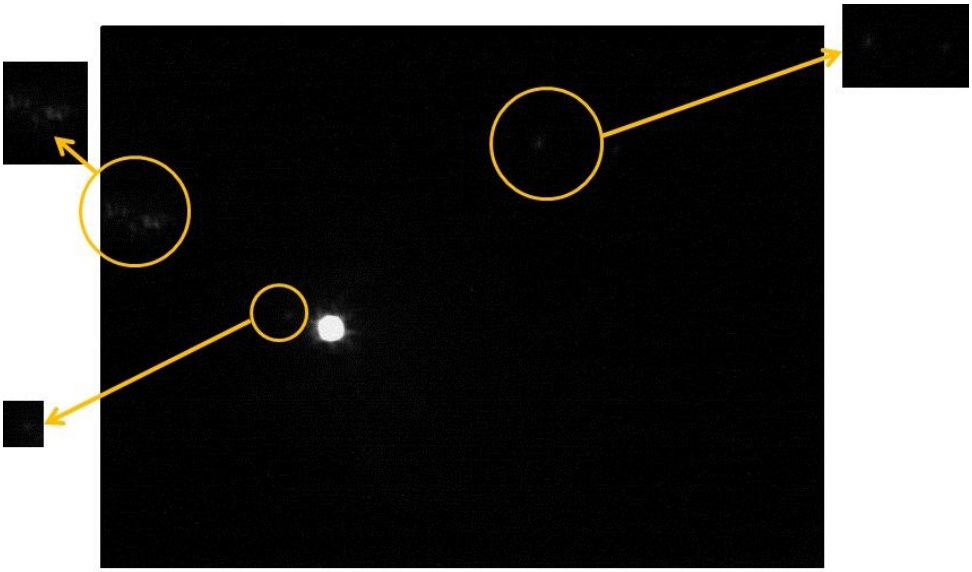

Figure 13. Light reflections on the sensor

In Figure 14, these same lights are shown with different balance in bright and contrast in the image in order to observe them more easily.

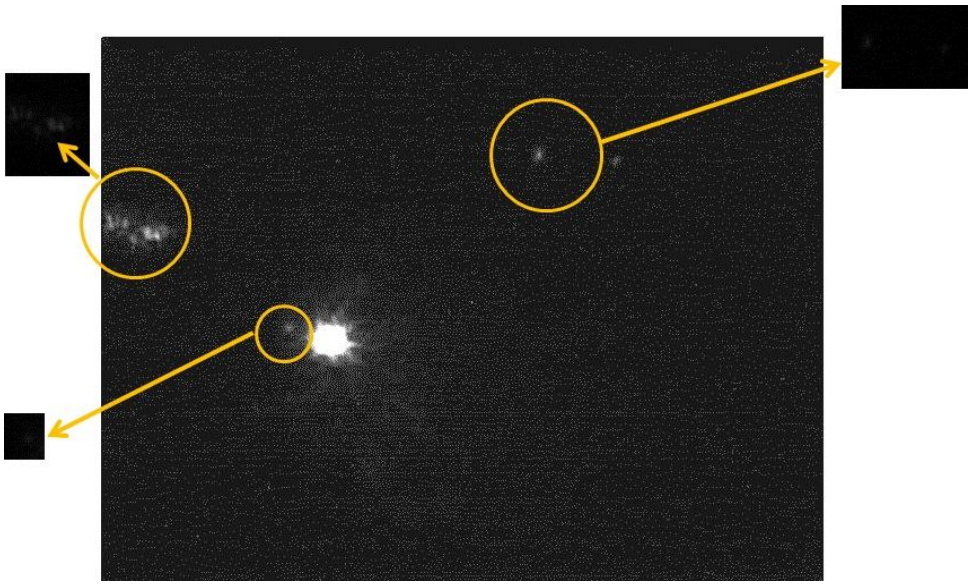

Figure 14. Light reflections on the sensor with contrast and bright adjusted

To reduce the unwanted lights in the image, the mechanical parts near the pin hole were painted in color black, improving the quality of the images. Nonetheless, it was not good enough to reduce the reflections. So, the decision taken was to rise the optical density of the filter further to 4,0.

Earth albedo and Moonlight could be unwanted lights in the image, but these are reduced with the neutral density filters, without extra attenuation.

After the attenuation given by the filters, some reflections remain; however, these were considered to define the threshold to eliminate data before calculations.

3.4. Threshold Analysis

After several images were unsuccessfully tested trying to reduce the reflections, the threshold applied to the image must be decided to eliminate the residual unwanted lights.

Threshold was defined by the percentage of the maximum value of the image; so these percentages were tested from 2 % up to 91 %.

Images with different threshold are show in Figure 15. (These images were modified in contrast and bright to have a better visualization)

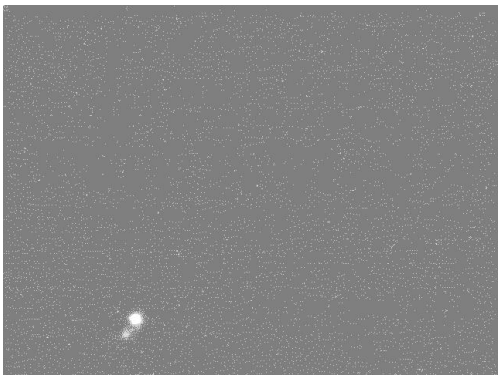

Image without threshold

Image with threshold = 25%

Image with threshold = 33%

Image with threshold = 50%

Figure 15. Different threshold applied to the images

Once the tests were done, the results indicate that the threshold must be higher than 28%, as it is possible appreciate in the Figure 15.

The results that support the threshold analysis are shown in Figure 16. (Horizontal axis is 1/percentage of maximum level)

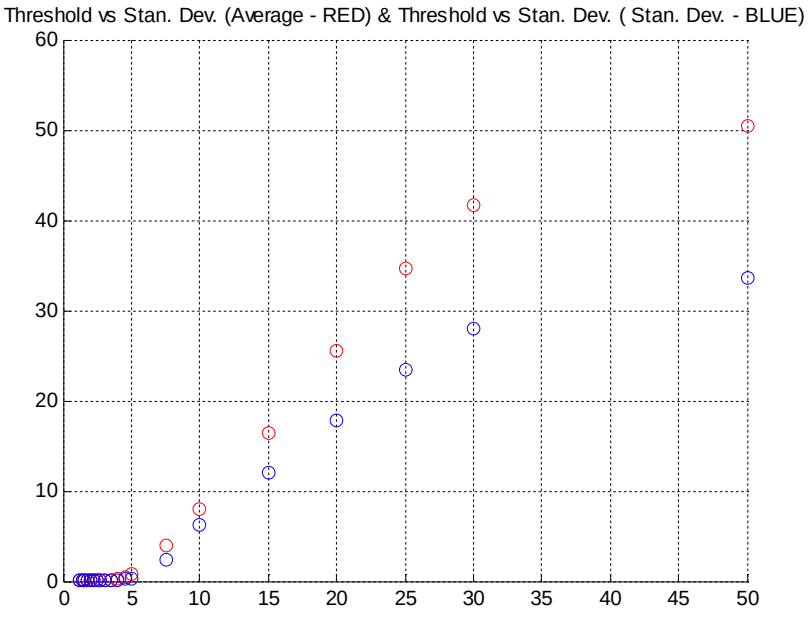

Figure 16. Example of image analysis with different thresholds

3.5. Center of mass

Once the threshold is applied, the center of mass of the Sun in the image can be calculated. There two values to calculate, one for rows and the other one for columns, as it is shown in Figure 17.

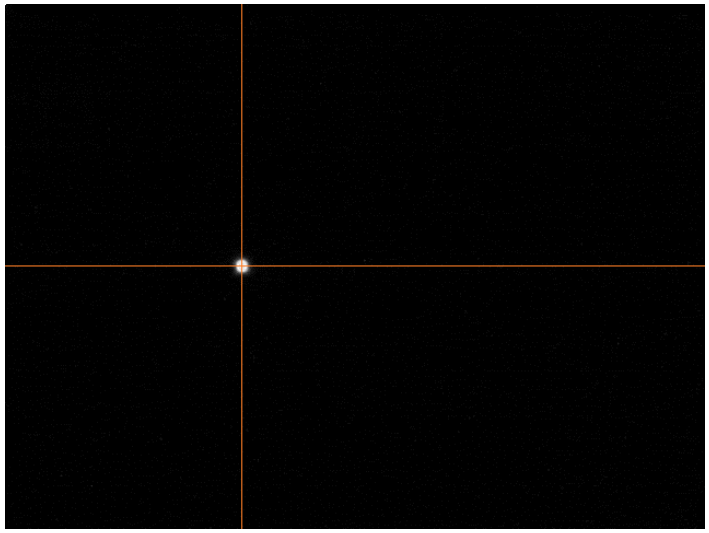

Figure 17. Center of mass in the image

With these two values the center of mass of the Sun in the image is defined. The equations to calculate the center of mass in rows are columns are respectively Equation 7 and Equation 8:

$$Center\ of\ mass\ (row) = \frac{\sum_{m=1}^{num\_col}\left[\sum_{n=1}^{num\_row} pixel(n,m)n\right]}{\sum_{m=1}^{num\_col}\left[\sum_{n=1}^{num\_row} pixel(n,m)\right]} \qquad \text{Equation } \mathbf{7}$$

$$Center\ of\ mass\ (column) = \frac{\sum_{m=1}^{num\_row}\left[\sum_{n=1}^{num\_col} pixel(n,m)m\right]}{\sum_{m=1}^{num\_row}\left[\sum_{n=1}^{num\_col} pixel(n,m)\right]} \qquad \text{Equation 8}$$

To observe more in detail the Sun, a zoom image in shown in Figure 18.

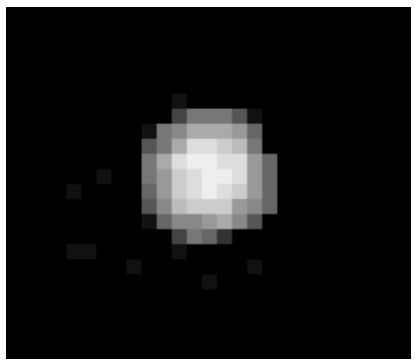

Figure 18. Image of the Sun

3.6. Difference between samples

Another parameter evaluated was the differences between images at different sample times.

The images shown in Figure 19 and Figure 20 have a 60 second difference between them. Center of mass is calculated in each one, with results shown below. Pixel (1,1) is located top left in the image.

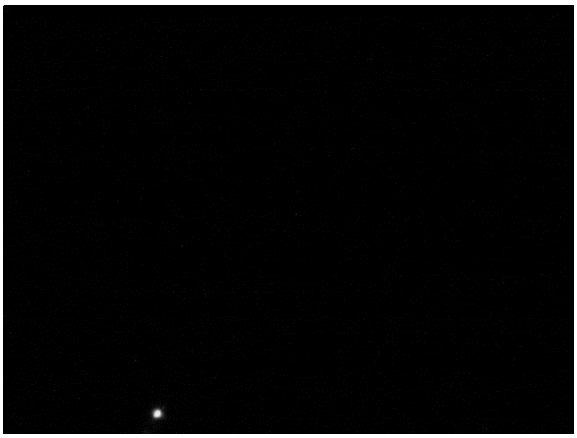

Figure 19. Image of the sun, with center of mass: (173.63, 457.83)

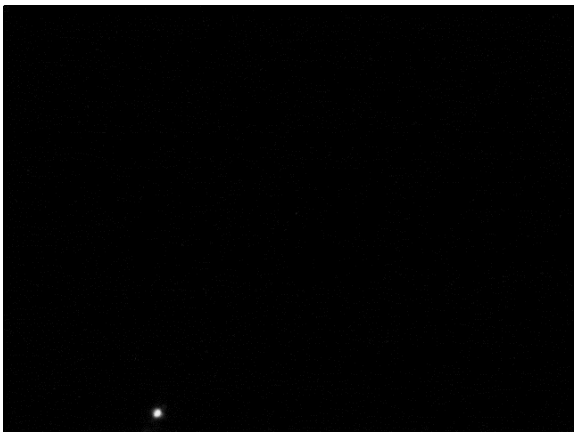

Figure 20. Image of the sun, with center of mass: (173.64, 459.52).

The difference between the centers of mass ((in pixels) in the images is:

Rows 173,64 – 173,63 = 0,01 pixels.

Columns 459,52 – 457,83 = 1,69 pixels.

It means that after one minute, the angle difference in columns direction is 0,22°. This result is more than satisfactory for the requirement about resolution. If the center of mass in the image is closer to the center of the image, this resolution is even bigger.

Also if we want to have a better resolution, we can do it by taking more samples in the time. CMOS sensor is programmable up to 30 frames per second.

Chapter 4

4.1. Introduction

This chapter is about the mean topic of the thesis, which is evaluating the possibility of estimating the attitude (the orientation of the satellite in space is called its attitude) in a cubeSat from images taken from it, using as references Sun, Moon and Earth.

The problem here described is limited only to static attitude determination, it means that time is not involve in the computations.

So, the first step is to describe the basic concepts and tools commonly used for attitude determination, specially attitude sensors and attitude determination algorithms.

Once the mathematical model is validated, it proceeds to develop a system to check the feasibility of the estimated attitude using only images. Since it is not easy to get these real images, because a satellite would have to be put into orbit, these images are generated by a simulator, which will evaluate the estimation algorithm.

4.2. Attitude determination

One of the modules that make up a CubeSat is the ADCS. The ADCS is divided essentially into three parts:

The first part is based on a set of sensors; they should have the ability to sense the current attitude of the platform.

The Second part consists in a set of mechanisms by which the attitude error can be controlled and corrected.

The Third and last part is composed by a microcontroller controlling the designed algorithms. They must have the ability to compute errors between current attitude and desired attitude.

Attitude is described at least by three attitude variables; furthermore, the mathematical analysis of attitude determination is complicated by the fact that attitude determination is necessarily either underdetermined or overdetermined.[3]

Attitude control and determination are necessary in many cases, for example, to prevent the satellite from tumbling in space and to ensure that antennas point in the proper directions, the same applies to on board cameras or earth-sensing instrumentation.

The satellite is exposed to several forces that change the orientation; for instance, initial force deployment by separating from the rocket launch and solar radiation drive constant orientation changes respect to the Earth. For these reasons it is required to estimate the attitude of the satellite to apply the appropriate correctives.

Attitude in a satellite can be determined using some of these sensors: Sun sensor, magnetometer, horizon scanner, star tracker, gyroscope [4].

4.3. Attitude sensors

There are two basic classes of attitude sensors. The first class makes absolute measurements; they are used in the static attitude determination algorithms.

Absolute sensor measurements are based on the fact that knowing the position of a spacecraft in its orbit makes it possible to compute the vector directions, with respect to an inertial frame, of certain astronomical objects, and of the force lines of the Earth's magnetic field. Among these, there are Sun sensors, horizon scanner, magnetometers, and GPS, for example.[3]

Absolute measurement sensors measure these directions with respect to a spacecraft or body-fixed reference frame, and by comparing the measurements with the known reference directions in an inertial reference frame, are able to determine (at least

approximately) the relative orientation of the body frame with respect to the inertial frame. [3]

The second class makes relative measurements. Relative measurement sensors belong to the class of gyroscopic instruments, including the rate gyro and the integrating gyro.[3]

4.4. Sun sensor [5][6]

If a vector pointing towards the Sun could be determined, this would aid the computing of the satellite attitude. The Sun is a useful reference direction because of its brightness relative to other astronomical objects, and its relatively small apparent radius as viewed by a spacecraft near the Earth.

The objective of a sun sensor is to provide an approximate unit vector, with respect to the body reference frame, that points towards the sun. A sun sensor determines the satellite's orientation relatively to the Sun by measuring the amount of light or shadow on it. It can be applied in its simplest form, a photodiode on each side of the satellite, which will tell which side is most likely to be towards the Sun. Another possibility for determining a sun vector is utilizing measurements of the currents from the solar panels. As there are only solar panels on five of the six sides of the satellite, a light dependent resistor, (LDR), could be placed on the unused side for determining whether it is pointing towards the sun or not.

Sun sensor can be used also to avoid the exposure of some sensitive instruments that must not aim to direct sunlight.

Analog sun sensors are based on photocells whose current output is proportional to the cosine of the angle α between the direction to the sun and the normal to the photocell. In our case, the sun sensor used is an image sensor, built with a CMOS sensor.

The way to work with the sun sensor to estimate the attitude is based on the following: if the position of the spacecraft in its orbit is known, along with the position

of the Earth in its orbit, then unit vector respect to the body frame reference is known. [3]. In our case it is not necessary, because we do not need to know the orbit parameters to estimate the attitude of the spacecraft.

The derived problem is that for sun vector acquisition (in a cubeSat) at least six photodiodes are used. The photodiodes can determine the incidence angle of the sun with respect to the normal of their plane. However, this will only provide knowledge about the cone on which the sun vector may lie. In many cases, attitude estimation is based on sun sensor, magnetometer and gyroscopes working together.

Sun sensor has the problem that it will not work in the eclipse of the orbit, which is approximately 35 minutes in a typical circular LEO. Because of this, usually a gyroscope is used as a relative attitude sensor.

4.5. Magnetometers [7]

A vector magnetometer returns a vector measurement of the Earth's magnetic field in a magnetometer-fixed reference frame. It is usually chosen with three axis measurements.

As with sun sensors, the orientation of the magnetometer frame with respect to the spacecraft body frame is determined by the designers.

When a magnetometer is used, care should be taken that no other parts have residual magnetic fields inside the satellite or near the magnetometer.

4.6. Star tracker [7]

The star tracker is by far the most accurate attitude determination system available, with accuracies down to a few thousand of a degree.

The image sensor produces an image of the stars. This image is compared with an on board catalogue of the starry sky to determine the attitude. The star tracker is, however, so heavy and big, especially the baffle needed to shield the sensor from sun,

earth, and moon shine, cover a small area of the sky, that it is infeasible for the cubeSats.

Our project is based on a star tracker, with the difference that we do not want to shield the light from Sun, Moon and Earth; on the contrary, we want to use them to estimate the attitude.

4.7. Horizon scanner [7]

The horizon scanner determines where the earth is relative to the spacecraft. Usually only two axes are determined, hence this sensor is best suited in combination with other sensors. Accurate horizon scanners are relatively expensive and big sensors, and hence not suited for the cubeSats

4.8. GPS [7]

A GPS receiver can also be used to determine the satellite attitude. By placing two antennas a distance apart from each other, and measuring the difference in carrier wave phase between the two antennas, the attitude, except for the rotation around the axis on which the two antennas is placed, can be determined with an integer ambiguity.

This requires an additional antenna and continuous measuring which obviously requires more power. Because of the problems concerning GPS receivers (disregarding space limitations or power consumption) a GPS was not taking account as attitude sensor.

Nonetheless, a GPS receiver is required on board CubeSat for the magnetometer measurement to be compared with the IGRF (International Geomagnetic Reference Field) model which uses satellite position to determine the magnetic field.

4.9. Reference Frames [3]

This section describes the definitions of the different reference frames. In order to describe the orientation and position of the satellite (both are expressed through vectors and matrices) and the mathematics behind sensor modeling, some notational defining is required. Also a vector or matrix needs a reference to be unambiguous.

4.10. Earth-Centered Inertial (ECI) Reference Frame [3]

The frame is located in the center of the earth and fixed towards the stars. The earth rotates around its z-axis. The x-axis points towards the vernal equinox, and the y-axis completes a right hand Cartesian coordinate system.

4.11. Earth-Centered Earth Fixed (ECEF) Reference Frame [3]

The frame shares its origin and z-axis with the ECI frame. The x-axis intersects the earth's surface at latitude 0° and longitude 0°. The y-axis completes the right hand system. The ECEF rotates with the earth with a constant angular velocity and is therefore not an inertial reference frame.

4.12. Orbit Reference Frame [3]

The orbit frame has its origin at the point which the spacecraft has its center. The origin rotates at an angular velocity relative to the ECI frame and has its z-axis pointed towards the center of the earth. The x-axis points in the spacecraft's direction of motion tangentially to the orbit. It is important to note that the tangent is only perpendicular to the radius vector in the case of a circular orbit.

The y-axis completes the right hand system as usual. The satellite attitude is described by roll, pitch and yaw which is the rotation around the x-, y-, and z-axis respectively.

4.13. Earth Centered Orbit Reference Frame [3]

The frame is centered at the earth's center, with x-axis towards perigee, y-axis along the semiminor-axis, and z-axis Focus perpendicular to the orbital plane to complete the right hand system.

4.14. Attitude Determination Theory [3]

Attitude determination uses a combination of sensors and mathematical models to collect vector components in the body and inertial reference frames. These components are used in one of several different algorithms to determine the attitude, typically in the form of Euler angles, a rotation matrix or a quaternion.

The attitude determination analyst needs to understand how various sensors measure the body-frame components, how mathematical models are used to determine the inertial-frame components, and how standard attitude determination algorithms are used to estimate the rotation matrix.

At least two vectors are required to determine the attitude. Recall that it takes three independent parameters to determine the attitude, and that a unit vector is actually only two parameters because of the unit vector constraint.

Therefore we require three scalars to determine the attitude. Thus the requirement is for more than one and less than two vector measurements. The attitude determination is thus unique in that one measurement is not enough, i.e., the problem is underdetermined, and two measurements is too many, i.e., the problem is overdetermined. The primary implication of this observation is that all attitude determination algorithms are really attitude estimation algorithms.

4.15. Star tracker using as a bodies reference Sun, Moon and Earth

Based on the Sun sensor hardware, we want to test whether it is possible to make a star tracker, but using the Sun, Moon and Earth as references. Sensor's viewing angle is wide enough how to use it only with the stars.

To test whether it is feasible, it performs a simulation model based on Matlab, which generates the images you would see the satellite. These image must be generated due to the difficulty to obtain real images with the parameters desired.

Based on these images we want to calculate the rotation (attitude) of the satellite.

4.16. Image simulator for model position and attitude sensor

This is a Matlab based simulation for attitude determination in NanoSatellites orbiting the Earth. Simulation is made using image sensors as unique way to determinate satellite’s attitude, without any other attitude sensor.

The parameters of the images are based in measurements done in images captured by CMOS sensors.

The model is based in cubic spacecraft (for instance, Aramis) with a set of six image sensors (one for each side). These sensors detect the celestial bodies (Moon, Sun and Earth) than will be used as reference.

The images generated by the simulator will be used such as real images captured by sensors to detect and identify the celestial bodies. Once the celestial bodies are identified, the algorithm to estimate the orientation will process them

From the images, will be obtained the position of the celestial bodies (measurement positions) than forward they will be compared with the actual position (given by celestial dynamic equations) to determinate the attitude of the satellite.

Remember that all these simulations are necessary due to the difficulty to test the hardware (real sensors) in real conditions and to have enough real data (images) to evaluate the system, calculations and results of the attitude algorithm.

Sensors (simulations) will provide images of Sun, Moon, and Earth along the satellite orbit., using the same format that the designed Sun sensor would use.

In the simulator, images sensors are able to difference the light coming from the Sun than the light coming from Earth and Moon. In the real model it is reachable, with a single sunlight detector (in each side) that must active the filters when the sun illuminates the side (to avoid saturation in the sensor) and take out the filters when light coming from other sources (moon and earth) to the image sensor.

4.17. Mathematical Model

The mathematical model used is based in a matrix rotation applied to an object in a three dimensional space. This rotation matrix gives us the elements to get the new representation of the rotated object and it is generated by the rotation angle in each axis.

So, in our case, the attitude of a satellite will be represented by a rotation matrix defined by Euler angles, corresponding to series of positive right hand rotations.

For instance, if a 1-2-3 Euler rotation is used, the rotation matrix will be:

$$Q = \begin{bmatrix} C\theta C\psi & C\theta S\psi & S\theta \\ S\phi S\theta C\psi + C\phi S\psi & -S\phi S\theta S\psi + C\phi C\psi & -S\phi C\theta \\ -C\phi S\theta C\psi + S\phi S\psi & C\phi S\theta S\psi + S\phi C\psi & C\phi C\theta \end{bmatrix} \qquad \text{Equation 9}$$

Where: ϕ, θ, and ψ are the Euler angles and C and S mean Cosine and Sine, for instance, the representation $C\theta = \cos\theta$, and $S\Psi = \sin\Psi$.

In the method that we will use to estimate the attitude of the satellite, two set of three vectors will be necessary. The first one will be given by measurements and observations of the celestial bodies made by the ADCS (in our case the image sensor).

The other set of three vectors corresponding, will be taken from the calculated position of the celestial bodies. These sets of vectors are related by a unknown rotation matrix, because the objects describes by the vectors are the same, but reference-axis are different.

In this way, if the first set is denoted as: $\boldsymbol{m_1}$, $\boldsymbol{m_2}$, and $\boldsymbol{m_3}$ (measurement vectors). The second set of vectors can be denoted as: $\boldsymbol{c_1}$, $\boldsymbol{c_2}$, and $\boldsymbol{c_3}$ (calculated vectors). So the relationship between the corresponding vectors is given by $\boldsymbol{Q}$, that is the rotation matrix. So, we can express the relationship between each vector as is shown in Equation 10.

$$m_i = Q \cdot c_i \qquad \text{Equation 10}$$

With $\boldsymbol{i=1, 2, 3}$.

So the relationship between the two set of vectors is defined in Equation 11.

$$M = Q \cdot C \qquad \text{Equation 11}$$

So, if we know the rotation matrix, we could know the rotation angles. Starting multiplying by the inverse matrix of calculated position (if it is possible):

$$M \cdot C^{-1} = Q \cdot C \cdot C^{-1} \qquad \text{Equation 12}$$

So, to solve the Equation 12, we can determinate the rotation matrix, as it is shown in Equation 13.

$$Q = M \cdot C^{-1} \qquad \text{Equation 13}$$

Our three vectors will be given by the positions of the three celestial bodies, Sun, Earth and Moon. Last sentence could indicated that the three objects must be visible at the same time, however only two vectors given for the objects will be necessary to calculate the rotation matrix, it means, that only two of the three celestial bodies must be visible at the same time to determine the rotation matrix.

The third vector can be obtained by the cross product between the two first vectors, where the first two vectors can be normalized to work as unit vectors. So finally, the set of vectors to calculate the rotation matrix are shown in Equation 14 and Equation 15:

$$\hat{m}_1 = \frac{m_1}{|m_1|},\ \hat{m}_2 = \frac{m_2}{|m_2|},\ \hat{m}_3 = \hat{m}_1 \times \hat{m}_2 \qquad \text{Equation 14}$$

$$\hat{c}_1 = \frac{c_1}{|c_1|},\ \hat{c}_2 = \frac{c_2}{|c_2|},\ \hat{c}_3 = \hat{c}_1 \times \hat{c}_2 \qquad \text{Equation 15}$$

4.18. Physical Model

Mathematically was proven that is possible to get the rotation matrix from a set of vectors. Now the physical model must be defined. To start, the frame of reference used will be with the center of coordinates locates in the center of the Earth, (ECEF – Earth center fixed reference). So, the position of each body is referenced to the center of mass of the Earth, the point (0,0,0).

Although the dynamic system composed by the Sun, Earth and Moon have elliptical orbital paths, to simplify the simulation model, orbits will be considered such circular paths, which implies that the distance between the bodies will be constant; for which distance is taken as the average distance between objects.

In the spacecraft, the frame reference used is thought to define an initial rotation, where each side is labeled (for instance, +X, +Y, +Z, and in the opposite side, -X, -Y, -Z, respectively), keeping the axis orientation; so the vectors that aim to the celestial bodies can be expressed unequivocally in each side. Azimuth angle (angle from X-axis perpendicular to Z-axis) is taken positive in direction X-axis to Y-axis. The elevation angle is positive in the direction X-Y plane to Z-axis. It is shown in Figure 21

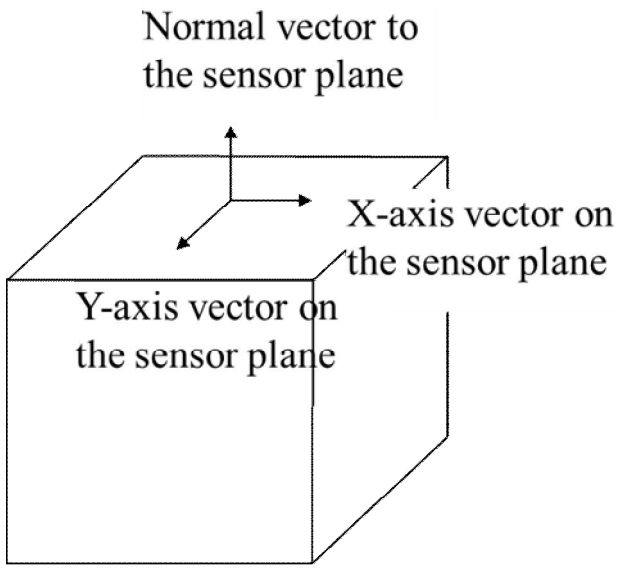

Figure 21. Vectors to defined a point on the satellite face

Initially -X side in the satellite is considered that faces to the Earth. Then, it will not keep necessarily through time, due to the dynamics of the spacecraft; however, the initial reference was already established and the rotation could be determined.

So, the reference frame uses these definitions:

+X direction is referred to 0° azimuth (with no initial rotation on the satellite), with the face looking outside the Earth

-X direction is 180° azimuth, it means, face satellite aims to the Earth (with no initial rotation on the satellite).

4.19. Creating Objects

There are only four objects in the model, these are: Earth, Moon, Sun and satellite. The first three objects are visible, but the fourth one (the satellite) is which register the images coming from the space with the three objects.

To create the three visible objects, the same spherical surface model will be used, but with different parameters. For example, Earth will have more points than Moon, due to the body's size in the sensor.

Spherical surface is divided in several segments (1000 for Earth and 100 for Sun and Earth) in horizontal and vertical direction. So, Earth will have near to 1'000.000 of points to compose the surface, instead, Sun and Moon will have near to 10.000 points on their surfaces. An example of as to generate the object is shown in Figure 22

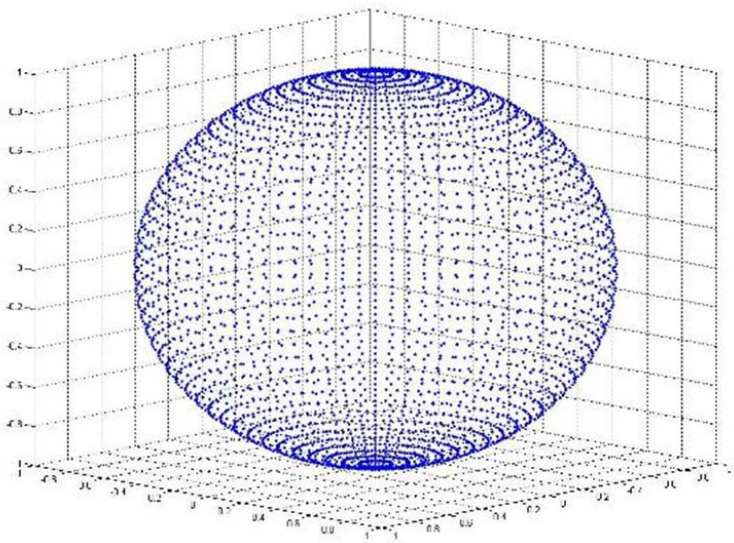

Figure 22. Distributed points on the surface of a spherical object

4.20. Simulation Parameters

The position of each object on the space (celestial bodies and satellite) is determined by three values: Distance (fixed for each object), Azimuth Angle and Elevation angle, as were already mentioned and explained previously. From these parameters, only azimuth and elevation angles must be calculated by the algorithm to determine the position of the object.

In the program simulator some parameters must be defined, before to start the simulation, theoretically, they are not going to change. Some of these are:

- Earth radius average. (Value used in the simulator 6378 x 10^3 m)
- Moon radius average. (Value used in the simulator 1738 x 10^3 m)
- Moon-Earth distance average. (Value used in the simulator 388,4 x 10^6 m)
- Sun-Earth distance average. (Value used in the simulator 149,6 x 10^6 m)
- Sun radius average. (Value used in the simulator 696 x 10^6 m)
- Height of the satellite orbit. (Value used in the simulator 800 x 10^3 m)
- Terrestrial albedo (coefficient reflection). (Value used in the simulator 0,36)
- Moon albedo. (Value used in the simulator 0,36).
- Solar constant. (Value used in the simulator 1366 W/m^2)
- Horizontal pixels on the image sensor. (Value used in the simulator 640 pixels)
- Vertical pixels on the image sensor. (Value used in the simulator 480 pixels)
- Pin-hole diameter. (Value used in the simulator 25 pixels. -100 µm-)
- Horizontal field of view. (Value used in the simulator 120°)
- Vertical field of view. (Value used in the simulator 90°)
- Satellite altitude. (Value used in the simulator 800 x 10^3 m)

There are other parameters that are able to be changed before to start the simulation, these are:

- Satellite orbit, it will be defined by the *inclination angle* and *angle step* around the Earth.
- Sun and Moon are located by *azimuth* and *elevation angles* for each one.

- Once the objects are located on the space, also the orientation of the satellite must specify (initial attitude)

Finally, after the given conditions and setting parameters in the simulator, images on each sensor in each side of the satellite will generated. So, now, the orientation would be calculated using matrices of vector measurement (M) and vector corresponding (C).

4.21. Simulation Procedure. Sun Light On Image Sensor

Every point of the near to one million on the Sun surface is evaluated with this procedure to determine how much light arrives to the sensors:

i. As the body has a spherical surface, only a half hemisphere can send light to the Earth (satellite). So first must be chosen the points that are visible from the Earth on the Sun surface. This decision is done using internal product between the Sun vector position (distance, azimuth and elevation angles) and vector pointed from the Sun center to the Sun surface point. After this step, half of the points on the surface of the Sun are discarded.

ii. Half remained points on the sun surface are evaluated if they are visible from the satellite. It is evaluated because Earth can hide some Sun points. Hidden lights produced by Moon are not taken. This evaluation is done with the internal product between the Sun and the satellite, so if the satellite is behind the earth, light coming from the Sun will not hit the sensors on the satellite..

iii. If there are some remained points, they are evaluated to know in which sides on the satellite they are visible. So each face of the satellite is evaluated with the internal product to know where the Sun is visible. It is done using internal product between first components of the

representation matrix of the each side (normal vector to the satellite face).

iv. Finally, when the visible points from the specific face of the satellite are chosen, they are evaluated between the field of view (FOV) of the sensor. If the point is outside of FOV, they do not count. The points into FOV, will be taken to generated the simulated images

4.22. Simulation Procedure. Moon Light and Earth On Image Sensor

This procedure is similar to it done with the points from the Sun surface; but in this case, there is an extra step that must be added, because these points from the Moon surface reflected the light coming from the Sun but they do not emit.

- First, the points on the Moon surface are evaluated if they receive the light emitted by the Sun. It is done with the internal product between the Sun vector and Moon surface point vector.
- After define the points that reflect the light, the next steps are similar to those done in the Sun points.

Finally, light coming from the Earth to the images sensor has the same procedure than Moon light does.

4.23. Image Simulated

Once the points on the surface‘s bodies that impact the image sensor are defined, each point is passed through a diffraction pattern to simulate the diffraction produce by the pin-hole, but also to apply a kind of renderization to the image.

The diffraction pattern used is a Gaussian curve (two-dimensions).

All the points in the images are added with the respective power (it means, light coming from the Sun are stronger than the light coming from earth and Moon), however, later these points are normalized to have all the information in the images.

After the normalization of light intensity, Sun and Moon will have a similar size viewed from the Earth, in ur case form the satellite. To avoid any misunderstood in the image analysis; Sun is simulated a little bit bigger than the Moon, but it does not affect the image analysis..

4.24. Image Visualization

Six faces images of the satellite are shown at the same time in the simulator.

For instance, in Figure 23, the six simulated images are shown.

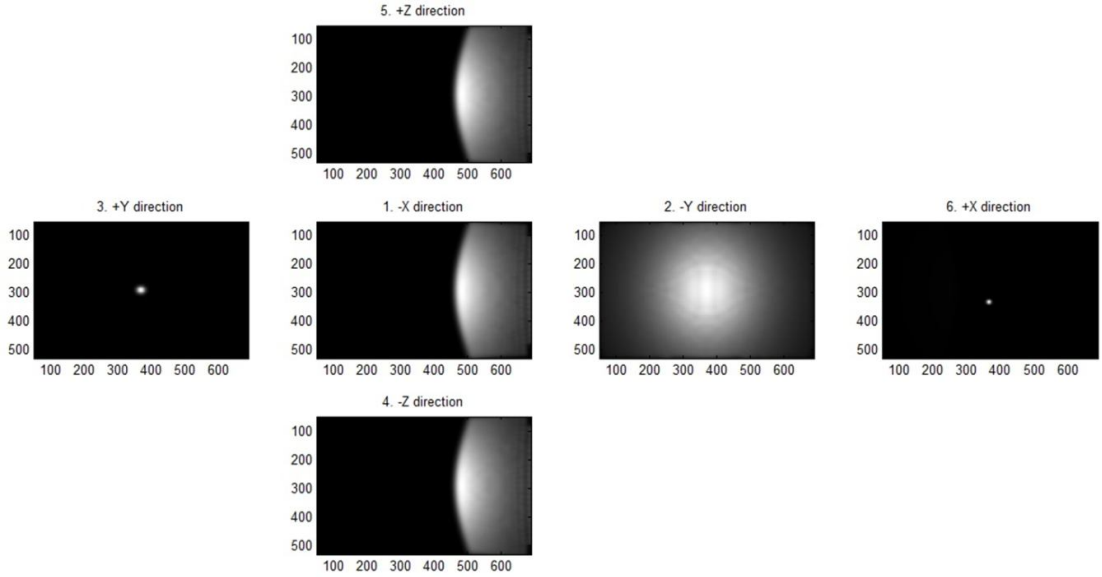

Figure 23. Image simulated, with visualization of Sun, Moon and Earth.

Images in Figure 23 were generated with the follow parameters:

Sun azimuth = 90°	Moon elevation = -10°	Satellite rotation on Z = 0°
Sun elevation = 0°	Satellite rotation on X = 0°	Satellite azimuth = 90°
Moon azimuth = 0°	Satellite rotation on Y = 0°	Satellite orbit angle = 0°

Comparing the parameters given, with the images, we can check the position of the celestial bodies. For instance, Sun in at the center of the image, it is compatible with the position parameters: 0° elevation, 90° azimuth. The satellite is located in orbit inclination 0°, azimuth angle 90°, so the Sun must appear in +Y side. Also occurs with the Moon, that is located 0° azimuth and -10° elevation angle, so this must be seen in +X side, a little bit lower than the horizontal.

Earth is viewed frontally from –Y side, because the satellite is located 90° azimuth. Also some parts form the earth are viewed in –X, -Z and +Z. Some part of the Earth also must be viewed in +X, but with the normalization, this part of the Earth will not appear.

To have another example of images simulated, in the Figure 24, there are in a sequence of six images but the orbit of the satellite is different. The parameters used this time in the Figure 24 were:

Sun azimuth = 90°	Moon elevation = -10°	Satellite rotation on Z = 0°
Sun elevation = 0°	Satellite rotation on X = 0°	Satellite azimuth = 90°
Moon azimuth = 0°	Satellite rotation on Y = 0°	Satellite orbit angle = 15°

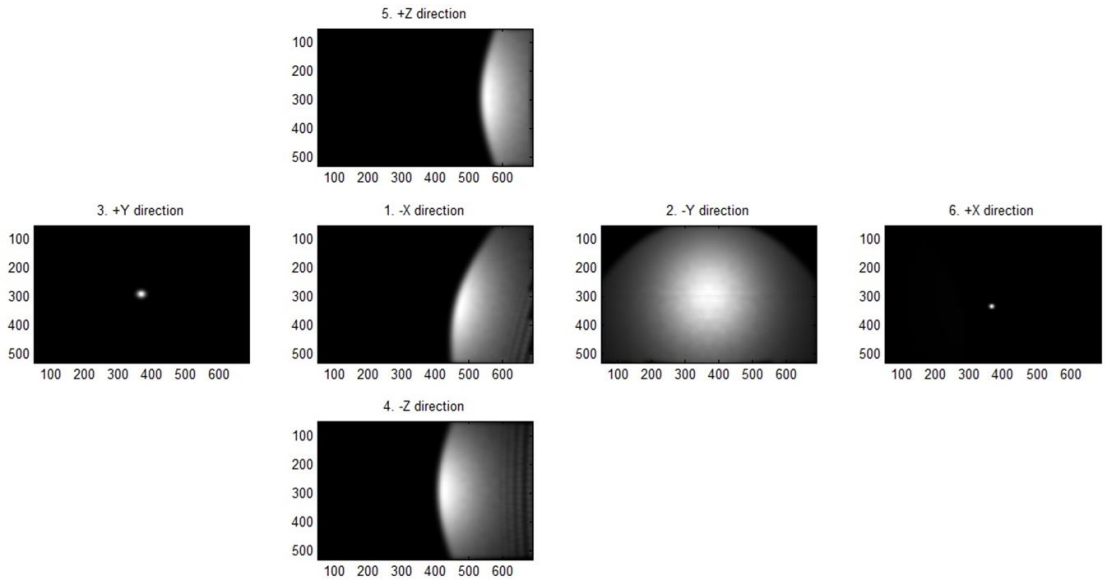

Figure 24. Image simulated, with visualization of Sun, Moon and Earth with satellite rotation

A simple sight, the changed more notorious is in the Earth, as the satellite is higher respect to the equatorial reference, so, the Earth is more visible in the –Z side of the satellite. However, there is not a notorious change in the Moon or Sun position.

Although the framework of the model is the center of the earth, the framework for the images should be the center of the satellite. Therefore it is introducing an error due to the two references; however the error is negligible for bodies located far away from the earth, therefore the satellite, such as the sun. For celestial objects closest to the earth as the moon, there is an error that can become up to 2°. This error introduces an error in the final result about attitude in the satellite.

Several images were generating to test the systems in different conditions and with different parameters. Some of these results are shown in results and conclusions items forward in this document.

4.25. Image Processing

As was written previously, at least two bodies in the images must be identified and located to estimate the rotation matrix. First step once the images are generated is detecting the celestial bodies. The easiest body to detect is the sun, due to its power illumination. Earth can be detected by its size and moon, as a combination of size and power illumination. When the Sun is in an image, the other object does not have enough energy to be visible.

Once the celestial bodies are identified in each image, to each one must be calculated its center. This can be done, calculating the center of mass of the image when moon or sun is in the images. To calculate the center of the Earth from the images, the transition between day and night must be detected and with this the center of the Earth.

A complete result can be obtained using only Sun and Moon. Earth will be a supplementary object when Sun or Moon is hidden. Because Sun or Moon process is faster and simpler than detect the center or the Earth.

The first images analyzed are which have the sun. If there are more than one, the image with the sun closer to the center will be taken as a primary reference. Once the sensor with sunlight is identified and then it is evaluated to find the center of the celestial body. The images without sunlight are evaluated to detect moonlight. Also the Moon is not unique in the set of six images, because it can appear in more than one image, so the image with the object closer to the center will be evaluated.

Once Sun and Moon are found in the images, the centers of two objects are calculated using center of mass. These positions on the images (X-Y values) have a spatial vector coordinates normalized. With this information, attitude determination method is applied.

4.26. Results

Here the results of the whole process are summarized.

Resolution of Sun sensor was enough even if the measurement is done each minute.

Full space localization for celestial bodies, full rotation satellite and free orbit rotation can be simulated, to have different theoretical cases and to create discriminated data bases for mathematical analysis.

At the end of each simulation, we have six images (one for each sensor in the face) and also we have the data set and evaluated parameters

Sun, Moon and Earth positions can be calculated from images, using only image sensors, then with this information and real positions of the celestial bodies, satellite attitude can be estimated, as it was wanted at the beginning.

All measurement positions of the celestial bodies have an error due that the center of coordinates is the earth, but the measurements are made respect to the satellite. Sun error is negligible, but moon error is appreciable, (more than 1°) when the satellite is higher than 45° respect to the ecliptic line. It occurs because the distance between Satellite and Earth respect to Sun is much larger than to the Moon.

There is an image error position that is produced by pixel quantization. Doing the calculations even taking the errors due to the different frames of reference and pixel quantization, the final attitude error angle is less than 1° in the each axis. This error can be high in some applications; however, it can be improved after iterations.

Finally, attitude in a spacecraft can be estimated using only image sensors

4.27. Conclusions

Simulator was successful to recreate four spatial objects (Sun, Moon, Earth and satellite) to have images simulated, with image sensors.

First object to detect must be the Sun, because it is unique. There is not another object with the same power light. It is done also as in the real life using neutral density filters.

Normalization of the power light emitted or received on the sensor simulates optic filters is used in the real image sensors, due to dynamic range of the sensor.

Image sensors could be used to estimate satellite attitude, detecting two objects between these three references (Sun, Moon and Earth).

Attitude estimator based on image sensor must have always at least two objects visible form the satellite.

There is not necessary to know the orbit of the satellite previously to estimate the attitude with image sensors, as it is necessary using a sun sensor with a magnetometer.

Error in the attitude angles must be compared using another mathematical model, as quaternion, to have another reference. Also a Kalman filter could be included to have better results.

Earth center algorithm estimator must be improved to use when Moon or Sun is hidden. Also, Earth center algorithm could be used to calculated satellite position.

Physical model of the image sensor designed and the simulation model are compatibles without to have big changes.

Bibliography

[1] Y. Winetraub, S. Bitan, A. Heller. Attitude Determination – Advanced Sun Sensors for Pico-satellites. Handasaim School, Tel-Aviv University, Isarael.

[2] E. Oland, A. Aas, T. Steihaug, S. Mathisen, F. Vedal. "CubeSat Design Specification. The CubeSat Program", Cal Poly SLO Rev 12. California Polytechnic State University

[3] C. Hall. Spacecraft Attitude Dynamics and Control. January 12, 2003. http://www.dept.aoe.vt.edu/~cdhall/courses/aoe4140/.

[4] W. Larson, J. Wertz. Space Mission Analysis and Design. 3rd Edtion. October 1999. ISBN-13. 978-188188104.

[5] V. Pisacane, R. Moore. Fundamentals of Space Systems. Oxford University Press. Oxford 1994

[6] M. Side. Spacecraft Dynamics and Control: A Practical enginnergind Approach. Cambridge University Press, Cambridge, 1997

[7] K. Svartveit. Attitude determination of the NCUBE satellite. Department of Engineering Cybernetics. June, 2003

[8] California State Polytechnic University. CubeSat Design Specification Rev. 12".. http://www.cubesat.org/images/developers/cds_rev12.pdf.